INTERNATIONAL REFERENCE GUIDE TO SPACE LAUNCH SYSTEMS

Fourth Edition




Steven J. Isakowitz
Joshua B. Hopkins
Joseph P. Hopkins Jr.

AIAA

INTERNATIONAL REFERENCE GUIDE TO SPACE LAUNCH SYSTEMS

FOURTH EDITION

STEVEN J. ISAKOWITZ • JOSHUA B. HOPKINS • JOSEPH P. HOPKINS JR.

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H-IIA, Dnepr, Athena, and Minotaur

Back cover, from left:
GSLV, Volna, Atlas V, and Delta IV

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Athena launch from Kodiak Island, Alaska

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FOREWORD

It was the best of times, it was the worst of times.
—Charles Dickens, A Tale of Two Cities

As we go to press with the fourth edition of the *International Reference Guide to Space Launch Systems*, the industry faces an uncertain future. The loss of the Space Shuttle *Columbia* has set back the civil space program and prompted plans to retire the remainder of the fleet upon finishing assembly of the International Space Station. The Titan launch vehicle approaches its last launch after decades of rocketing military payloads into space. Space agencies in Europe, Japan, and Brazil have struggled with serious launch failures of their own. The commercial market that only years ago seemed on the verge of filling the night sky with satellites has fallen flat as terrestrial networks have filled the market void. All of these setbacks have occurred since the last edition of this guide was published in 1999.

On the other hand, NASA has embraced a new vision for space exploration that calls for a return of humans to the moon and promises to dramatically increase the demand for affordable and reliable access to space. The Department of Defense is helping to bring forward a new generation of heavy-lift expendable launch vehicles. Similarly, foreign space organizations are also enhancing the capacity of their own rockets. Others are entering the club for human access to space. The Chinese have launched their first astronaut into orbit and created an instant national hero. A privately financed rocket plane designed by noted aerospace engineer Burt Rutan has lofted the first non-government astronaut into suborbital space. Other emerging U.S. companies show great promise at filling a niche for affordable launch. The entrance of these new players has sparked real excitement about the possibilities for innovation and breakthroughs in the launch market.

Such dynamic shift in the marketplace makes it even more important to have a reliable source of information for individuals who want to understand the launch industry. That is where the *International Reference Guide* comes in. In one book, the reader can find a succinct summary of the design and capabilities of existing launch vehicles. Such information helps to understand investments made to date, the diversity of choices that exist for buyers, and a benchmark for future vehicles.

I am proud to be associated with individuals who find that creating such a source of information is a must. Those individuals are Josh Hopkins and Joe Hopkins who for the second consecutive edition have made a major personal commitment to make this guide a reality. A great deal of thanks is owed to Josh and Joe for completing this fourth edition. It is no small effort. Much time is needed to update the information in this guide through communicating with source experts, ensuring consistency in the formats, and tirelessly editing the technical material for quality and ease of use. I know they have undertaken this effort because they are committed to the promise of space travel and this guide's role in facilitating the undertaking. I hope you, the reader, will find this book as indispensable as I do in keeping abreast of the industry – with all its ups and downs.

Steven J. Isakowitz
July 2004

ACKNOWLEDGMENTS

The members of the AIAA Space Transportation Technical Committee (STTC) take great pleasure in nurturing and supporting something as substantial and useful within our industry as the *International Reference Guide to Space Launch Systems*. The quality of this product shows the level of cooperation possible in the space transportation arena when the product benefits the industry as a whole and politics can be avoided. Special acknowledgments are also owed to Lockheed Martin Corporation and SpaceWorks Engineering who financially supported much of the editorial work. The STTC also acknowledges the early commitment of our colleague John Olds and his corporation SpaceWorks Engineering. This support made the improvement of key artwork possible.

AIAA Space Transportation Technical Committee

A book like this would not be possible without the help of a large number of individuals who provided information, support, and suggestions. Listed are the organizations and individuals who provided technical data, artwork, and other support for this edition. Several people deserve special thanks. The members of the AIAA Space Transportation Technical Committee, particularly Jason Andrews and Wayne VanLerberghe, provided the impetus to update the book and helped us receive the various approvals which were required for publication. The professional staff in the AIAA publications department, Rodger Williams and Jennifer Stover, provided invaluable support in data gathering, layout, and publishing. Jonathan McDowell provided his outstanding launch logs as a primary reference for the enhanced launch records in this edition, as well as other useful reference material. Dennis Jenkins provided artwork for the Space Shuttle chapter. We are also indebted to the friends and colleagues who provided encouragement and support while this edition was being developed. Josh Hopkins would like to thank Al Simpson and Larry Price. Joe Hopkins thanks Dana Andrews, Marian Joh, Livingston Holder, and Curtis Gifford.

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For my wife, Monica, and children, Matthew, Jennifer, Rachel, and Sophie.
—Steven J. Isakowitz

For Amy, Alex, and Mary.
—Joshua B. Hopkins

For Todd Hawley.
—Joseph P. Hopkins Jr.

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We can lick gravity but sometimes the paperwork is overwhelming.
—Werner von Braun

Introduction

When the first edition of this book was published in 1991, only 18 launch vehicle families were operational, with most activity centered in the United States and the secretive Soviet Union. Europe had just begun operating the new Ariane 4, which would come to dominate the commercial launch market. Since that time, sweeping changes have been documented in subsequent editions of this book. The opening of the former Soviet space industry; the commercialization of the launch market; the globalization of the launch business; the boom of the late 1990s and the subsequent bust. This edition, the fourth, documents the state of space launch systems industry through the end of 2003. The past few years have been another period of transition. The number of active launch systems has increased by one-third since 1991, but the total number of launches has declined, leaving many systems idle. In the United States the newly designed Atlas V and Delta IV have become operational, heralding the end of the older missile-derived Atlas, Delta, and Titan lines that have performed most American launches since the opening of the space age. The highly successful Ariane 4 has been retired and replaced by the Ariane 5. China has launched its first citizen into space, while India has developed a capability to launch satellites into geostationary orbit for the first time.

Despite the changes, the function of this guide remains the same: to provide an authoritative reference with readily accessible information in a standard format that is useful for engineers, policy-makers, students, planners, insurers, managers, and those who simply find launch systems to be interesting. Previous editions of this book have been used to purchase launch services, design new launch systems, prevent typical causes of launch failures, and even to regulate corporate mergers.

As noted in the title, these launch vehicles are described as "systems." Launch systems include not only the launch vehicle hardware itself, but also the whole process that makes up a successful launch: production, assembly facilities, and operations plus historical, programmatic, and organizational information. Thirty-one launch system families are included in this guide. They include 25 vehicle families that are active – i.e. they have performed a launch attempt since the previous edition of this guide was released in 1999 – and six developmental vehicles that are planned for launch before the end of 2006. Inclusion or exclusion of vehicles in development should not be interpreted to reflect the opinion of the editors or AIAA as to the credibility of proposed systems. Inclusion was determined only by the planned launch date and the availability of sufficient data to compile a chapter. Four new vehicle families have been added to this edition. All are small launch vehicles: the Chinese Kaituozhe, the European Vega, and the U.S. Falcon and Scorpius. Two vehicle families have been removed from the book. Beal Aerospace cancelled its BA-2 launcher in late 2000. Japan also cancelled its J-1 small launch vehicle program in 2000. Neither had performed an orbital launch.

Information for this updated edition was collected from a variety of sources. Copies of the previous edition were provided to the organizations responsible for each vehicle to solicit their updates and comments. That information was combined with data from a variety of public sources including user's guides, news reports, technical papers, and financial statements. The revised chapters were then checked again by the responsible organizations and by outside reviewers. While every effort has been made to verify the accuracy of data presented herein, errors may still occur, and some information may vary depending on application or change over time. Launch service providers can also change vehicle capabilities and interfaces to meet user needs. It is therefore highly recommended that readers who require accurate information for decision making purposes contact the responsible organization listed on the first page of each vehicle chapter.

Additional information or corrections will be included in future editions of this guide. Corrections, suggestions or inquiries are welcome and should be addressed to

International Reference Guide to Space Launch Systems
c/o AIAA Technical Publications
1801 Alexander Bell Drive, Suite 500
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How to Use This Book

To ensure quick and easy retrieval of data and to simplify comparison of different launch systems, this guide uses a standard format for each launch system. Maximum use of a tabular format with numerical data has been emphasized. Text is used to add information that numbers alone cannot convey.

Data in this edition are presented in the international standard SI units, with nonmetric units usually accompanying the text within parentheses. Dates are provided based on Universal Coordinated Time (UCT). The following symbols are used frequently:

k = thousand

M = million

? = information not available or data shown is uncertain

— = not applicable, not present

The standard elements of each chapter are:

Cover Page

An illustration or photograph of the launch vehicle, with address and contact information for one or more organizations responsible for the launch system and its marketing.

General Description

A one-page summary of each launch system, including a brief description of the vehicle, responsible organizations, launch cost, performance, and available launch sites. The typical flight rate is listed, based on 1999–2002 for existing systems or reflecting the planned rate for systems that are not yet operational.

How to Use This Book

Nomenclature

This provides a description and explanation of the various designations used for the launch system. Nomenclature for Russian and Ukrainian launch systems is described in an additional appendix. In cases where the launch system name is in a language other than English, the spelling used follows the convention used by the launch system operator when communicating to English-speaking readers. Alternative spellings (e.g., Chang Zeng for Long March) are also discussed in this section.

Cost

Launch service price or cost information was requested from the responsible organization for each launch system. If the information was not provided, an estimated cost or price range is provided by the Office of the Federal Aviation Administration Associate Administrator for Commercial Space Transportation based on open source data. In some cases launch prices are too variable or too uncertain to provide an estimate, in which case only 'price negotiable' is indicated. However, it should be understood that the price of any launch vehicle may be negotiated. In most cases the launch service provider had an opportunity to review the values shown and to comment if they believed the data to be inaccurate. In each case, the source of the data is listed. Prices and costs are listed using the currency in which the value was originally quoted. A conversion to U.S. dollars is shown using year-end 2002 exchange rates if the values were not quoted in dollars, but the reader may wish to recalculate conversions as exchange rates fluctuate.

Launch service prices depend on mission specific services and options, the terms and conditions of the contract (such as payment schedule, insurance, etc.), market conditions at the time of purchase, and a variety of other factors. Therefore, price ranges shown should be considered as approximate values only, and readers must contact the responsible business development organization directly for actual price estimates.

Availability

Information on the commercial availability, initial launch date, and flight rate of the launch system is provided in this section.

Performance

This section includes standardized performance charts, with text discussing issues that affect performance capabilities.

Flight History

A flight history bar graph indicates the number of launches and their results for each year, with the bottom of each bar being the earliest launch for a given year. A table provides data on each orbital launch since 1979. Earlier launches are included if space permits.

The following codes are used in the Flight History Table.

Launch Results:

No code = Success: The spacecraft was deployed safely in its intended orbit.

P = Launch vehicle partial failure: The flight was not successful, but the spacecraft was still of use to its owner. For example, spacecraft may be delivered to an incorrect orbit from which some planned tasks can be performed or from which the spacecraft can reach the correct orbit using onboard propulsion.

F = Launch vehicle failure: The spacecraft was not deployed in its planned orbit and is not useful to original owner. The spacecraft may be destroyed or may be deployed in a useless orbit.

S = Spacecraft or upper-stage anomaly: Despite a successful launch, the spacecraft failed to become operational because of a failure of the spacecraft itself or an upper stage that is not part of the launch vehicle. These events are documented to indicate that the failure was not caused by the launch vehicle. This list of spacecraft failures is not complete as it is outside the scope of this book.

Descriptions of each failure are provided in a table following the launch record.

Manifest code:

When more than one payload is launched, the means of manifesting are identified when possible.

No code = Single primary payload: Most space launches carry a single payload that determines the schedule and destination for the launch.

C = Comanifest primary payloads: A comanifest launch splits the cost of launch between two or more different primary payloads, usually for different customers. The launch schedule and delivery orbit is selected to meet the needs of both customers.

M = Multiple Manifest: A multiple manifest launch carries two or more identical payloads for a single customer. These typically support constellations with many communications or navigation satellites.

A = Auxiliary payloads: Also called secondary or "piggyback" payloads, these are generally small spacecraft that are launched on a noninterference basis with the primary payload(s), making use of surplus volume and performance capacity. Auxiliary payloads generally pay a minimal launch fee, but have little control over the destination or schedule of the launch.

Vehicle Designation:

Where available, the designation for each flight is given. This may be a vehicle number (e.g., a serial number tied to production sequence) or a flight number (relating to the flight sequence).

How to Use This Book

Payload Designation:

The payload designation code is usually the unique COSPAR identifier assigned to all objects that reach orbit. The COSPAR designator includes the year of launch, a number indicating the order of the launch within the year, and a letter for each distinct object orbited as a result of the launch. Launches that fail to reach orbit are not given COSPAR designations. Jonathan McDowell, an astrophysicst who is widely known for publicizing the launch of recent space objects, has therefore extended the designation system by indicating with an F launches that failed to reach orbit. Thus, the third launch failure in 1988 would be indicated 1988 F03. This edition expands upon McDowell's system by adding a letter designation to each payload that was expected to orbit to enable unique identification of each payload. Thus, individual payloads would be labeled 1998 F03A, 1998 F03B, etc.

Orbit:

LEO = Low Earth Orbit: Typically less than 2000 km (1000 nmi) altitude

STA = Space Station Orbit : A 51.6-deg orbit typically used by Mir and the International Space Station

SSO = Sun-Synchronous Orbit: Retrograde orbit used for Earth observation

MEO = Medium Earth Orbit: Orbits between LEO and GEO altitudes

GTO = Geosynchronous Transfer Orbit: Elliptical orbit used to reach GEO. Subsynchronous (GTO–) or supersynchronous (GTO+) transfer orbits are indicated when known.

GEO = Geosynchronous and Geostationary Earth Orbits: Standard orbit for communications satellites

EEO = Elliptical Earth Orbit: Elliptical orbits that cross at least one altitude category

Missions that leave Earth orbit list the destination (such as Mars, the moon, etc.)

Market:

The market identifies the payload/satellite owner by the market it represents. These classifications indicate the way the satellite is used, not the way the launch services were procured. Some payloads serve more than one market or are ambiguous; the market is indicated based on the editors' best judgment.

CIV—These are government missions that are civilian, not military in nature. There exists some ambiguity regarding the annotation of Soviet/Russian and Chinese missions; these reflect the editors' best judgement.

CML—These are missions that have a commercial purpose. This classification includes satellites launched by commercial companies as well as national post, telephone, and telegraph agencies that primarily have a commercial telecommunication purpose.

MIL—These are government military missions. Often military missions are classified, and little is publicly available describing mission details. There exists some ambiguity regarding the annotation of Soviet/Russian and Chinese missions; these reflect the editors' best judgement.

NGO—These are nongovernmental organization missions, which are neither commercial nor government in nature. The sponsors may be nonprofit organizations, universities, colleges or schools (including state-funded learning institutions), or amateur organizations such as AMSAT.

Vehicle Design

This section comprises a diagram of the overall launch vehicle configuration, text and tabular data on each stage and its propulsion system, descriptions of attitude control system and avionics, and a diagram of payload fairing(s).

Payload Accommodations

This section provides data pertinent to the payload such as the payload envelope, launch window constraints, flight environments, the injection accuracy, and the availability of comanifest or auxiliary payload opportunities.

Production and Launch Operations

Descriptions of production, diagrams and descriptions of launch site and facilities, launch processing, and flight sequence are included in this section. Launch sites supporting multiple systems are also described in the Spaceports chapter.

Vehicle Upgrade Plans

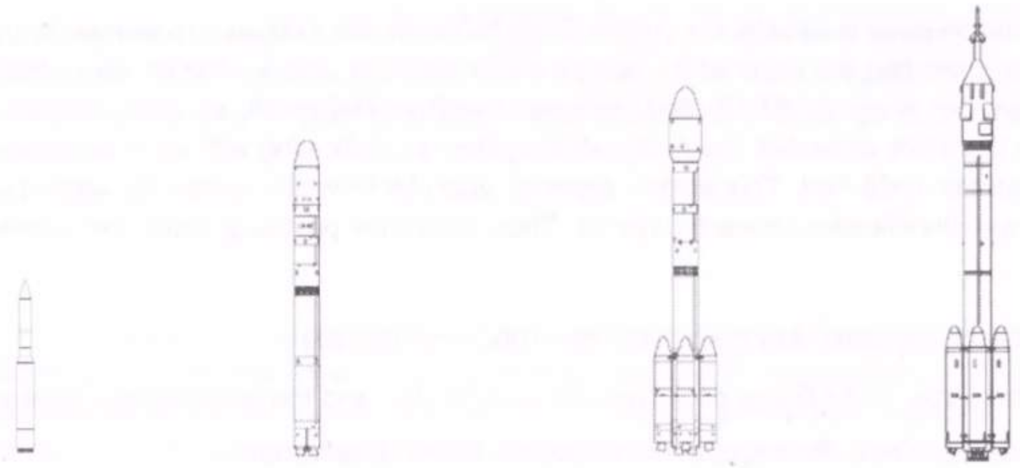
This section is a description of future vehicle plans.

Vehicle History

This section includes diagrams showing vehicle evolution, descriptions of historical vehicle configurations, a summary of launch system history, and background.

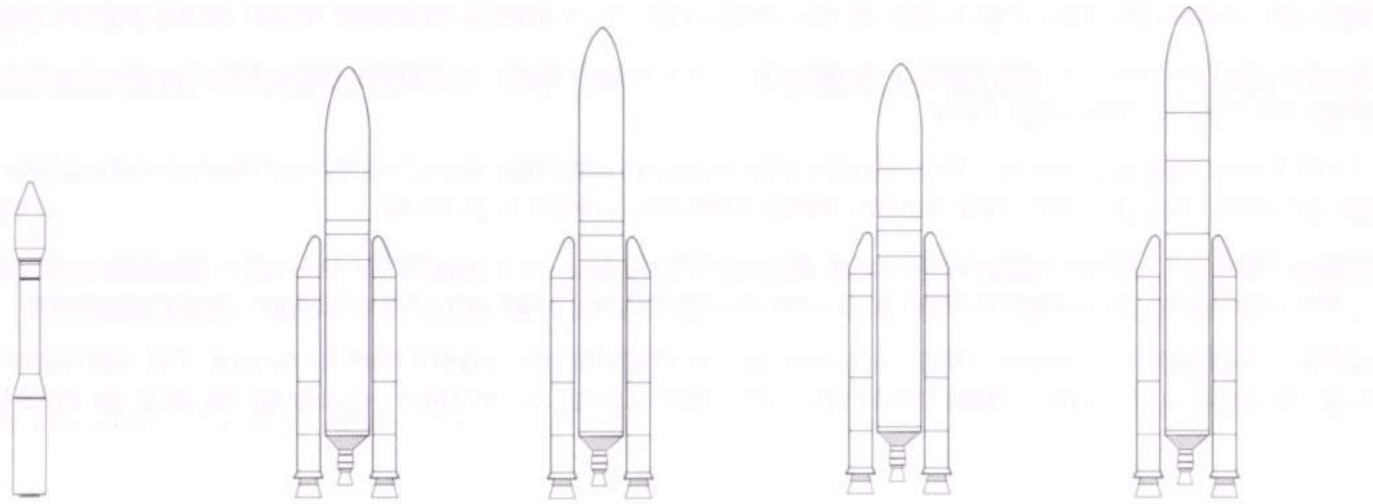
Brazil

China



Vehicle	VLS-1	VLM	Kaituozhe KT-1	LM-2C, 2C/SD, 2C/CTS	LM-2E, 2E/ETS	LM-2F
Performance						
LEO Maximum	380 kg (840 lbm)	100 kg (200 lbm)	?	4400 kg (8800 lbm)	9500 kg (20,950 lbm)	?
SSO	80 kg (175 lbm)	18 kg (40 lbm)	?	1600 kg (3525 lbm)	?	?
GTO	—	—	—	1400 kg (3100 lbm)	3500 kg (7710 lbm)	—
Cost	$8 million	$4 million	?	$20–25 million	?	?
First Flight	1997	TBD	2002	1975	1990	1999
Launch Site(s)	Alcântara	Alcântara	Taiyuan	Taiyuan Jiuquan Xichang	Xichang Jiuquan	Jiuquan

Europe



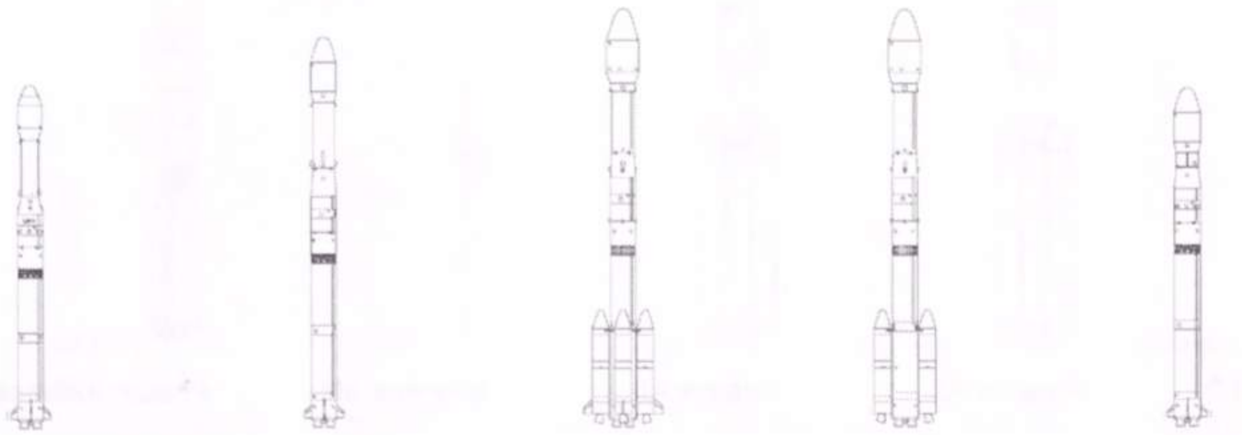
Vehicle	Vega	Ariane 5G	Ariane 5ECA	Ariane 5ES	Ariane 5ECB
Performance					
LEO Maximum	?	?	?	?	?
SSO	1395 kg (3075 lbm)	9500 kg (20,950 lbm)	?	?	?
GTO	—	6700 kg (14,770 lbm)	10,050 kg (23,150 lbm)	7575 kg (16,700 lbm)	12,000 kg (26,500 lbm)
Cost	$20 million	$125–155 million	$125–155 million	$125–155 million	$125–155 million
First Flight	2006	1996	2002	2005	TBD
Launch Site(s)	CSG (Kourou)	CSG (Kourou)	CSG (Kourou)	CSG (Kourou)	CSG (Kourou)

India

Israel

Vehicle	PSLV	GSLV Mark I	GSLV Mark II	Shavit 1, LK-A
Performance				
LEO Maximum	3700 kg (8150 lbm)	5000 kg (11,000 lbm)	5000 kg (11,000 lbm)	350 kg (770 lbm)
SSO	1350 kg (2975 lbm)	2000 kg (4400 lbm)	2000 kg (4400 lbm)	—
GTO	1050 kg (2315 lbm)	1900 kg (4200 lbm)	2100 kg (4660 lbm)	—
Cost	$15–17 million	$35 million	$35 million	?
First Flight	1993	2001	2005?	1995
Launch Site(s)	Satish Dhawan	Satish Dhawan	Satish Dhawan	Palmachim

China (continued)

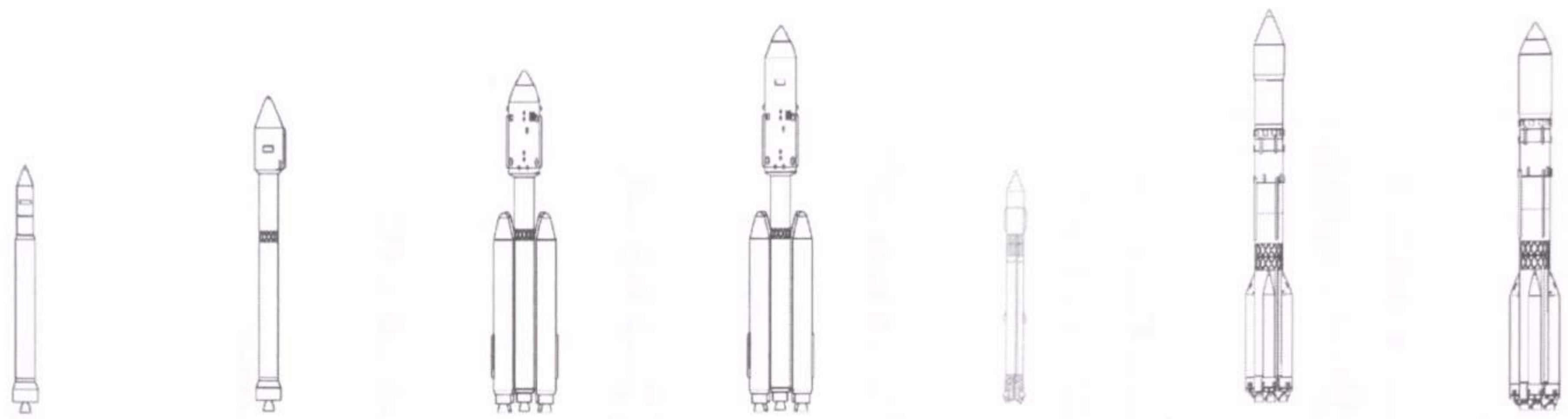


LM-3	LM-3A	LM-3B	LM-3C	LM-4B
?	6000 kg (13,225 kg)	11,200 kg (24,700 lbm)	9100 kg (20,000 lbm)	?
?	?	6000 kg (1325 lbm)	?	2800 kg (6170 lbm)
1500 kg (3300 lbm)	2600 kg (5700 lbm)	5100 kg (11,250 lbm)	3800 kg (8400 lbm)	—
\$35–40 million	\$45–55 million	\$50–70 million	?	\$25–35 million
1984	1994	1996	?	1999
Xichang	Xichang	Xichang	Xichang	Taiyuan Jiuquan

Japan

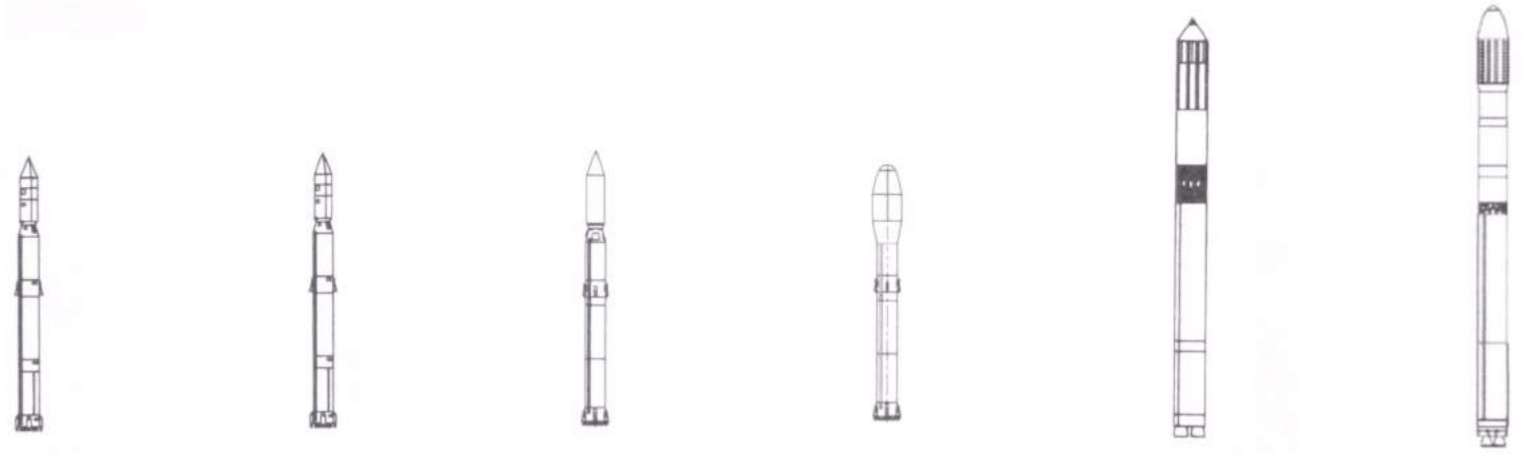
H-IIA 202	H-IIA 204	M-V
9940 kg (21,900 lbm)	?	1900 kg (4200 lbm)
4350 kg (9590 lbm)	?	960 kg (2100 lbm)
4100 kg (9040 lbm)	5800 kg (12,800 lbm)	1280 kg (2800 lbm)
\$70 million	\$83 million	\$57 million
2001	?	1997
Tanegashima	Tanegashima	Kagoshima

Russia



Vehicle	Angara 1.1	Angara 1.2	Angara A3	Angara A5	Kosmos 3M	Proton K/Block DM	Proton M/Breeze M
Performance							
LEO Maximum	2000 kg (4400 lbm)	3700 kg (8160 lbm)	14,000 kg (31,000 lbm)	24,500 kg (54,000 lbm)	1500 kg (3300 lbm)	19,760 kg (43,560 lbm)	21,000 kg (46,300 lbm)
SSO	?	?	?	?	775 kg (1700 lbm)	3620 kg (7980 lbm)	?
GTO	—	—	2500 kg (5500 lbm)	6400 kg (14,000 lbm)	—	Up to 4930 kg (10,846 lbm)	5500 kg (12,100 lbm)
Cost	?	?	?	?	$12 million	Price negotiable	Price negotiable
First Flight	TBD	TBD	2006	2006	1967	1967	2001
Launch Site(s)	Plesetsk	Plesetsk	Plesetsk	Plesetsk	Plesetsk Kapustin Yar	Baikonur	Baikonur

Ukraine



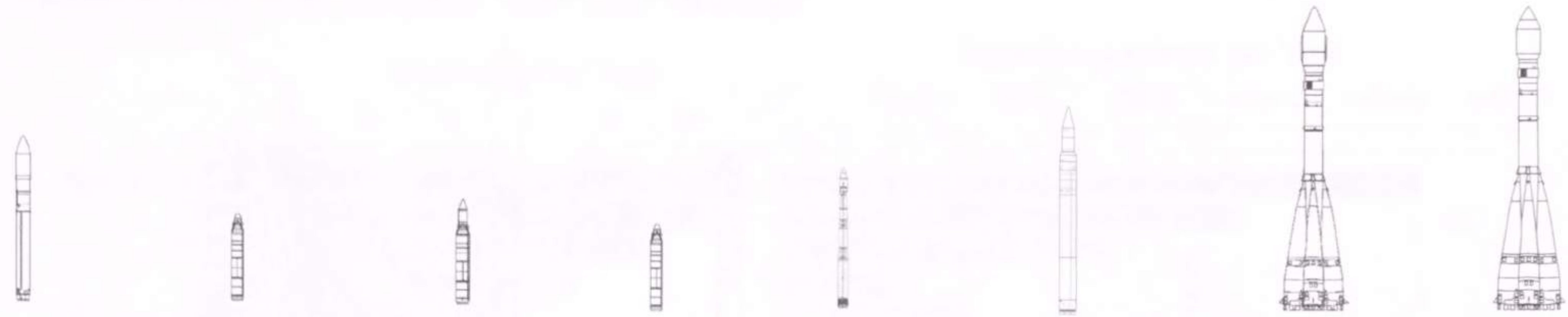
Vehicle	Cyclone 2	Cyclone 3	Cyclone 2K	Cyclone 4	Zenit 2	Zenit 3SL/3SLB
Performance						
LEO Maximum	3350 kg (7370 lbm)	4100 kg (9020 lbm)	2750 kg (6060 lbm)	5860 kg (12,920 lbm)	13,920 kg (30,700 lbm)	?
SSO	?	—	1500 kg (3300 lbm)	3800 kg (8375 lbm)	4900 kg (10,800 lbm)	?
GTO	—	—	?	1560 kg (3440 lbm)	—	6066 kg (13,373 lbm)
Cost	$20–25 million	$20–25 million	?	?	?	Price negotiable
First Flight	1967	1977	2004	2006	1985	1999
Launch Site(s)	Baikonur	Plesetsk	Baikonur	Alcântara	Baikonur	Sea Launch Odyssey Baikonur

United States

Vehicle	Athena I	Athena II	Atlas IIAS	Atlas IIIA	Atlas IIIB	Atlas V 400	Atlas V 500
Performance							
LEO Maximum	820 kg (1805 lbm)	2065 kg (4520 lbm)	8618 kg (19,000 lbm)	8640 kg (19,050 lbm)	10,759 kg (23,720 lbm)	12,500 kg (27,558 lbm)	Up to 20,520 kg (45,239 lbm)
SSO	360 kg (790 lbm)	1165 kg (1565 lbm)	?	—	—	?	?
GTO	—	590 kg (1290 lbm)[1]	3719 kg (8200 lbm)	4037kg (8900 lbm)	4119 kg (9081 lbm)	Up to 4950 kg (10,913 lbm)	8670 kg (19,114 lbm)
Cost	$40–45 million	$45–50 million	Price negotiable	Price negotiable	Price negotiable	Price negotiable	Price negotiable
First Flight	1995	1998	1993	2000	2002	2002	2003
Launch Site(s)	Cape Canaveral Kodiak	Cape Canaveral Kodiak	Cape Canaveral Vandenberg	Cape Canaveral	Cape Canaveral	Cape Canaveral	Cape Canaveral

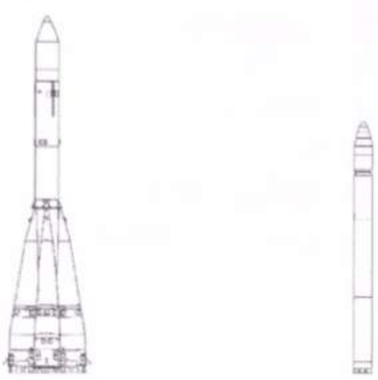
Notes:
[1] ***Requires available kick motor***

Russia (continued)



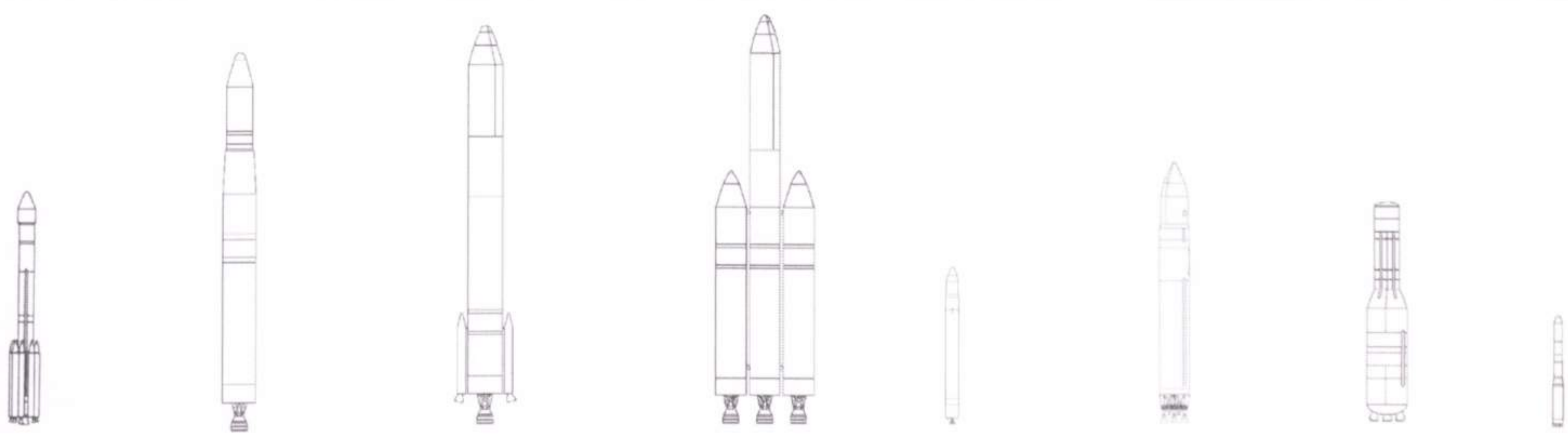
Rockot	Shtil-1	Shtil-2	Volna	Start-1	Strela	Soyuz U	Soyuz FG
1950 kg (4300 lbm)	140 kg (309 lbm)	Up to 220 kg (485 lbm)	180 kg (400 lbm)	632 kg (1393 lbm)	1560 kg (3440 lbm)	7000 kg (15,430 lbm)	7000 kg (15,430 lbm)
1000 kg (2205 lbm)	—	Up to 200 kg (440 lbm)	40 kg (88 lbm)	167 kg (368 lbm)	700 kg (1540 lbm)	4300 kg (9480 lbm)	4300 kg (9480 lbm)
—	—	—	—	—	1660 kg (3360 lbm)	1660 kg (3660 lbm)	
$12–15 million	$1.4–2.1 million	$3–4.5 million	$1–1.5 million	$9 million	$10.5 million	$30–50 million	$30–50 million
1994	1998	TBD	2004	1993	2003	1973	2001
Plesetsk	Delphin Submarine (Barents Sea)	Nenoksa	Kalmar Submarine (Barents Sea)	Svobodny Plesetsk	Svobodny Baikonur	Baikonur Plesetsk	Baikonur Plesetsk

Russia (continued)



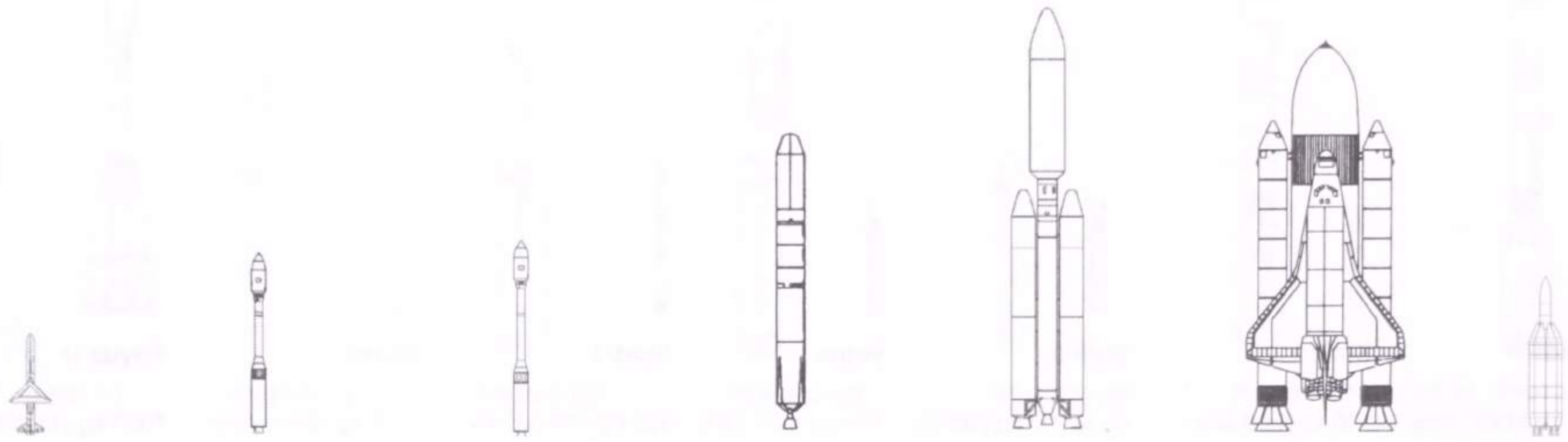
Molniya M	Dnepr-1
3700 kg (8150 lbm)	—
1500 kg (3300 lbm)	300 kg (661 lbm)
—	–
$30–40 million	$8–11 million
1960	1999
Plesetsk	Baikonur

United States (continued)



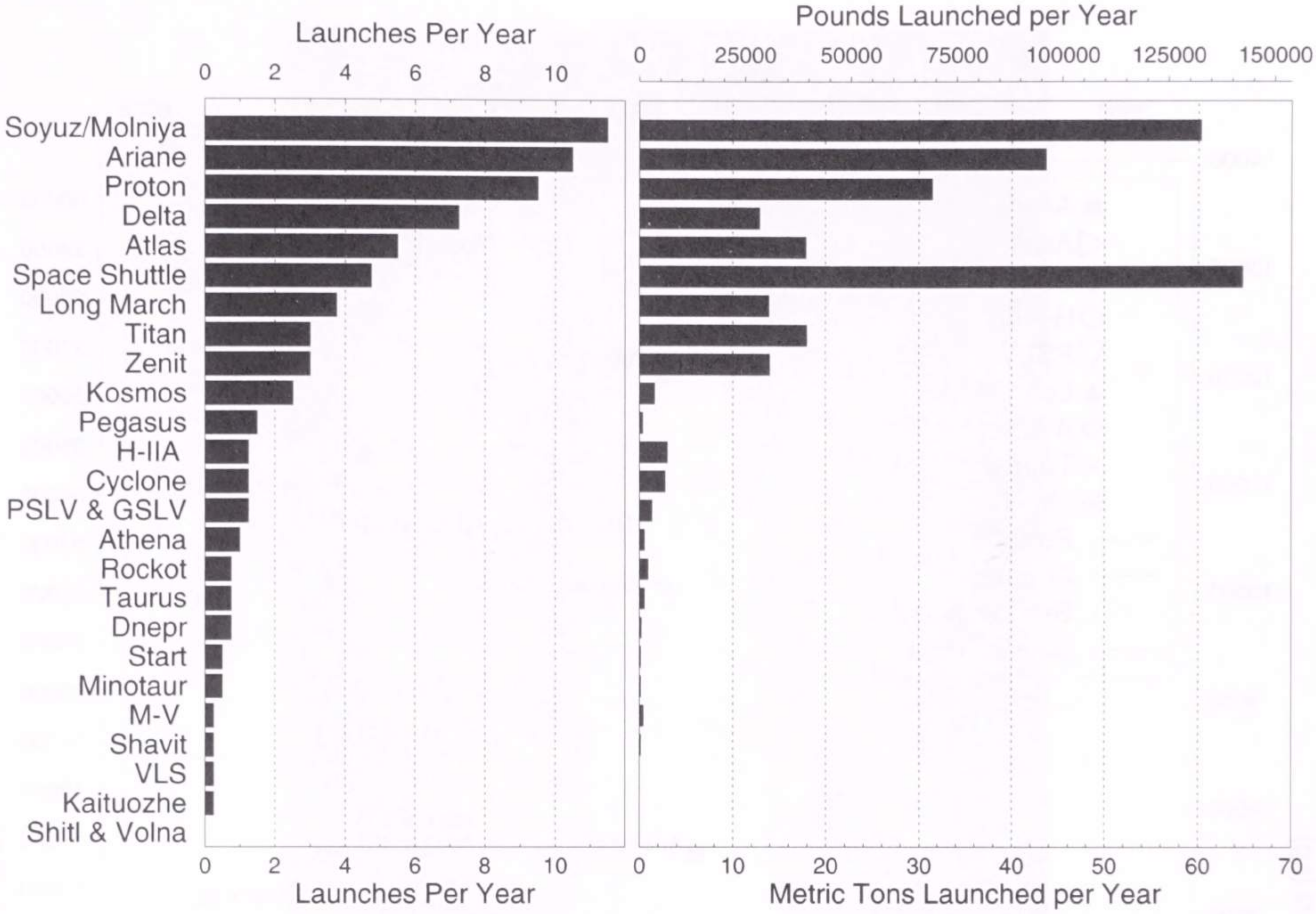
Delta II	Delta IV Medium	Delta IV Medium+	Delta IV Heavy	Falcon I	Falcon V	K-1	Minotaur
Up to 5102 kg (11,249 lbm)	8870 kg (19,554 lbm)	Up to 13,327 kg (29,381 lbm)	23,260 kg (51,280 lbm)	668 kg (1472 lbm)	5040 kg (11,088 lbm)	4600 kg (10,150 lbm)	607 kg (1339 lbm)
Up to 3186 kg (7025 lbm)	6832 kg (15,062 lbm)	10,863 kg (23,949 lbm)	19,665 kg (43,354 lbm)	408 kg (898 lbm)	3173 kg (6980 lbm)	1250 kg (2750 lbm)	317 kg (700 lbm)
Up to 1841 kg (4058 lbm)	3934 kg (8672 lbm)	6411 kg (14,135 lbm)	12,369 kg (27,269 lbm)	—	1500 kg (3300 lbm)	1570 kg (3460 lbm)	—
Price negotiable	Price negotiable	Price negotiable	Price negotiable	$5.9 million	$12 million	$17 million	$15–20 million
1990	2002	2002	2004	2004	2005	?	2000
Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Woomera Nevada Test Site	Vandenberg Others

United States (continued)



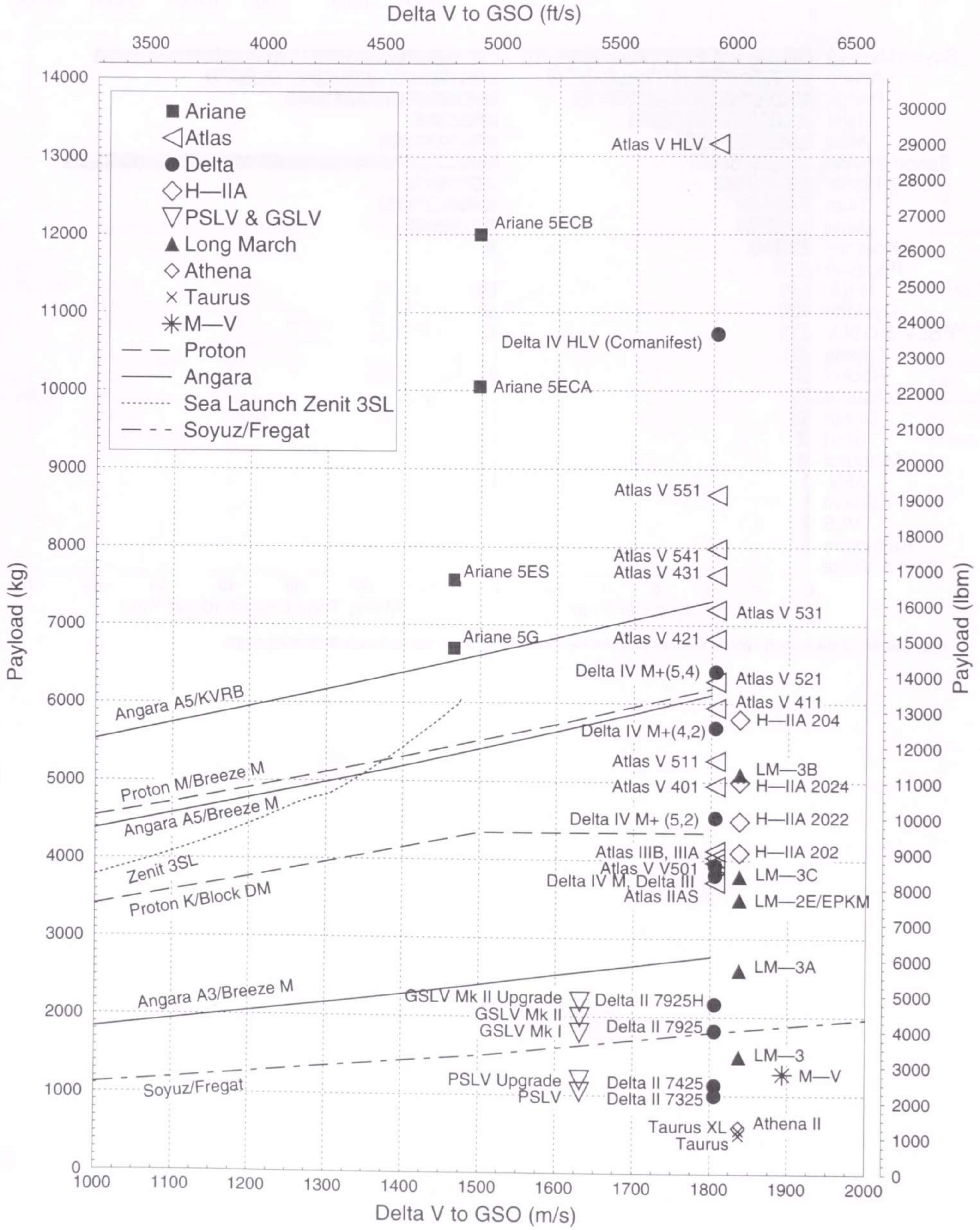
Vehicle	**Pegasus XL**	**Commercial Taurus**	**Taurus XL**	**Titan II**	**Titan IVB**	**Space Shuttle**	**Scorpius**
Performance							
LEO Maximum	443 kg (997 lbm)	1370 kg (3020 lbm)	1590 kg (3505 lbm)	1900 kg (4200 lbm)	21,680 kg (47,800 lbm)	28,800 kg (63,500 lbm)	314 kg (700 lbm)
SSO	190 kg (420 lbm)	720 kg (1590 lbm)	860 kg (2000 lbm)	1100 kg (2425 lbm)	?	—	125 kg (276 lbm)
GTO	—	495 kg (1090 lbm)	557 kg (1228 lbm)	—	?	—	—
Cost	$15–25 million	$25–47 million	$25–47 million	$30–40 million	$350–450 million	$450–750 million	$2.9 million
First Flight	1994	1998	2004	1988	1997	1981	2006
Launch Site(s)	Vandenberg Wallops Cape Canaveral Others	Vandenberg Others	Vandenberg Others	Vandenberg	Cape Canaveral Vandenberg	Kennedy Space Center	Vandenberg Cape Canaveral Wallops

Launch Vehicle Utilization, 1999–2002 Average



One–quarter of the launch vehicle families perform two-thirds of all launches and carry 80% of all cargo.

GTO Performance Capability



ANGARA

Courtesy Khrunichev.

The Angara is a new family of Russian launch vehicles designed by Khrunichev State Research and Production Center. Angara includes a variety of configurations from a Rockot-class small launch vehicle up to a heavy-lift vehicle to replace Proton. This full-scale prototype was exhibited at the Paris Air Show.

Contact Information

Marketing and Sales:
International Launch Services
1660 International Drive
McLean, VA 22102
USA
Phone: +1 (571) 633-7400
Fax: +1 (571) 633-7500
Web site: www.ilslaunch.com

Angara 1

Angara 1.1

Angara 1.2

General Description

Summary

The Angara is a new, modular family of launch vehicles designed by Khrunichev of Russia. The Angara program has three goals. First, it will provide Russia with a launch vehicle that is produced entirely domestically. Existing systems such as Proton rely on some Ukrainian components. Second, Angara can perform geosynchronous launches from the Russian launch site at Plesetsk, to reduce dependence on Kazakhstan for use of the Baikonur Cosmodrome. Third, it will use more environmentally friendly propellants than most of Russia's current vehicles. The Angara family will include a variety of launch vehicles in different sizes. The Angara 1 is a small launch vehicle. It uses a new LOX/kerosene first stage with one of two existing upper stages. The Angara 1.1 configuration uses the Breeze M core upper stage from the Rockot, whereas the Angara 1.2 uses a new kerosene/oxygen stage powered by the RD-0124A engine.

Status

In development. Launch date TBD.

Origin

Russia

Key Organizations

Marketing Organization	International Launch Services
Launch Service Provider	Khrunichev State Research and Production Center
Prime Contractor	Khrunichev State Research and Production Center

Primary Missions

Small to medium LEO satellites

Estimated Launch Price

No data available

Spaceport

Launch Site	Plesetsk LC 35, Pad 1
Location	62.9° N, 40.7° E
Available Inclinations	63, 73, 82, 86, 90, 93, 96, 98 deg

Performance Summary

	Angara 1.1	Angara 1.2
200 km (108 nmi), 63 deg	2000 kg (4400 lbm)	3700 kg (8160 lbm)
200 km (108 nmi), 90 deg	1650 kg (3640 lbm)	3200 kg (7060 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No direct capability—requires orbital plane change	
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	?	?
GTO	No capability	No capability
Geostationary Orbit	No capability	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	0

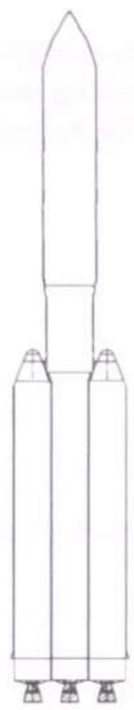
Flight Rate

Unknown

Angara A3 and A5

General Description

Angara A3



Angara A5

Summary

The Angara A3 and A5 are the medium- and heavy-lift configurations in the Angara family, designed to carry Zenit- and Proton-class payloads respectively. They use the first and second stage of the Angara 1.2, and add two or four strap-on boosters based on the first stage Universal Module. The Angara A5 also adds a KVRB hydrogen upper stage for GTO missions. A future super-heavy-lift configuration of the A5 replaces the second stage with a large cryogenic upper stage called the Unified Oxygen/Hydrogen Module (UOHM).

Status

In development. Angara A5: First launch in 2006.

Origin

Russian Federation

Key Organizations

Marketing Organization	International Launch Services
Launch Service Provider	Khrunichev State Research and Production Center
Prime Contractor	Khrunichev State Research and Production Center

Primary Missions

Medium and heavy LEO, GTO, or GEO spacecraft

Estimated Launch Price

No data available

Spaceport

Launch Site	Plesetsk LC 35, Pad 1
Location	62.9° N, 40.7° E
Available Inclinations	63, 73, 82, 86, 90, 93, 96, 98 deg

Performance Summary

	Angara A3	Angara A5	Angara A5/UOHM
200 km (108 nmi), 63 deg	14,000 kg (31,000 lbm)	24,500 kg (54,000 lbm)	28,500 kg (62,800 lbm)
200 km (108 nmi), 90 deg	12,600 kg (27,800 lbm)	22,400 kg (49,000 lbm)	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No direct capability—requires orbital plane change		
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	?	?	?
GTO: 5500×35,786 km (2970×19,323 nmi), 25 deg	2500 kg (5500 lbm)	6600 kg (14,000 lbm)	8000 kg (17,600 lbm)
Geostationary Orbit	1000 kg (2200 lbm)	4000 kg (8800 lbm)	5000 kg (11,000 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	0

Flight Rate

Unknown

Nomenclature

The name Angara comes from the river in Siberia that drains Lake Baykal. It may not be a coincidence that Angara was also an early code name for the Plesetsk missile base. The Angara family of launch vehicles is modular and therefore includes several different configurations. These configurations are designated by a number that indicates the number of first-stage universal modules. The Angara 1 has a single first-stage core, whereas the Angara A3 adds two more of these modular units as strap-on boosters, and the Angara A5 has four strap-on boosters.

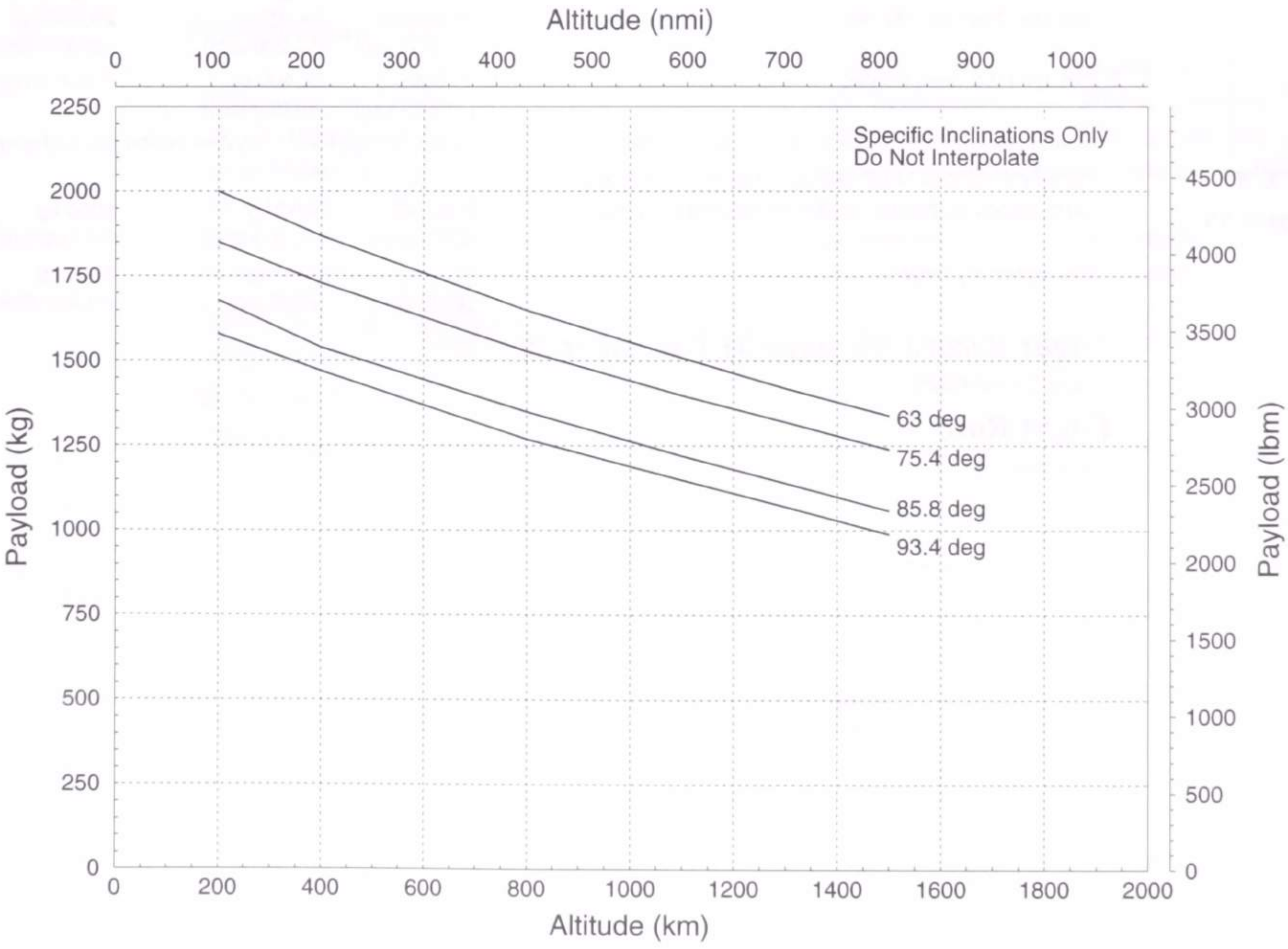
Cost

Commercial launch prices for Angara have not been announced. Although Angara is an official program of the Russian federal government, a general shortage of funds for space activities means that relatively little financial support is actually provided by either the Russian military or the Russian space agency, Federal Space Agency (FSA). Development costs are being paid primarily by Khrunichev from its commercial revenues, supplemented by a $40 million loan from Vneshtorgbank. In 1999 ILS agreed to pay a $68 million franchise fee for the marketing rights to Angara, but has not invested in the program. The Russian Defense Ministry is responsible for building the Angara launch facilities at Plesetsk. Promised funding was delayed until 2001, when state funding was received to begin construction.

Availability

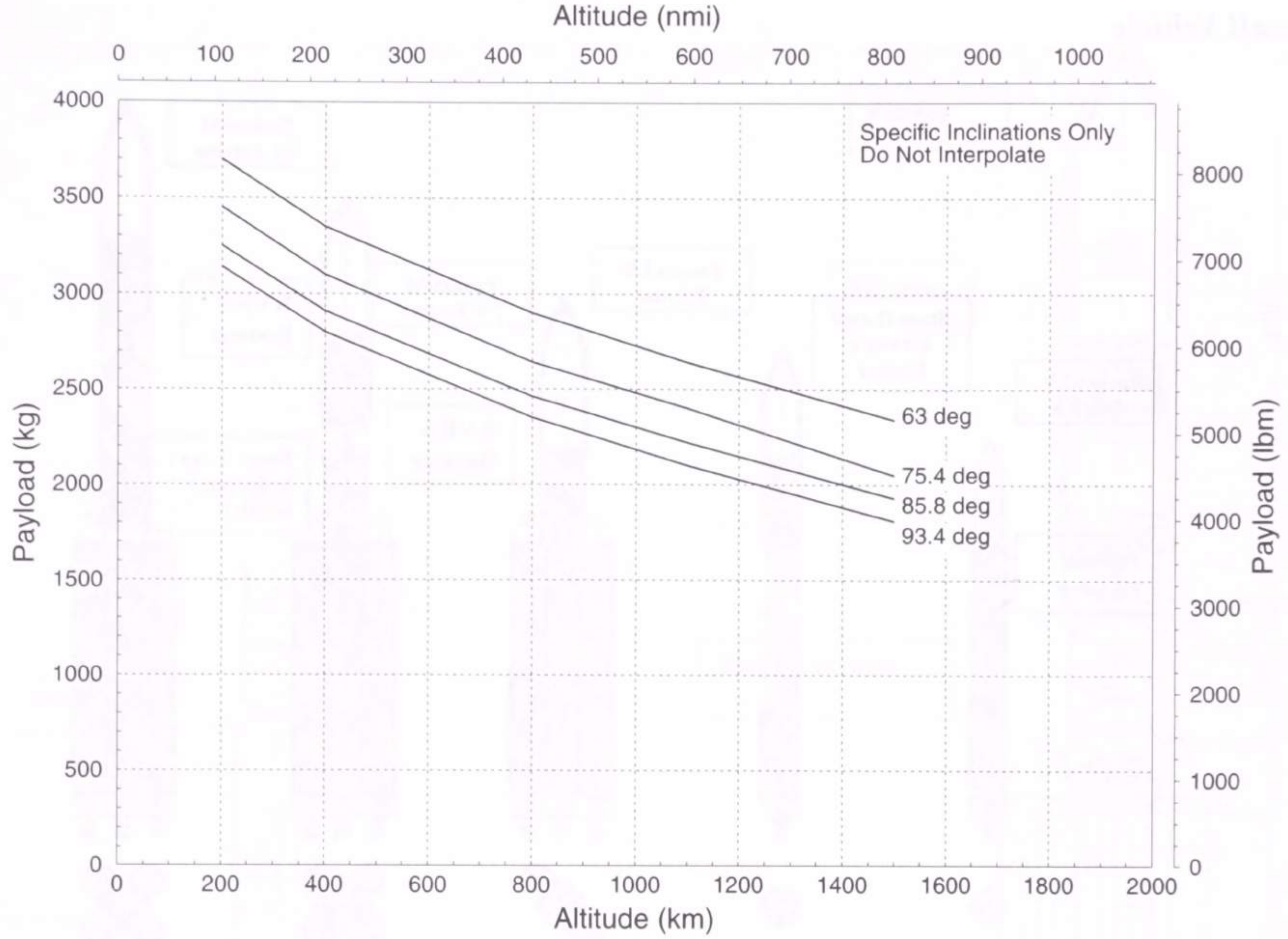
The program will proceed directly to the Angara 5. The first launch of Angara A5 planned for 2006. The schedule is dependent on the availability of Russian government funding. Previously Angara had been scheduled to debut in 2001. Development of the launch vehicle is progressing relatively smoothly, but a lack of funds for the launch complex has delayed the first launch. Commercial Angara launch services are marketed by International Launch Services, which also markets Atlas and Proton.

Performance

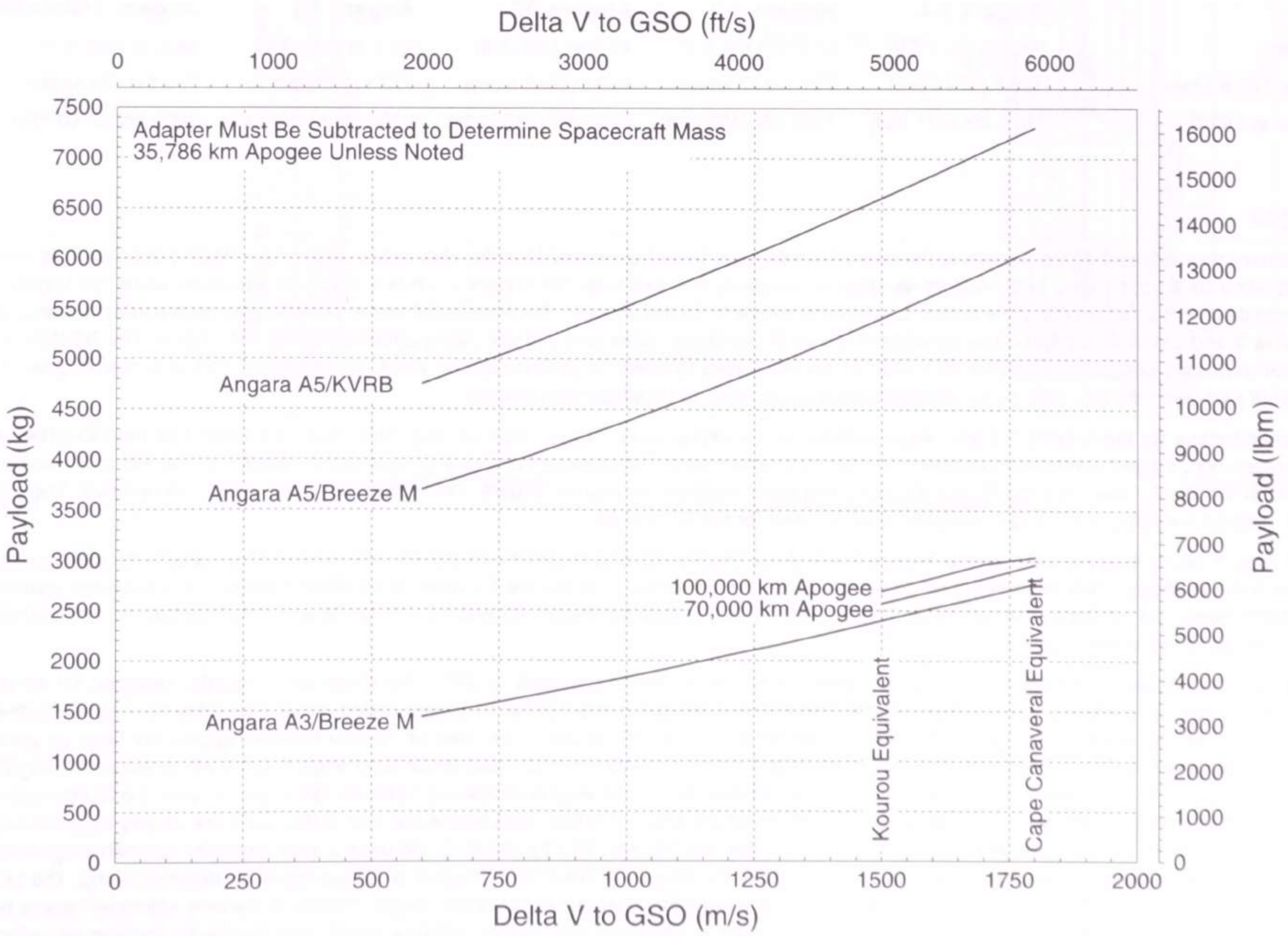


Angara 1.1 Performance

Performance



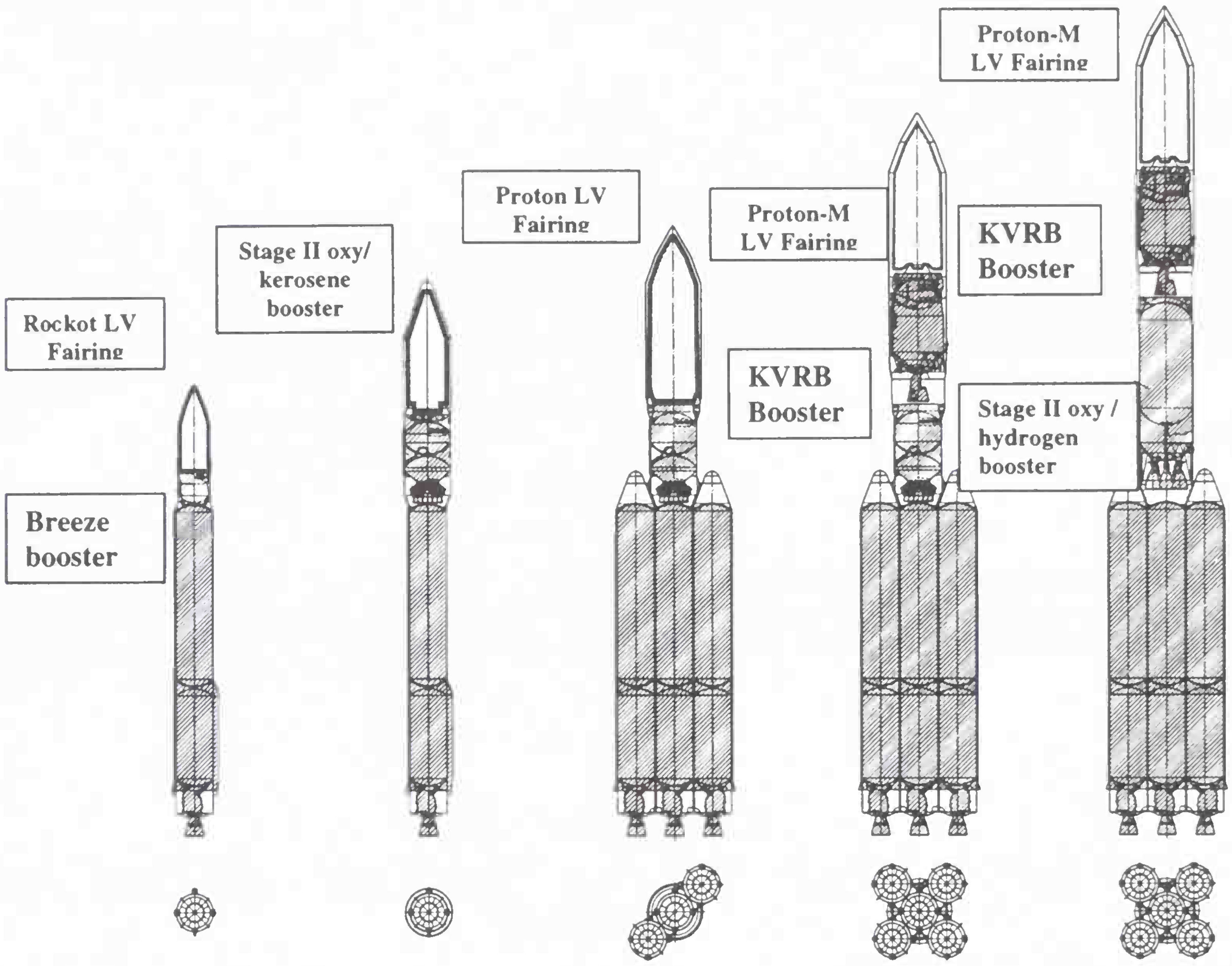
Angara 1.2 Performance



Angara GTO Performance

Vehicle Design

Overall Vehicle



Courtesy Khrunichev State Space Research and Production Center.

	Angara 1.1	Angara 1.2	Angara A3	Angara A5	Angara A5/UOHM
Height	34.9 m (114.5 ft)	41.5 m (136.5 ft)	45.8 m (150.2 ft)	55.4 m (181.7 ft)	64.0 m (209.9 ft)
Gross Liftoff Mass	149 t (329 klbm)	171 t (377 klbm)	480 t (1055 klbm)	773 t (1705 klbm)	790 t (1742 klbm)
Thrust at Liftoff	1920 kN (431 klbf)	1920 kN (432 klbf)	5760 kN (1259 klbf)	9600 kN (2160 klbf)	9600 kN (2160 klbf)

Stages

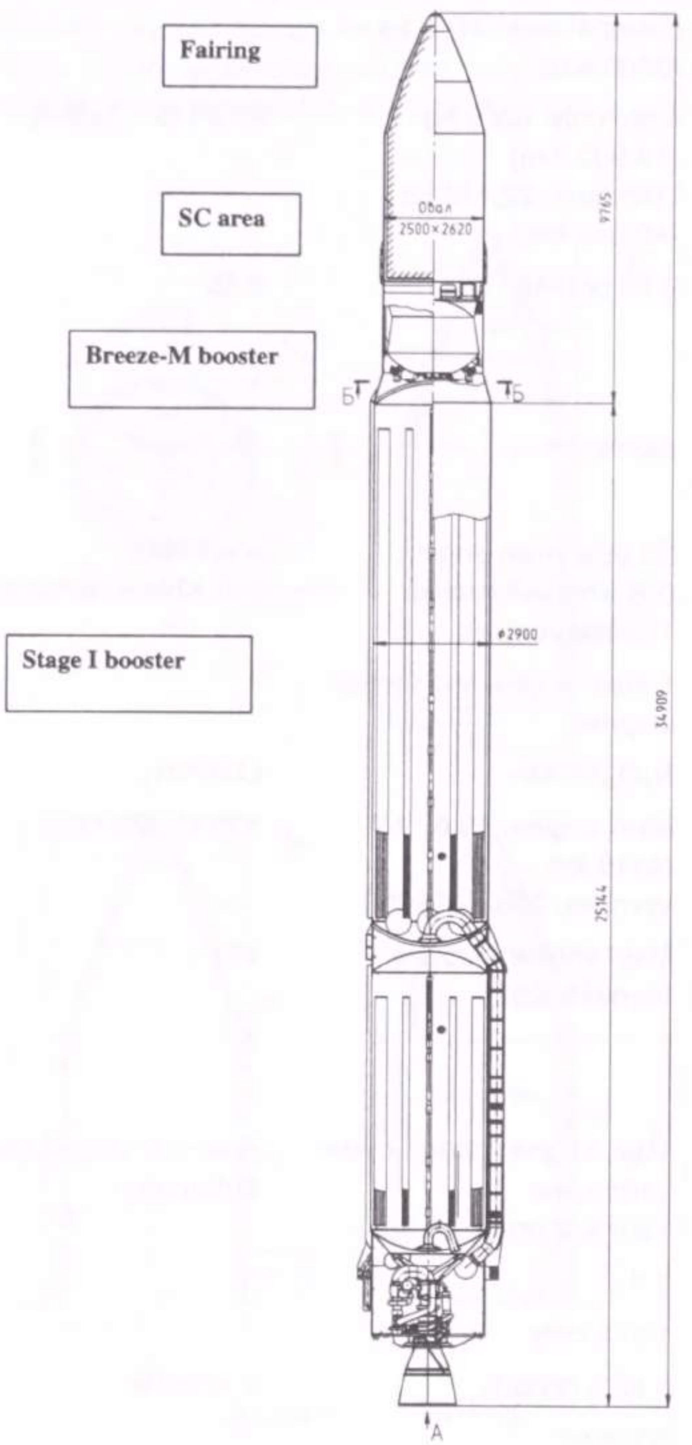
The primary new element of the Angara is the new LOX/kerosene fueled Universal Module, also called a Common Rocket Module. This modular unit is used alone as a core stage, or in clusters as strap-on boosters. For example, the Angara 1.1 has a single core module, while the Angara A5 has a core module with four additional units added as strap-on boosters. In this respect, the Angara follows a modular approach similar to that used on the new Atlas V and Delta IV. Each module consists of a pair of cylindrical tanks and a single high-performance RD-191 engine. The RD-191 is a single-chamber derivative of the four-chamber RD-170/171 engines used to power both Energia and Zenit. While the RD-191 is a new engine, it shares a number of parts with the RD-170, so its development is considered to be simple and low-risk.

The second stage for the Angara 1.2 and larger vehicles is a LOX/kerosene fueled stage as well. This stage is powered by the RD-0124A engine, a new oxygen-rich, staged combustion engine with four thrust chambers. Development of this engine originally started as the RD-0124 for the planned upgrade of the Block I stage for the Soyuz ST, but it has been modified for use on Angara. Two variations of this stage will be used. The smaller one will be used on the Angara 1.2 and a stretched version used for the A3 and A5.

The Breeze M upper stage is used on the Angara 1.1 and the Angara A3, and optionally on the A5. The Breeze M is fueled with storable propellants, and has a toroid-shaped tank around the core stage that can be jettisoned. The Angara 1.1 uses as its second stage only the central element without the external tank. The full Breeze M can be used as a GTO injection stage on Angara A3 and A5. Breeze M was developed for Proton, and more information can be found in that chapter.

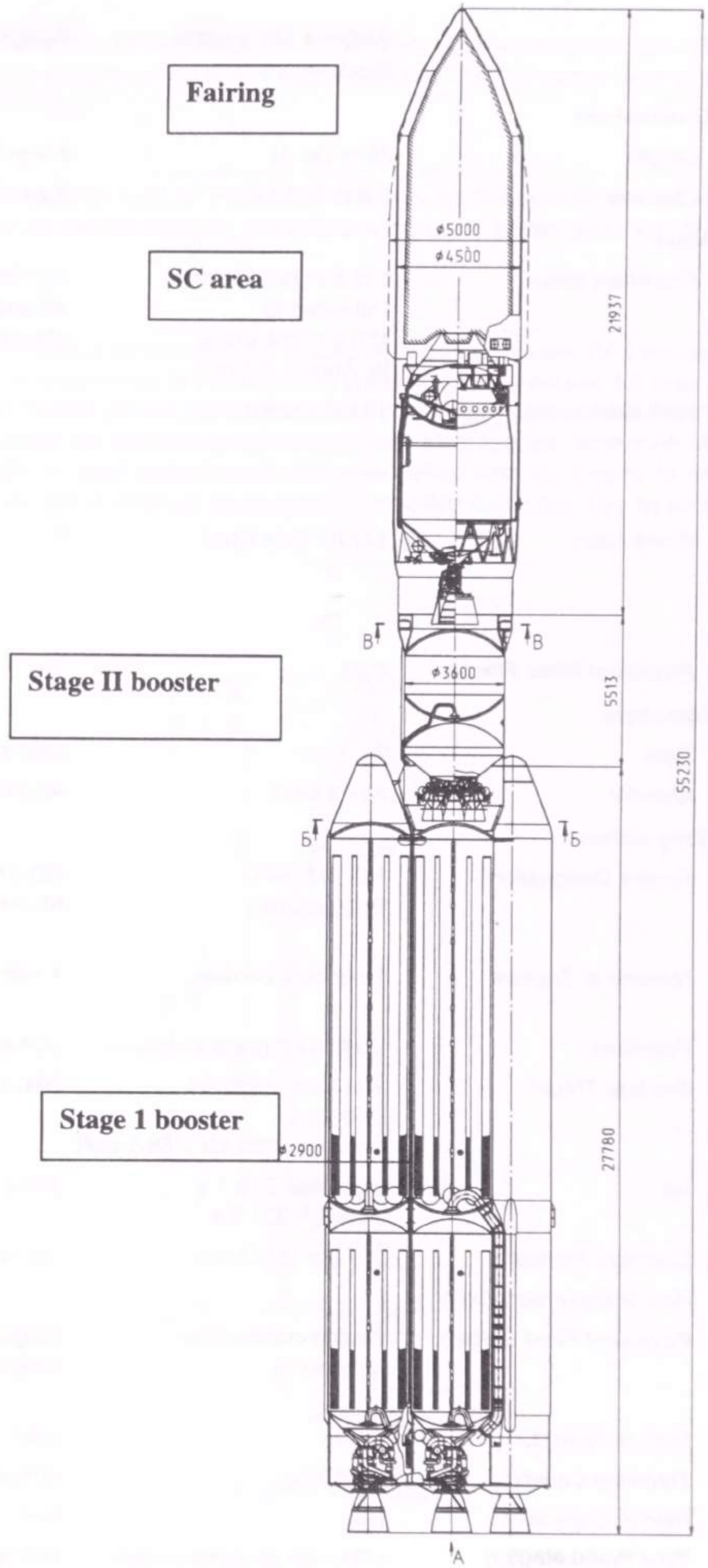
The Angara A5 will use a cryogenic upper stage called the KVRB to deliver payloads to GTO. The stage was originally designed for an upgrade of Proton, but it has not yet been used on that vehicle. The KVRB is related to the 12KRB hydrogen-fueled upper stage that Khrunichev built for India's GSLV launch vehicle, but it has roughly 50% more propellant. The KVD-1M3 engine to be used on Angara has also apparently been uprated from the basic KVD-1M used on GSLV. Previously this has been described as a 75-kN (17-klbf) class thrust-level engine, but it will produce 100 kN (23 klbf) for Angara. The KVD-1M, also known as the 11D56M, is a derivative of the 11D56 engine developed between 1965 and 1972 for the fourth stage of a variant of the N-1 super-heavy lift launcher. It was never flown. Whereas engines in this class built in the rest of the world are usually expander cycles (like the RL10, Vinci, and LE-5) or gas-generator cycles (such as the HM-7B, and YF-75), the KVD-1M uses a more complex fuel-rich staged-combustion cycle similar to the Space Shuttle Main Engine with a single shaft turbopump. The KVRB stage is enclosed inside an external fairing. The LOX tank is mounted above the hydrogen tank, opposite the layout typically used for other cryogenic upper stages. Russia is the only significant space power that operates no cryogenic stages, a gap that would be filled when the KVRB is put into service. A future growth plan for the A5 involves replacing the second stage with a large hydrogen-fueled stage dubbed the Universal Oxygen/Hydrogen Module (UOHM, or UOHS, where S stands for "stage"). This would be powered by four KVD-1M engines.

Vehicle Design



Courtesy Khrunichev.

Angara 1.1



Courtesy Khrunichev.

Angara A5 (with Kerosene Stage 2)

Vehicle Design

	Angara Universal Module	Stage 2	Breeze M	KVRB
Dimensions				
Length	25 m (82 ft)	6.9 m (22.6 ft)	2.61 m (8.6 ft)	10.1 m (33.1 ft)
Diameter	2.9 m (9.5 ft)	3.6 m (11.8 ft)	2.5 m (8.2 ft)	3.8 m (12.5 ft)
Mass				
Propellant Mass	132.6 t (292.3 klbm) Offloaded to 128.8 t (284 klbm) for Angara 1.2 only	1.2: 25.7 t (56.6 klbm) A3 and A5: 35.7 t (78.7 klbm)	Core tank: 5200 kg (11,500 lbm) External tank: 14,600 kg (32,200 lbm)	19.8 t (43.6 klbm)
Inert Mass	10 t (22 klbm)	?	Core tank: 1100 kg (2400 lbm) External tank: 1600 kg (3500 lbm)	3.5 t (7.7 klbm)
Gross Mass	142.6 t (314 klbm)	?	Core only: 6300 kg (13,900 lbm) With tank: 22,470 kg (49,540 lbm)	23.3 t (51.3 klbm)
Propellant Mass Fraction	0.93	?	0.83 or 0.88	0.85
Structure				
Type	?	Skin-stringer	?	?
Material	Aluminum?	Aluminum	Aluminum	Aluminum?
Propulsion				
Engine Designation	RD-191 (NPO Energomash)	RD-0124A (KB Khimavtomatiki)	S5.98M main engine (KB Khimavtomatiki) 11D458 verniers	KVD-1M3 (KB Khimavtomatiki)
Number of Engines	1 per core booster	1 with 4 thrust chambers	1 main engine + 4 vernier engines	1
Propellant	LOX/RG-1 grade kerosene	LOX/kerosene	N_2O_4/UDMH	LOX/LH_2
Average Thrust	Sea level: 1920 kN (431 klbf) Vacuum: 2084 kN (468.5 klbf)	294.3 kN (66.16 klbf)	Main engine: 19.62 kN (4410 lbf) Verniers: 396 N (89 lbf)	100 kN (23 klbf)
Isp	Sea level: 310.7 s Vacuum: 337.5 s	359 s	Main engine: 325.5 s Verniers: 252 s	461 s
Chamber Pressure	257 bar (3725 psi)	162 bar (2350 psi)	?	?
Nozzle Expansion Ratio	?	?	?	?
Propellant Feed System	Staged-combustion turbopump	Staged-combustion turbopump	Main engine: closed-cycle turbopump Verniers: pressure fed	Fuel-rich staged combustion turbopump
Mixture Ratio (O/F)	2.6:1	2.6:1	2.0:1	?
Throttling Capability	30–100%	60% or 100%	100% only	?
Restart Capability	No	No?	8 total restarts	5 restarts
Tank Pressurization	LOX: Hot gaseous oxygen Fuel: Hot helium	Hot helium	Nitrogen	?
Attitude Control				
Pitch, Yaw	Main engine gimbal ±8 deg	Main engine gimbal single plane ±4 deg	12 ACS thrusters	Main-engine gimbal ±4 deg
Roll	Four roll thrusters and two aerodynamic surfaces	Main engine gimbal single plane ±4 deg	12 ACS thrusters	Roll-control thrusters
Staging				
Nominal Burn Time	300 s maximum	300 s ?	425 s	?
Shutdown Process	Command?	Command shutdown	Command shutdown	Command shutdown
Stage Separation	4 retro-rockets	Spring ejection	ACS/Spring ejection of payload	?

Vehicle Design

Attitude Control System

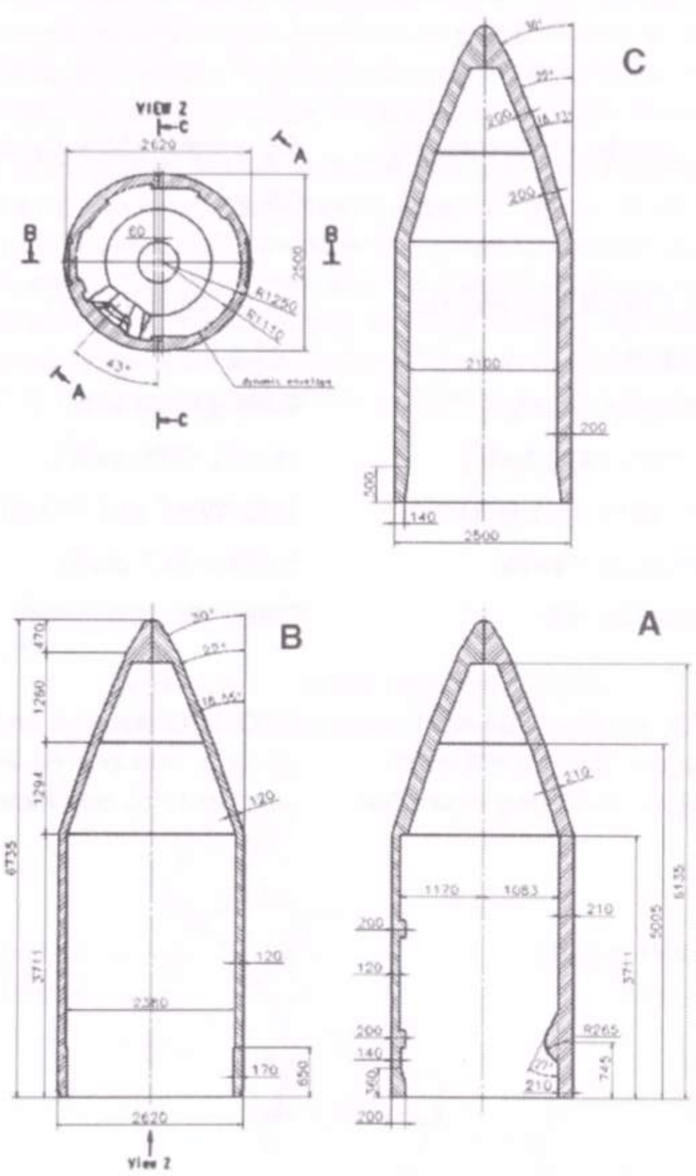
The first stage controls attitude using a hydraulic nozzle gimbal on the RD-191 engine. The Breeze stage has 12 thrusters for attitude control. The RD-0124A engine can gimbal each of its four thrust chambers in a single plane to provide attitude control for the second stage. The KVRB upper stage uses a main engine nozzle gimbal as well as small roll control thrusters fueled by UDMH/N_2O_4.

Avionics

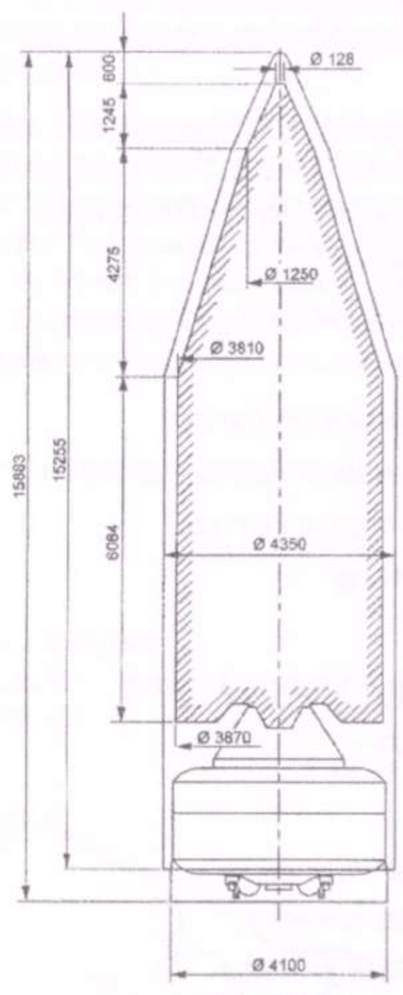
Angara uses an inertial guidance system and a digital flight computer derived from systems used on Proton M and Breeze M. The avionics are mounted on the Universal Module and on the second stage. The Breeze M upper stage has its own avionics, including power supply, inertial guidance, digital flight computer, and a telemetry system with onboard recording capability.

Payload Fairing

Several different payload fairings will be used on different Angara configurations. All fairings have options for spacecraft access doors and RF windows. The Angara 1.1 uses a modified Rockot payload fairing, which is designed to be compatible with the Breeze core upper stage. The Angara 1.2 uses a new PLF design in the standard two-stage configuration, though it too can use the Rockot payload fairing if a Breeze third stage is added. The Angara A3 will used a stretched version of the Proton M/Breeze M payload fairing, with the shorter, standard length version available if needed. Angara A5 will have different payload fairings for different upper-stage configurations. If the Breeze M upper stage is used, the same fairings from the Angara A3 are used. If the KVRB upper stage is used, a cylindrical fairing encapsulates the stage with a separate payload fairing mounted to the top. Two types of payload fairing will be available, a 4.3-m-diam and a larger 5.1-m-diam fairing.



Angara 1.1 fairing



Angara 3 fairing

	Angara 1.1	Angara 1.2	Angara A3	Angara A5
Length	6.735 m (22.1 ft)	9.83 m (32.3 ft)	11.6 m or 15.26 m (38.1 or 50.1 ft)	4-m-diam fairing: 15.26, 16.57, 17.18 m (50.1, 54.4, 58.4 ft) 5-m-diam fairing: 19.65 m (64.5 ft)
Primary Diameter	2.5 x 2.62 m (8.2 x 8.6 ft) oval	3.87 m (12.7 ft)	4.35 m (14.3 ft)	4.35 m or 5.1 m (14.3 or 16.7 ft)
Mass	710 kg (1565 lbm)	?	2200 kg or 2600 kg (4850 or 5730 lbm)	?
Structure	Carbon–fiber composite sandwich over aluminum honeycomb core		Carbon–fiber composite sandwich over aluminum honeycomb core	
Material	Composite and aluminum			

Payload Accommodations

	Angara 1.1	Angara A3	Angara A5
Payload Compartment			
Maximum Payload Diameter	2100 x 2380 mm (82.7 x 93.7 in)	3821 mm (150.4 in)	3707 mm or 4357 mm (145.9 or 171.5 in)
Maximum Cylinder Length	3711 mm (146.1 in)	4166 mm (164.0 in)	3936 mm or 4918 mm (155.0 or 193.6 in)
Maximum Cone Length	2504 mm (98.6 in)	5468 mm (215.3 in)	3814 mm or 6577 mm (150.2 or 258.9 in)
Payload Adapter Interface Diameter	2390 mm (94.1 in)	2490 mm (98.0 in)	KVRB: 2490 mm (98.0 in) No upper stage: 4130 mm (126.6 in)
Payload Integration			
Nominal Mission Schedule Begins	New spacecraft: 24 months Reflights: 12 months	New spacecraft: 24 months Reflights: 12 months	New spacecraft: 24 months Reflights: 12 months
Launch Window			
Last Countdown Hold Not Requiring Recycling	?	?	?
On-Pad Storage Capability	?	?	?
Last Access to Payload	?	?	?
Environment			
Maximum Axial Load	4.6 *g* static ±10% dynamic	4.5 *g* static ±10% dynamic	4.5 *g* static ±10% dynamic
Maximum Lateral Load	0.6 *g*	0.7 *g*	0.9 *g*
Minimum Lateral/Longitudinal Payload Frequency	?	?	?
Maximum Acoustic Level	133.5 dB at 160 Hz	132.1 dB at 200-250 Hz	133.5 dB at 160 Hz
Overall Sound Pressure Level	141.6 dB	140.8 dB	141.6 dB
Maximum Flight Shock	5000 *g* at 49 kHz	5000 *g* at 49 kHz	5000 *g* at 49 kHz
Maximum Dynamic Pressure on Fairing	30.9 kPa (640 lbf/ft^2)	22.4 kPa (470 lbf/ft^2)	18 kPa (375 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 $BTU/ft^2/s$)	1135 W/m^2 (0.1 $BTU/ft^2/s$)	1135 W/m^2 (0.1 $BTU/ft^2/s$)
Maximum Pressure Change in Fairing	5 kPa/s (0.7 psi/s)	5 kPa/s (0.7 psi/s)	5 kPa/s (0.7 psi/s)
Cleanliness Level in Fairing	Class 100,000	Class 100,000	Class 100,000
Payload Delivery			
Standard Orbit Injection Accuracy (3 sigma)	LEO: 2% in altitude, 0.05 deg inclination	GTO: ±100 km (54 nmi) apogee, 300 km (64 nmi) perigee, 0.15 deg inclination	GTO: ±100 km (54 nmi) apogee, 300 km (64 nmi) perigee, 0.15 deg inclination
Attitude Accuracy (3 sigma)	?	?	?
Nominal Payload Separation Rate	?	?	?
Deployment Rotation Rate Available	?	?	?
Loiter Duration in Orbit	5 h	5 h	7.5 h
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes	Yes
Multiple/Auxiliary Payloads			
Multiple or Comanifest	Unknown	Unknown	Unknown
Auxiliary Payloads	Unknown	Unknown	Unknown

Production and Launch Operations

Production

Angara is produced by the Khrunichev State Research and Production Center. A key goal of the Angara program is to produce the vehicle entirely within Russia, to eliminate the reliance on countries such as Ukraine for components. Khrunichev produces most elements of the launch vehicle except the engines and avionics.

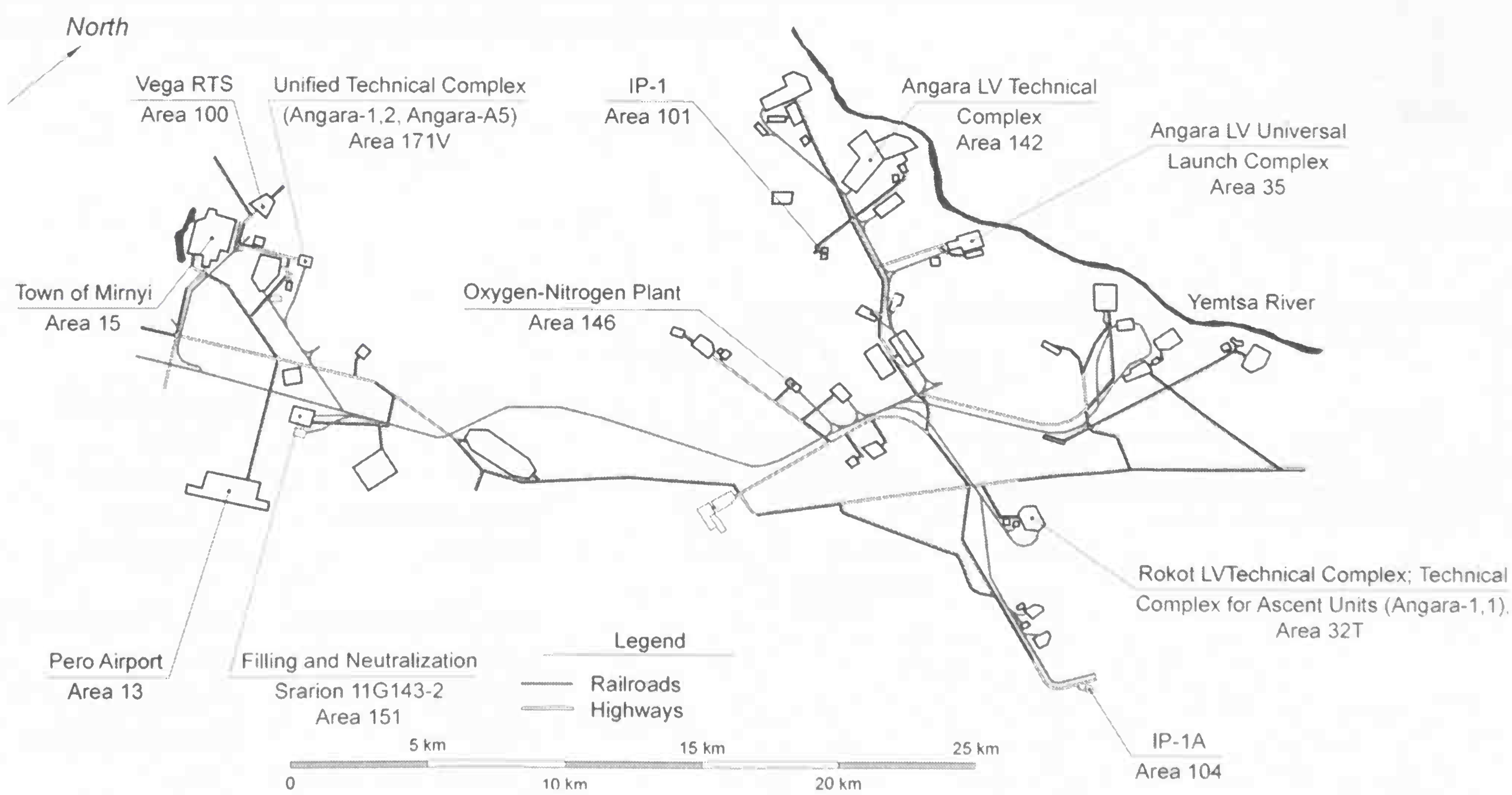
Organization	Responsibility
Khrunichev State Research and Production Center	Prime contractor
NPO Energomash	RD-191 first-stage engine
KB Khimavtomatiki	Upper-stage engines

Launch Operations—Plesetsk

Angara will launch from the Plesetsk Cosmodrome in northern Russia. A primary purpose of the Angara program is to provide Russia with a medium- to heavy-lift capability from a spaceport on Russian soil. Currently the largest launch vehicle supported at Plesetsk is the medium-lift Soyuz.

Angara will use the launch complex that was started, but not completed, for the Zenit launch vehicle. Until the breakup of the Soviet Union, it was planned that Zenit would be launched from two pads at the new Launch Complex 35 in Plesetsk. Following the breakup, Russia decided not to invest in infrastructure for a launch vehicle that was primarily built by Ukraine. As a result, the Zenit pads were not completed. For the Angara program, Pad 1 at Launch Complex 35 will be completed with a modified design to support Angara instead of Zenit. Because Angara's RD-191 engine is derived from the Zenit first-stage engine and Angara's aft compartment is designed to be similar to Zenit, the smaller Angara configurations are largely compatible with the existing pad design. The larger Angara configurations need a modified launch mount for the strap-on boosters and cryogenic propellant storage and fueling facilities. Angara will be integrated and tested horizontally in the assembly facilities built for Zenit.

Angara is integrated, tested, and checked out horizontally in the Assembly and Test Building at Area 142. Spacecraft operations for small spacecraft are performed at the Rockot facilities at Area 32T about 10 km from the launch complex. A larger Unified Technical Complex is being built at Area 171 that will allow processing of both large and small spacecraft as well as their upper stages. Spacecraft processing is conducted in parallel with launch vehicle processing. The spacecraft and the Breeze stage are fueled separately, then mated and enclosed in the payload fairing. The integrated assembly is then transported to the Assembly and Test Building for integration onto the launch vehicle in the horizontal position. When ready, the launch vehicle is transported to the launch pad horizontally and erected into the vertical position. Umbilical connections and fueling lines in the launch pad automatically connect to mating adapters on the bottom of the first stage. The second stage and KVRB umbilicals and feed lines are connected to a service tower.



Courtesy ILS.

Plesetsk map

Vehicle Upgrade Plans

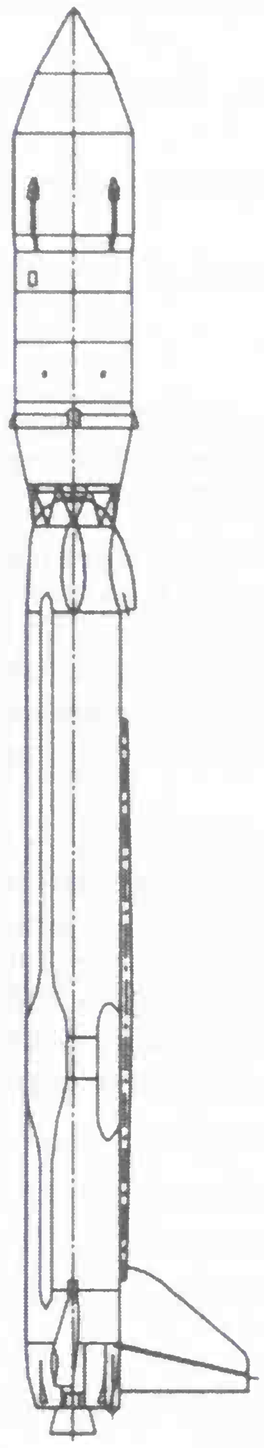
Baikal

A primary goal of advanced launch system development in Russia has been to eliminate the need for the highly restrictive stage impact zones that result from Russia's inland launch sites. These zones restrict available launch azimuths and orbit inclinations, and the impact of thousands of rocket stages in these zones has caused serious environmental and economic problems in nearby areas. To eliminate this problem and increase flexibility, Khrunichev has proposed an Angara version with a fly-back first stage. A rotating, single-piece wing would be mounted to the center of the first stage. At launch the wing would be aligned parallel with the launch vehicle so as to have little aerodynamic effect. After main-engine burnout, the wing would rotate perpendicular to the stage to provide lift for the return to the launch site. A jet engine in the nose of the vehicle would provide sufficient power to return the first stage to the Plesetsk runway. The stage would also be fitted with an aerodynamic three-surface tail for control and stability.

The design of Baikal is aimed at reducing development and operational costs. The low staging velocity means that only limited TPS is required, and conventional structural materials can be used. The RD-35 jet engine is already in use on the Yak 130 trainer, while the landing gear are from a Sukhoi fighter. Development of the Baikal reusable stage is estimated to cost $130–150 million, of which half is to be funded by NPO Molniya. The airframe would be designed for 50–100 flights, but the RD-191 engines would initially be replaced every 15–20 flights.

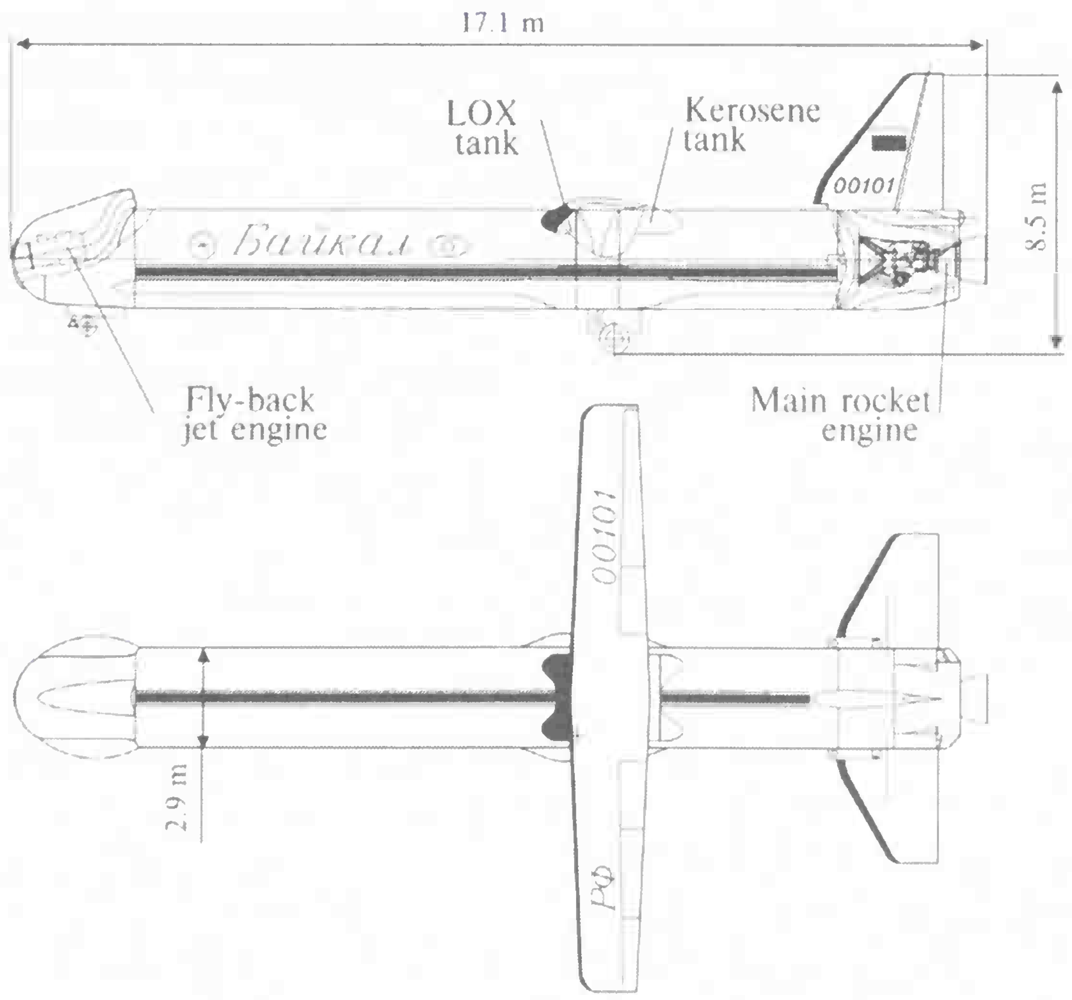
The Baikal flight profile begins with a normal ascent, followed by staging around Mach 5.5. The wings rotate into position as the booster coasts ballisticly to an altitude of almost 100 km (54 nmi). During the high-speed portion of reentry Baikal flies inverted, with the tail pointing down, and rotates into a normal position as it decelerates through Mach 1.7. The jet engine ignites to fly approximately 400 km (215 nmi) back to the runway at Mirny, near Plesetsk.

There are several possibilities for the future development of Baikal. Initial plans focus on an application of Baikal as the first stage of a vehicle like the Angara 1.2. In this configuration it could lift 1900 kg (4200 lbm) to a 200-km (108-nmi), 63-deg orbit. First flight would occur a few years after the first Angara flight, perhaps as early as 2006. Later developments could put up to four Baikal boosters in the place of the Angara strap-on boosters for a configuration like the Angara A5, delivering 18.4 t (40.5 klbm) to LEO or 2500 kg (5500 lbm) to GEO. Khrunichev has also promoted Baikal or variations of it as a reusable strap-on booster for Ariane 5, Delta IV, and the Space Shuttle. However, it appears unlikely that it will be adopted for these applications.



Courtesy Khrunichev.

Liftoff Configuration with



Reusable Stage 1

Courtesy Khrunichev.

Landing Configuration of Reusable Stage 1

	Baikal
Length	27.1 m (88.9 ft)
Height (gear to tail)	8.5 m (28 ft)
Wingspan	17.1 m (56.1 ft)
Rocket Propellant Mass	109.7 t (241.8 klbm)
Jet Fuel Mass	2.9 t (6.4 kblm)
Inert Mass	17.8 t (39.2 klbm)
Gross Mass	130.4 t (287.4 klbm)
Propellant Mass Fraction	0.84
Cruise Return Range	410 km (220 nmi)
Cruise Speed	490 km/h (265 knots)
Jet Engine Thrust	50 kN (11 klbf)

Vehicle History

Historical Summary

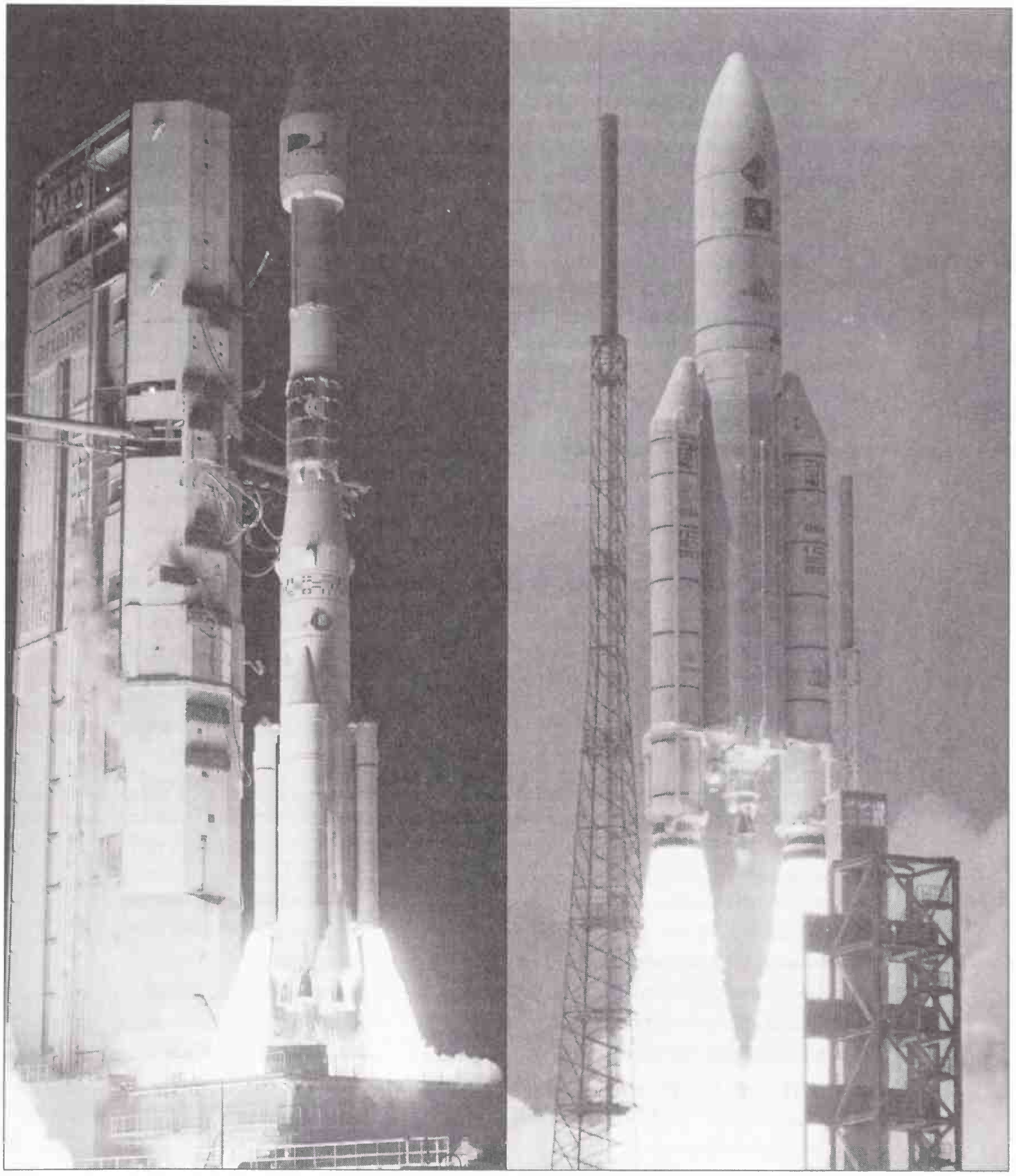
The Angara program began as a preliminary design study sponsored by the Federal Space Agency (FSA) and military space forces for a Proton-class launch vehicle called Yenisey. In 1993–1994 the Russian government conducted a competition to develop a successor to the Proton launch vehicle. The goals were to provide heavy-lift capability from the Plesetsk Cosmodrome operated by Russia, rather than the Baikonur Cosmodrome in Kazakhstan, and to reduce or eliminate the use of toxic propellants. There were two primary candidates, Energia-M from RSC Energia and Angara from Khrunichev. The Makeyev design bureau also proposed a design. The Energia-M was a smaller derivative of the Energia super-heavy-lift vehicle, using two Energia boosters and a smaller core stage (see Isakowitz, *International Reference Guide to Space Launch Systems, Second Edition,* 1995, p. 126 for additional information). The original Angara design had a 4.1-m (13.5-ft) diameter LOX/kerosene first stage and a cryogenic upper stage of the same diameter. Additional propellant for both stages would be carried in an unusual configuration of two side-mounted external tanks for each stage. This design was driven by the need to restrict the size of the stage elements to enable transportation to the launch site. The first stage would have used an RD-170/171 engine, similar to Zenit. While the Energia-M was closely based on the existing Energia, the brand-new Angara was selected instead. The Energia-M was rejected because it was dependent on launch infrastructure at Baikonur and possibly because it would have relied on strap-on boosters built in Ukraine.

In 1994 the FSA and military space forces signed a cooperative agreement to develop Angara, and the plan was backed by a January 1995 decree from President Boris Yeltsin. Funding levels and program schedules were set to launch the first Angara by 2005 from LC 35 at Plesetsk, using the partially completed Zenit launch infrastructure there. By 1996, however, the future of the Angara program was in doubt. Only half of the necessary funding was provided for the draft design phase, and the FSA stopped participating in joint meetings with the military and industry saying that Proton was sufficient for present launch needs.

During 1996–1998, the program was underfunded by the government, and Khrunichev's interest in the commercial launch market was growing. Angara was redesigned to reduce costs and make the vehicle more flexible. The initial Angara design addressed only Proton-class heavy payloads and would not have provided a modern replacement for smaller launch vehicles such as Kosmos, Cyclone, or Soyuz. The new Angara design is based on common first-stage modules that burn LOX/kerosene. Using only one core stage with a variety of upper stages already in development for Proton M and Soyuz 2 results in small- to medium-class launch vehicles that can replace Kosmos, Rockot, and Cyclone. By clustering three to five boosters together, a larger Proton-class launch vehicle is created. The modular design approach and use of previously funded upper stages would allow the first Angara to fly as soon as 2001, with larger vehicles following in 2003. The revised design was approved by the Ministry of Defense and FSA in mid 1998, but the financial collapse in Russia that year apparently ended the already limited flow of government funding. Khrunichev elected to continue Angara's development using its revenues from commercial Proton launches. Khrunichev announced the redesigned Angara at the 1999 Paris Airshow, and shortly thereafter announced that the ILS joint venture with Lockheed Martin, which markets Proton and Atlas, would also market Angara. Khrunichev hoped by designing and flying the smaller configurations first, it could bootstrap the development process using proceeds from the first launches to complete development of the larger vehicles.

Technical development of the vehicle continued relatively smoothly, although the design continued to evolve somewhat. Design documentation was released in 1999 and 2000 allowing the start of construction. The RD-191 main engine was test fired for the first time in 2001. In contrast to the progress on the commercially financed launch vehicle, essentially no progress was being made on the launch site, which was to be funded by both FSA and the Strategic Missile Force. Despite the official priority of the program, funding did not become available until 2001. Now that funding is available for the launch site, the prospects for near-term launches of Angara have improved.

ARIANE



Ariane 4 photo courtesy Aerospatiale. Ariane 5 photo courtesy Aerospatiale Matra Lanceurs.

The Ariane 4 launches successfully (left). Ariane 503 launches 21 October 1998 (right).

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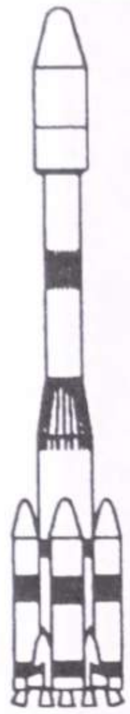
Ariane 4

General Description



Ariane 4

Summary

The Ariane 4 series has been one of the world's most successful commercial launch vehicles since its first launch in 1988. The combination of a near-equatorial launch site at Kourou, the ability to split launch costs between two payloads, and flexible performance using different combinations of strap-on boosters made it a frequent choice for launching commercial communications satellites. In response to increasing satellite masses and dropping prices, the Ariane 4 was phased out, with its final launch February 2003. The larger, less-expensive Ariane 5 will take its place.

Status

Retired. First launch in 1988. Final launch in 2003.

Origin

Europe

Key Organizations

Ariane development programs are funded by ESA and managed by CNES. Once development is complete, responsibility for operational flights is given to Arianespace.

Marketing Organization	Arianespace
Launch Service Provider	Arianespace
Industrial Architect	EADS

Primary Missions

Commercial GTO payloads

Estimated Launch Price

No recent price information is available for Ariane 4 because it is no longer marketed. See the Cost section for historical pricing information.

Spaceport

Launch Site	Guiana Space Center, ELA-2
Location	5.2° N, 52.8° W
Available Inclinations	5.2–100.5 deg (launch azimuth is –10.5–91.5 deg)

Performance Summary

The performance capability quoted below includes the spacecraft mass plus the mass of any required adapter and/or dual-payload structures. Performance of the largest configuration is shown. For information on other Ariane 4 configurations, see the Performance section.

200 km (108 nmi), 5.2 deg	AR44L: 10,200 kg (22,500 lbm)
200 km (108 nmi), 90 deg	AR44L: 8200 kg (18,200 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	AR44L: 6485 kg (14,300 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 7 deg	AR44L: 4770 kg (10,560 lbm)
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

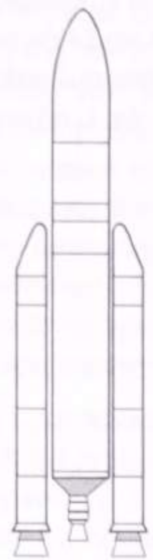
Total Orbital Flights	116
Launch Vehicle Successes	113
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	3

Flight Rate

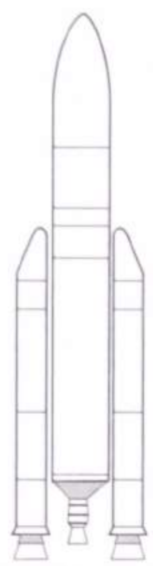
6–9 per year

Ariane 5

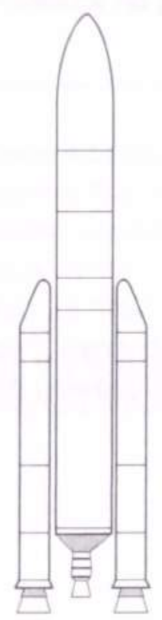
General Description



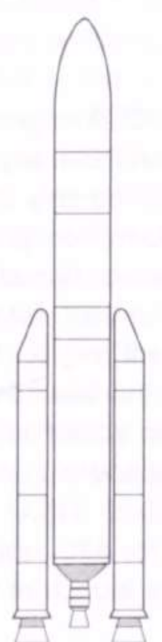
Ariane 5



Ariane 5ES



Ariane 5ECA



Ariane 5ECB

Summary

In 1988 Europe began developing the Ariane 5 as a higher-performance, lower-cost replacement for the Ariane 4. The first launch in 1996 ended in failure, resulting in an extended flight test period that delayed entry into commercial service until 1999. Like Ariane 4, Ariane 5 is designed to provide lower-cost launch services by comanifesting two or more satellites on each launch. Because commercial satellites have increased in mass significantly, the initial Ariane 5G version is being upgraded to the Ariane 5E configuration, and two new cryogenic upper stages are being developed. These changes, introduced beginning in late 2002, will eventually double performance of Ariane 5 to 12 t to GTO.

Status

Ariane 5G: Operational. First Launch in 1996.

Ariane 5ECA: Flight Test. First Launch in 2002.

Ariane 5ES: In Development. First launch planned in 2005.

Ariane 5ECB: In Development. First launch TBD.

Origin

Europe

Key Organizations

Ariane development programs are funded by ESA and managed by CNES. Once development is complete, responsibility for operational flights is given to Arianespace.

Marketing Organization	Arianespace
Launch Service Provider	Arianespace
Industrial Architect	EADS

Primary Missions

Commercial GTO payloads

Estimated Launch Price

\$125–155 million (FAA 2002)

Spaceport

Launch Sites	Guiana Space Center, ELA-3
Location	5.2° N, 52.8° W
Available Inclinations	5.2–100.5 deg (launch azimuth is –10.5–91.5 deg)

Performance Summary

The performance capability quoted below includes the spacecraft mass plus the mass of any required adapter and/or dual-payload structures.

	Ariane 5G	Ariane 5ES	Ariane 5ECA	Ariane 5ECB
200 km (108 nmi), 5.2 deg	?	?	?	?
200 km (108 nmi), 90 deg	?	?	?	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	16,000 kg (35,300 lbm)	20,000 kg (44,110 lbm)	?	?
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	9500 kg (20,950 lbm)	?	?	?
GTO: 580×35,786 km (313×19,323 nmi), 7 deg	6700 kg (14,770 lbm)	7575 kg (16,700 lbm)	10,050 kg (23,150 lbm)	12,000 kg (26,500 lbm)
Geostationary Orbit	No capability	No capability	No capability	?

Flight Record (through 31 December 2003)

	Ariane 5G	Ariane 5E
Total Orbital Flights	16	1
Launch Vehicle Successes	13	0
Launch Vehicle Partial Failures	1	0
Launch Vehicle Failures	2	1

Flight Rate

1–4 per year.

Nomenclature

Ariane is the French spelling for the heroine from Greek mythology usually called Ariadne in English. Ariane, the daughter of Minos of Knossos, gave Theseus the thread with which to find his way out of the Minotaur's labyrinth. The current Ariane configurations are the Ariane 4 and Ariane 5. The many variations of the Ariane 4 can be distinguished by the vehicle designation, explained below. Each Ariane stage has an alphanumeric designation reflecting the type of propellant and the approximate propellant mass in metric tons. The letter P refers to solid propellants, L refers to liquid hydrazine/N_2O_4, and H refers to LH_2/LOX propellants. For example, the L220 stage carries 220 t of hydrazine/N_2O_4.

Example **AR4 4 L 0 2 1 -3**

Designates Ariane 4 vehicle

Upper Stage Enhancement
+ = H10+ stretched third stage
-3 = H10-3 third stage

Number of strap-on boosters
(0, 2, or 4)

Sylda multi-payload structure
0 = No Sylda
1 = Sylda 4400

Type of strap-on booster
P = solid
L = liquid
LP = solid and liquid

Fairing Type
1 = Short Fairing, 8.6 m (28.2 ft)
2 = Long Fairing, 9.6 m (31.5 ft)
3 = Extra Long Fairing, 11.1 m (36.4 ft)

Spelda multi-payload structure
0 = No Spelda
1 = Spelda
3 = Mini-Spelda

The initial version of Ariane 5 is designated the 5G, for generic. It is also referred to as the baseline Ariane 5. The upgraded version, developed under the Ariane 5 Evolution program is designated Ariane 5E. The Ariane 5E is available with three upper stages. Configurations with the original storable-propellant upper stage are designated 5ES. Vehicles with a cryogenic upper stage are designated 5ECA or 5ECB depending on whether the Type A or Type B cryogenic stage is used. The Ariane 5ECA has also been referred to as the 10-ton Ariane 5.

Cost

Because Ariane launches are frequently conducted with two comanifested payloads, the typical cost for each launch vehicle is not well defined. Prices also vary depending on desired services, contract terms and conditions, and market competition. Ariane 4 is no longer marketed, so the following price data is for historical reference only, based on information provided by Arianespace in 1999.

AR40	$65–85 million
AR42P	$70–90 million
AR44P	$80–100 million
AR42L	$80–100 million
AR44LP	$90–110 million
AR44L	$100–125 million

In 2002 the FAA estimated current Ariane 5 prices to be $125–155 million, based upon open source information.

Prices for a number of Ariane launch contracts have been publicly disclosed, particularly those for European government payloads. ESA originally agreed to pay 134 million ECU (about $170 million in 1996) for the second Ariane 5 flight, but Arianespace reduced the price by 50 million ECU as part of its contribution to the recovery from the failure of the first flight. ESA paid $154 million for the launch of the XMM space telescope as the only passenger on an Ariane 5G in 1999. For the launch of Rosetta, as a single passenger on an Ariane 5G scheduled for 2003, ESA paid 120 million ($118 million). The launch of Envisat on a dedicated Ariane 5G in 2002 cost 140 million ($137 million). The 3100 kg (6800 lbm) Artemis satellite was launched as a copassenger on an Ariane 5G for 80 million ($79 million) in 2001. Finally, ESA signed a contract worth about 1 billion ($1 billion) in 2000 for the launch of nine Automated Transfer Vehicles (ATV) on Ariane 5ESV rockets between 2003 and 2014. CNES paid 122 million ($120 million) for the launch of SPOT 5 on a dedicated Ariane 42P in 2002. The French space agency also paid 70 million ($69 million) for a shared ride on the first Ariane 5E to launch the 2200 kg (4850 lbm) Stentor spacecraft. Apparently, CNES did not receive the type of discount that is frequently offered to attract a customer to the first launch of a new vehicle type. Indian officials have also announced the cost for several of their contracts with Arianespace to launch INSAT series satellites. The cost of launching the 2550 kg (5600 lbm) INSAT 2E on a dedicated Ariane 42P in 1999 was around $69 million. INSAT 3A was launched in early 2003 for $74 million, while the 2750 kg (6060 lbm) INSAT 3C was launched on a dedicated Ariane 42L in 2002 for $70 million. In late 2002, PT Telekomunikasi Indonesia disclosed in quarterly financial documents that it had signed a $62.9 million contract for the launch of the Telkom 2 satellite in late 2004.

Cost

Arianespace has survived three consecutive years of financial losses beginning in 2000. This had a variety of causes, including increased competition, and the cost of operating two launch vehicles – Ariane 4 and Ariane 5 – at the same time. In response, a great deal of emphasis has been placed on reducing the cost of the Ariane 5. The first production batch of vehicles, dubbed P1 and ordered in 1995, cost more than 12 billion French francs, equal to about $2.4 billion or roughly $170 million per vehicle using the exchange rate at the time. The second batch of vehicles ordered by Arianespace was reduced in cost by 35% compared to the first batch. For example, Cryospace, which produces the first stage tanks, has improved manufacturing efficiency to reduce its cost from an average of 6.4 million each in the first batch to 4.5 million in the second batch, and cut production time in half. Vulcain 2 engines for the second batch were required to cost less than the original Vulcain engine, despite having higher performance. The third batch order, currently being negotiated, is expected to cost about half the price per vehicle of the first batch. Even this may not be enough, however. Therefore, European governments are also considering ways to provide up to $150 million per year in increased financial support for Ariane. Proposals include paying part of the cost for operations at the launch site, which currently cost Arianespace around $10 million per launch, or requiring European governments to use Ariane for government funded payloads, as the United States does, or even guaranteeing a minimum number of Ariane launches per year.

The total development costs for the Ariane 5G were roughly $8–9 billion dollars. The Ariane 5 Evolution program was budgeted at 1.026 billion ECU in 1995 ($1.3 billion dollars at 1995 exchange rates). Development of the cryogenic upper stages was initiated separately under the Ariane 5 Plus program, budgeted at 1.3 billion ECU ($1.4 billion in 1998). In the past the development costs for Ariane have been paid for by ESA member governments. Recently, however, industry has begun to share a portion of the cost. Arianespace paid for many of the small improvements in the Perfo 2000 program which gradually upgraded the Ariane 5G, and also contributed to the cost of building the new S5 satellite processing facility at Kourou. Industrial contractors also paid a portion of the roughly $350 million dollar cost associated with recovering from the failure of the first Ariane 5.

Arianespace shareholders

Country	Percentage
France	57.7
Germany	18.4
Italy	7.2
Belgium	4.2
Switzerland	2.58
Spain	2.49
Sweden	2.29
United Kingdom	2.12
The Netherlands	1.97
Denmark	0.58
Norway	0.3
Ireland	0.17

Availability

Ariane 4 began operation since 1988, but it has been phased out in favor of the Ariane 5. The last Ariane 4 was launched in early 2003. The Ariane 5G became the primary configuration in use as of 2003. It is to be replaced by the Ariane 5ECA. However, following the failure of the first Ariane 5ECA vehicle, six additional Ariane 5Gs were ordered to ensure continuity. The Ariane 5ECB was expected to fly for the first time in 2006, but following the failure of the first Ariane 5ECA in December 2002, the funding that had been earmarked to fund the Vinci engine development was reassigned to the return to flight effort. As a result, first flight of the Ariane 5ECB has been postponed to around 2009.

With the addition of a second mobile launch table, Ariane 5 can now support a flight rate of about 8 launches per year. Manufacturing facilities are also designed for this rate. Arianespace has demonstrated contract-to-launch times of only a few months.

Performance

The Ariane 4 was developed specifically to carry commercial payloads to GTO, and can be configured with different combinations of strap-on boosters to tailor the performance to mission requirements. Performance to GTO was improved in the decade following the first Ariane 4 launch, thanks primarily to improvements in the upper stage. Because Ariane frequently carries two payloads, it can still be an appropriate choice for satellites that are much lighter than the total launch vehicle capacity. GTO payloads are typically delivered to an inclination of 7 deg rather than 5.2 deg because Ariane must fly slightly north before turning east. Spacecraft carrying large amounts of propellant pose a larger public risk, requiring Ariane to fly slightly farther north before turning to the desired flight azimuth to avoid range safety concerns around the city Kourou. Performance values shown here are for spacecraft with no more than 450 kg (990 lbm) of hydrazine or 600 kg (1320 lbm) of oxidizer in any single tank. As with many other launch systems, Arianespace has begun offering higher performance by reducing the flight performance reserve propellants retained on the third stage. This results in increased performance, at the risk of somewhat larger variations in the orbit apogee altitude achieved. The option is therefore suitable for spacecraft with liquid propulsion systems that can adjust their final orbit accordingly. Rather than performing a Minimum Residual Shutdown (MRS) in which the propellants are burned to depletion, Arianespace simply reduces the amount of propellant margin, typically to provide a 50% probability of guidance command shutdown rather than the full 99%. Further performance tailoring can be performed by raising or lowering the orbital perigee between 185 and 250 km (100 and 135 nmi), or by reducing the orbital inclination. For example, the maximum payload carried to GTO by an Ariane 4 is 4947 kg (10,906 lbm) on flight V113, 177 kg (390 lbm) higher than the standard capability.

PERFORMANCE

Like Ariane 4, Ariane 5 is primarily used to carry multiple spacecraft to GTO or other high-energy orbits. The core EPC stage burns out in a near-orbital trajectory with an apogee of several hundred kilometers, and the upper stage then fires to place the spacecraft into GTO. As a result, the standard GTO for Ariane 5G and 5ES has a perigee around 560 km (303 nmi), though this can be varied to tailor the trajectory. Like Ariane 4, Ariane 5 performance can be increased with reduced performance reserves and variations in the orbital targets.

Because injection into GTO must be done near the equator, most other launch vehicles in Ariane's class have upper stages with restartable engines, which can inject into a parking orbit, coast to the equator, and ignite again for injection into GTO. Ariane has never needed this capability, since its near-equatorial launch site allows the upper stage to reach a low orbit and continue thrusting directly into GTO without stopping. As a result, no previous Ariane configuration has had a restartable upper stage. However, future missions, such as launching satellites directly into GEO, deploying the Galileo constellation in MEO, or delivering the ATV to the International Space Station, will require a more flexible mission profile. The "Versatile" version of the EPS stage, and the ESC-B stage, will both have a multiple restart capability in order to perform these missions.

Performance capabilities shown below include the mass of the spacecraft, a dual launch structure (if needed), and payload adapter(s). Adapters typically have a mass of 50–75 kg (110–165 lbm) for Ariane 4 and 75–100 kg (165–220 lbm) for Ariane 5. The masses of standard dual-payload structure are provided in the Payload Fairing section. The upper stage flight performance reserve provides a 99% probability of injection into the target orbit, unless otherwise noted. Available inclinations range from 5.2 deg to 100.5 deg.

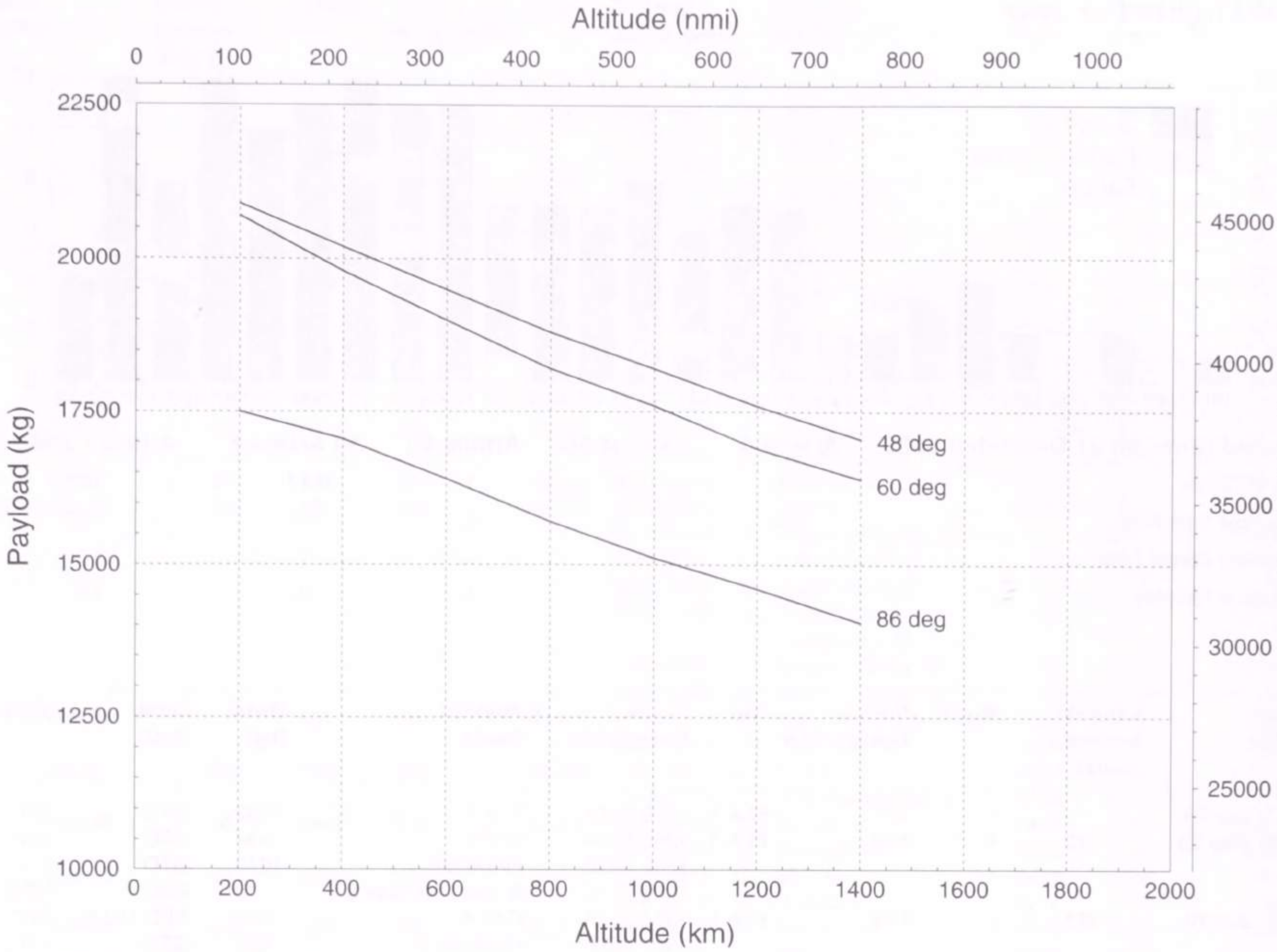
Ariane 4

	200 km (108 nmi), 5.2 deg	**200 km (108 nmi), 90 deg**	**Space Station Orbit: 407 km (220 nmi), 51.6 deg**	**Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg**	**GTO: 185×35,786 km (100×19,323 nmi), 7 deg**
AR40	5000 kg (11,000 lbm)	4000 kg (8800 lbm)	—	2845 kg (6270 lbm)	2175 kg (4895 lbm)
AR42P	6600 kg (14,600 lbm)	5200 kg (11,500 lbm)	—	3845 kg (8475 lbm)	2890 kg (6370 lbm)
AR44P	7600 kg (16,800 lbm)	6000 kg (13,200 lbm)	—	4560 kg (10,050 lbm)	3465 kg (7640 lbm)
AR42L	7900 kg (17,400 lbm)	6200 kg (13,700 lbm)	—	4810 kg (10,600 lbm)	3590 kg (7910 lbm)
AR44LP	9100 kg (20,000 lbm)	7200 kg (15,900 lbm)	—	5660 kg (12,475 lbm)	4290 kg (9460 lbm)
AR44L	10,200 kg (22,500 lbm)	8200 kg (18,100 lbm)	—	6485 kg (14,300 lbm)	4790 kg (10,560 lbm)

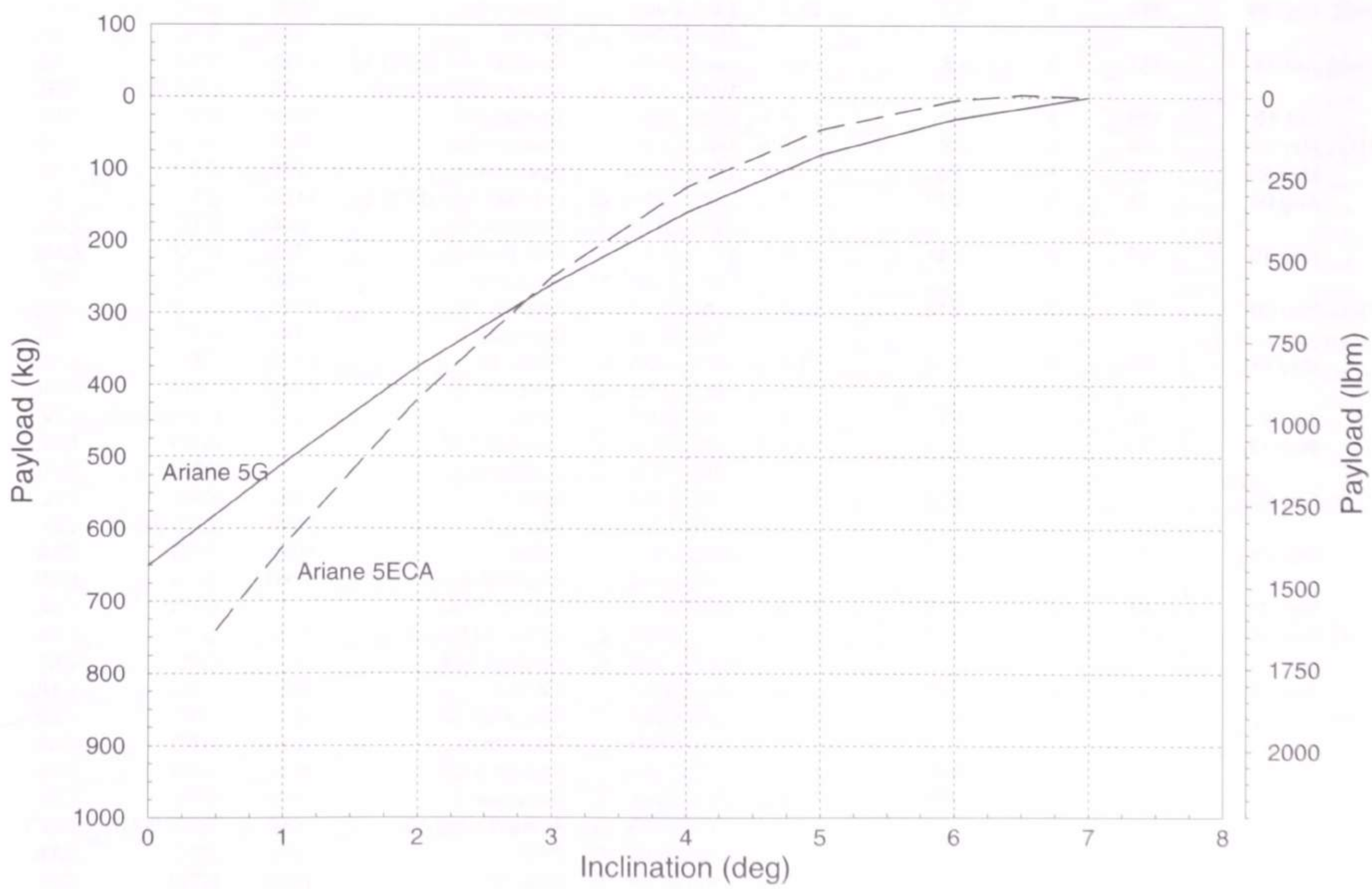
Ariane 5

	Ariane 5G	**Ariane 5ES**	**Ariane 5ECA**	**Ariane 5ECB**
200 km (108 nmi), 5.2 deg	?	?	?	?
200 km (108 nmi), 90 deg	?	?	?	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	16,000 kg (35,300 lbm)	20,000 kg (44,100 lbm)	?	?
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	9500 kg (20,950 lbm)	?	?	?
GTO: 580x 35,786 km (313 x 19,323 nmi), 7 deg	6700 kg (14,770 lbm) D	7575 kg (16,700 lbm)	10,050 kg (23,150 lbm) (250 km, 135 nmi perigee)	12,00 kg (26,500 lbm)
Geosynchronous Orbit	No capability	No capability	No capability	?

Performance



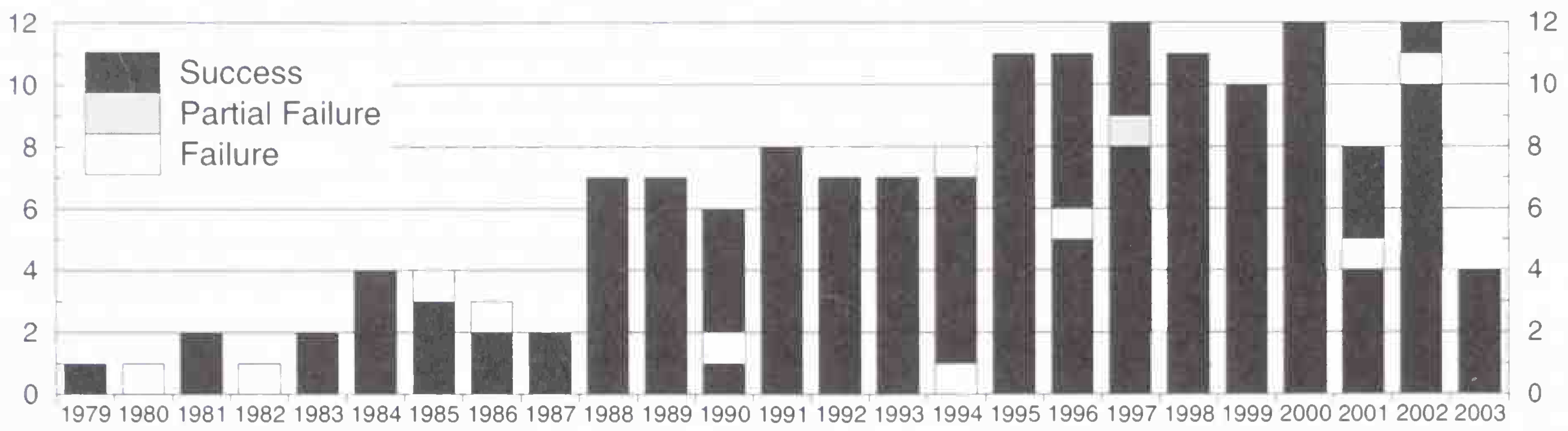
Ariane 5ES: Performance in LEO



Ariane 5G and 5ECA: GTO Performance Sensitivity to Inclination

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Ariane 4	Ariane 5G	Ariane 5E	All Ariane 5	Ariane Family Total
Total Orbital Flights	116	16	1	17	161
Launch Vehicle Successes	113	13	0	13	150
Launch Vehicle Partial Failures	0	1	0	1	1
Launch Vehicle Failures	3	2	1	3	10

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1	1979 Dec 24	—	1	L01	ELA 1	1979 104A		CAT 1	1602	GTO	CIV	Europe
F,T	**2**	**1980 May 23**	**152**	**1**	**L02**	**ELA 1**	1980 F01A		CAT 2	334	GTO	CIV	Europe
							1980 F01B		Firewheel	1073	GTO	CIV	Germany
							1980 F01C	A	Amsat 3A/Oscar 9	75	EEO	NGO	
T	3	1981 Jun 19	393	1	L03	ELA 1	1981 057A		CAT 3	266	EEO (10.4)	CIV	Europe
							1981 057B	C	Meteosat 2	697	GTO	CIV	Europe
							1981 057C	C	Apple	670	GTO	CIV	India
T	4	Dec 20	185	1	L04	ELA 1	1981 122A		CAT 4	1080	GTO	CIV	Europe
							1981 122B	A	MARECS A	217	EEO (10.4)	CIV	Europe
								A	Thesee	39	GTO	CIV	
F	**5**	**1982 Sep 09**	**264**	**1**	**L5**	**ELA 1**	1982 F06A	C	Marecs B	1080	GTO	CIV	Europe
							1982 F06B	C	Sirio 2	419	GTO	CML	Italy
	6	1983 Jun 16	281	1	L6	ELA 1	1983 058A		Eutelsat 101 (ECS 1)	1185	GTO	CML	Europe
S							1983 058B	A	Amsat 3B/Oscar 10	70	EEO (27.1)	NGO	Europe
	7	Oct 19	126	1	L7	ELA 1	1983 105A		Intelsat 507	1928	GTO	CML	Int'l
	8	1984 Mar 05	139	1	L8	ELA 1	1984 023A		Intelsat 508	1928	GTO	CML	Int'l
	9	May 23	80	1	V9	ELA 1	1984 049A		Spacenet 1	705	GTO	CML	USA
	10	Aug 04	74	3	V10	ELA 1	1984 081A	C	Eutelsat 102 (ECS 2)	1185	GTO	CML	Europe
							1984 081B	C	Telecom 1A	690	GTO	CML	France
	11	Nov 10	99	3	V11	ELA 1	1984 114A	C	Spacenet 2	705	GTO	CML	USA
							1984 114B	C	Marecs 2	1080	GTO	CIV	Europe
	12	1985 Feb 08	91	3	V12	ELA 1	1985 015A	C	Arabsat 1A	1170	GTO	CML	Mid East
							1985 015B	C	Brasilsat 1	1140	GTO	CML	Brazil
	13	May 08	90	3	V13	ELA 1	1985 035A	C	GStar 1A	1270	GTO	CML	USA
							1985 035B	C	Telecom 1B	1210	GTO	CML	France
	14	Jul 02	56	1	V14	ELA 1	1985 056A		Giotto	512	Comet Halley	CIV	Europe
F	**15**	**Sep 12**	**73**	**3**	**V15**	**ELA 1**	1985 F03A	C	Eutelsat 103		GTO	CML	Europe
							1985 F03B	C	Spacenet 3		GTO	CML	USA
	16	1986 Feb 22	164	1	V16	ELA 1	1986 019A	C	Spot 1	1830	SSO	CIV	France
							1986 019B	C	Viking 1	538	EEO (98.8)	CIV	Sweden
	17	Mar 28	35	3	V17	ELA 2	1986 026A	C	GStar 2	1243	GTO	CML	USA
							1986 026B	C	Brasilsat 2	1140	GTO	CML	Brazil
F	**18**	**May 31**	**65**	**2**	**V18**	**ELA 1**	1986 F05A		Intelsat 514A		GTO	CML	Int'l
	19	1987 Sep 16	474	3	V19	ELA 1	1987 078A	C	Optus 3 (Aussat 3)	1195	GTO	CML	Australia
							1987 078B	C	Eutelsat 104	1175	GTO	CML	Europe
S	20	Nov 21	67	2	V20	ELA 2	1987 095A		TV-Sat 1	2077	GTO	CML	Germany
	21	1988 Mar 11	112	3	V21	ELA 1	1988 018A	C	Spacenet 3R	1212	GTO	CML	USA
							1988 018B	C	Telecom 1C	1210	GTO	CML	France
	22	May 17	68	2	V23	ELA 1	1988 040A		Intelsat 513A	2013	GTO	CML	Int'l
	23	Jun 15	30	44LP	V22	ELA 2	1988 051A	C	Meteosat 3	696	GTO	CIV	Europe
							1988 051B	A	Amsat/Oscar 13	150	EEO (57.4)	NGO	USA
							1988 051C	C	PAS 1	1220	GTO	CML	USA
	24	Jul 21	37	3	V24	ELA 1	1988 063A	C	Insat 1C	1190	GTO	CIV	India
							1988 063B	C	Eutelsat 105 (ECS 5)	1185	GTO	CML	Europe

ELA-1, -2, and -3 are at the Guiana Space Center near Kourou, French Guiana

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
S	25	Sep 08	50	3	V25	ELA 2	1988 081A	C	GStar 3	1270	GTO	CML	USA
							1988 081B	C	SBS 5	1241	GTO	CML	USA
	26	Oct 28	51	2	V26	ELA 1	1988 098A		TDF 1	2136	GTO	CML	France
	27	Dec 11	45	44LP	V27	ELA 2	1988 109A	C	Skynet 4B	1433	GTO	MIL	UK
							1988 109B	C	Astra 1A	1780	GTO	CML	Luxem.
	28	1989 Jan 27	48	2	V28	ELA 1	1989 006A		Intelsat 515A	1977	GTO	CML	Int'l
	29	Mar 06	39	44LP	V29	ELA 2	1989 020A	C	JCSat 1	2280	GTO	CML	Japan
							1989 020B	C	MOP 1 (Meteosat 4)	681	GTO	CIV	Europe
	30	Apr 02	28	2	V30	ELA 1	1989 027A		Tele-X	2142	GTO	CML	Sweden
	31	Jun 05	65	44L	V31	ELA 2	1989 041A	C	Superbird A	2489	GTO	CML	Japan
							1989 041B	C	DFS Kopernikus 1	1416	GTO	CML	Germany
	32	Jul 12	38	3	V32	ELA 1	1989 053A		Olympus	2595	GTO	CIV	Europe
	33	Aug 08	28	44LP	V33	ELA 2	1989 062A	C	TV-Sat 2	2130	GTO	CML	Germany
S							1989 062B	C	Hipparcos	1130	GTO	CIV	Europe
	34	Oct 27	81	44L	V34	ELA 2	1989 087A		Intelsat 602	4215	GTO	CML	Int'l
	35	1990 Jan 22	88	40	V35	ELA 2	1990 005A		Spot 2	1870	SSO	CIV	France
							1990 005B	A	Webersat/Oscar 18	12	SSO	NGO	USA
							1990 005C	A	Lusat/Oscar 19	12	SSO	NGO	Argentina
							1990 005D	A	Dove/Oscar 17	12	SSO	NGO	Brazil
							1990 005E	A	Pacsat/Oscar 16 (Microsat 1)	12	SSO	NGO	USA
							1990 005F	A	Uosat 3/Oscar 14	46	SSO	NGO	UK
							1990 005G	A	Uosat 4/Oscar 15	48	SSO	NGO	UK
F	**36**	**Feb 22**	**32**	**44L**	**V36**	**ELA 2**	1990 F01A	C	Yuri 2X (BS 2X-NHK)	1250	GTO	CML	Japan
							1990 F01B	C	Superbird 1B	2496	GTO	CML	Japan
	37	Jul 24	153	44L	V37	ELA 2	1990 063A	C	TDF 2	2096	GTO	CML	France
							1990 063B	C	DFS Kopernikus 2	1418	GTO	CML	Germany
	38	Aug 30	38	44LP	V38	ELA 2	1990 079A	C	Skynet 4C	1463	GTO	MIL	UK
							1990 079B	C	Eutelsat 201	1878	GTO	CML	Europe
	39	Oct 12	44	44L	V39	ELA 2	1990 091A	C	SBS 6	2478	GTO	CML	USA
							1990 091B	C	Galaxy 6	1212	GTO	CML	USA
	40	Nov 20	40	42P	V40	ELA 2	1990 100A	C	Satcom C1	1169	GTO	CML	USA
							1990 100B	C	GStar 4	1295	GTO	CML	USA
	41	1991 Jan 15	57	44L	V41	ELA 2	1991 003A	C	Italsat 1	1865	GTO	CML	Italy
							1991 003B	C	Eutelsat 202	1878	GTO	CML	Europe
	42	Mar 02	47	44LP	V42	ELA 2	1991 015A	C	Astra 1B	2620	GTO	CML	Luxem.
							1991 015B	C	MOP 2 (Meteosat 5)	681	GTO	CIV	Europe
	43	Apr 04	34	44P	V43	ELA 2	1991 026A		Anik E2	2923	GTO	CML	Canada
	44	Jul 17	105	40	V44	ELA 2	1991 050A		ERS 1	2384	SSO	CIV	Europe
							1991 050B	A	UOSat 5/Oscar 22	50	SSO	NGO	UK
S							1991 050C	A	Orbcomm X	22	SSO	CML	USA
							1991 050D	A	Tubsat A	38	SSO	NGO	Germany
							1991 050E	A	SARA	26	SSO	NGO	France
	45	Aug 14	29	44L	V45	ELA 2	1991 055A		Intelsat 605	4296	GTO	CML	Int'l
	46	Sep 26	44	44P	V46	ELA 2	1991 067A		Anik E1	2977	GTO	CML	Canada
	47	Oct 29	34	44L	V47	ELA 2	1991 075A		Intelsat 601	4330	GTO	CML	Int'l
	48	Dec 16	49	44L	V48	ELA 2	1991 085A	C	Telecom 2A	2275	GTO	CML	France
							1991 085B	C	Inmarsat 203	1310	GTO	CML	Int'l
	49	1992 Feb 26	73	44L	V49	ELA 2	1992 010A	C	Superbird 1BR	2560	GTO	CML	Japan
							1992 010B	C	Arabsat 1C	1310	GTO	CML	Mid East
	50	Apr 15	50	44L+	V50	ELA 2	1992 021A	C	Telecom 2B	2275	GTO	CML	France
							1992 021B	C	Inmarsat 204	1310	GTO	CML	Int'l
	51	Jul 09	86	44L	V51	ELA 2	1992 041A	C	Insat 2A	1906	GTO	CIV	India
							1992 041B	C	Eutelsat 204	1877	GTO	CML	Europe
	52	Aug 10	33	42P	V52	ELA 2	1992 052A		Topex/Poseidon	2402	LEO (66)	CIV	USA
							1992 052B	A	Uribyol 1/Oscar 23 (Kitsat A)	50	LEO (66)	NGO	Korea
							1992 052C	A	S80/T	50	LEO (66)	CIV	France
	53	Sep 10	32	44LP+	V53	ELA 2	1992 060A	C	Hispasat 1A	2194	GTO	CML	Spain
							1992 060B	C	Satcom C3	1375	GTO	CML	USA
	54	Oct 28	49	42P+	V54	ELA 2	1992 072A		Galaxy 7	2968	GTO	CML	USA
	55	Dec 01	35	42P+	V55	ELA 2	1992 085A		Superbird 1AR	2780	GTO	CML	Japan
	56	1993 May 12	163	42L	V56	ELA 2	1993 031A		Astra 1C	2790	GTO	CML	Luxem.
S							1993 031B	A	Arsene	154	EEO (1.4)	NGO	France
	57	Jun 25	45	42P+	V57	ELA 2	1993 040A		Galaxy 4	2988	GTO	CML	USA
	58	Jul 22	28	44L+	V58	ELA 2	1993 048A	C	Hispasat 1B	2120	GTO	CML	Spain
							1993 048B	C	Insat 2B	1931	GTO	CIV	India
	59	Sep 26	67	40	V59	ELA 2	1993 061A		SPOT 3	1907	SSO	CIV	France
							1993 061B	A	Stella	48	SSO	CIV	France
							1993 061C	A	Eyesat 1	13	SSO	CML/NGO	USA

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
							1993 061D	A	Itamsat	12	SSO	NGO	Italy
							1993 061E	A	Healthsat 1	50	SSO	NGO	USA
							1993 061F	A	Posat 1	50	SSO	CIV	Portugal
							1993 061G	A	Uribyol 2 (Kitsat 2)	50	SSO	NGO	Korea
	60	Oct 22	27	44LP	V60	ELA 2	1993 066A		Intelsat 701	3650	GTO	CML	Int'l
	61	Nov 20	30	44LP+	V61	ELA 2	1993 073A	C	Solidaridad 1	2776	GTO	CML	Mexico
							1993 073B	C	Meteosat 6	704	GTO	CIV	Europe
	62	Dec 18	29	44L+	V62	ELA 2	1993 078A	C	DBS 1	2860	GTO	CML	USA
							1993 078B	C	Thaicom 1	1080	GTO	CML	Thailand
F	**63**	**1994 Jan 24**	**38**	**44LP+**	**V63**	**ELA 2**	1994 F01A	C	Turksat 1A	1783	GTO	CML	Turkey
							1994 F01B	C	Eutelsat 205	1880	GTO	CML	Europe
	64	Jun 17	145	44LP+	V64	ELA 2	1994 034A		Intelsat 702	3695	GTO	CML	Int'l
							1994 034B	A	STRV 1A	50	EEO (6.8)	MIL	UK
							1994 034C	A	STRV 1B	53	EEO (7.5)	MIL	UK
	65	Jul 08	22	44L+	V65	ELA 2	1994 040A	C	PAS 2	2290	GTO	CML	USA
							1994 040B	C	Yuri 3N (BS 3N - NHK)	1210	GTO	CML	Japan
	66	Aug 11	35	44LP+	V66	ELA 2	1994 049A	C	Brasilsat B1	1765	GTO	CML	Brazil
							1994 049B	C	Turksat 1B	1779	GTO	CML	Turkey
S	67	Sep 09	30	42L+	V67	ELA 2	1994 058A		Telstar 402	3331	GTO	CML	USA
	68	Oct 08	30	44L+	V68	ELA 2	1994 065A	C	Solidaridad 2	2776	GTO	CML	Mexico
							1994 065B	C	Thaicom 2	1080	GTO	CML	Thailand
	69	Nov 01	25	42P+	V69	ELA 2	1994 070A		Astra 1D	2924	GTO	CML	Luxem.
F	**70**	**Dec 01**	**31**	**42P-3**	**V70**	**ELA 2**	1994 F04A		PAS 3	2985	GTO	CML	USA
	71	1995 Mar 28	118	44LP+	V71	ELA 2	1995 016A	C	Brasilsat B2	1780	GTO	CML	Brazil
							1995 016B	C	Hotbird 1	1800	GTO	CML	Europe
	72	Apr 21	25	40+	V72	ELA 2	1995 021A		ERS 2	2516	SSO	CIV	Europe
	73	May 17	27	44LP-3	V73	ELA 2	1995 023A		Intelsat 706A	4180	GTO	CML	Int'l
	74	Jun 10	25	42P-3	V74	ELA 2	1995 029A		DBS 3	2934	GTO	CML	USA
	75	Jul 07	28	40-3	V75	ELA 2	1995 033A		Helios 1A	2537	SSO	MIL	France
							1995 033B	A	Cerise	50	SSO	MIL	France
							1995 033C	A	UPM/Sat 1	47	SSO	NGO	Spain
	76	Aug 03	28	42L-3	V76	ELA 2	1995 040A		PAS 4	3043	GTO	CML	USA
	77	Aug 29	27	44P-3	V77	ELA 2	1995 044A		N-Star A	3410	GTO	CML	Japan
	78	Sep 24	27	42L-3	V78	ELA 2	1995 049A		Telstar 402R	3410	GTO	CML	USA
	79	Oct 19	26	42L-3	V79	ELA 2	1995 055A		Astra 1E	3010	GTO	CML	Luxem.
	80	Nov 17	30	44P-3	V80	ELA 2	1995 062A		ISO	2498	EEO (4.9)	CIV	Europe
	81	Dec 06	20	44L-3	V81	ELA 2	1995 067A	C	Insat 2C	2050	GTO	CIV	India
							1995 067B	C	Telecom 2C	2283	GTO	CML	France
	82	1996 Jan 12	38	44L-3	V82	ELA 2	1996 002A	C	PAS 3R	2918	GTO	CML	USA
							1996 002B	C	Measat 1	1450	GTO	CML	Malaysia
	83	Feb 05	25	44P-3	V83	ELA 2	1996 007A		NStar B	3420	GTO	CML	Japan
	84	Mar 14	39	44LP-3	V84	ELA 2	1996 015A		Intelsat 707	4175	GTO	CML	Int'l
	85	Apr 20	38	42P-3	V85	ELA 2	1996 022A		MSat 1	2855	GTO	CML	Canada
	86	May 16	27	44L-3	V86	ELA 2	1996 030A	C	Palapa C2	2989	GTO	CML	Indonesia
							1996 030B	C	Amos 1	996	GTO	CML	Israel
F,T	**87**	**Jun 04**	**20**	**5G**	**V88 (501)**	**ELA 3**	1996 F03	M	Cluster 1 thru 4	4@1180		CIV	Europe
	88	Jun 15	12	44P-3	V87	ELA 2	1996 035A		Intelsat 709	3420	GTO	CML	Int'l
	89	Jul 09	25	44L-3	V89	ELA 2	1996 040A	C	Arabsat 2A	2617	GTO	CML	Mid East
							1996 040B	C	Turksat 1C	1743	GTO	CML	Turkey
	90	Aug 08	31	44L-3	V90	ELA 2	1996 044A	C	Italsat 2	1990	GTO	CML	Italy
							1996 044B	C	Telecom 2D	2260	GTO	CML	France
	91	Sep 11	35	42P-3	V91	ELA 2	1996 055A		Echostar 2	2885	GTO	CML	USA
	92	Nov 13	64	44L-3	V92	ELA 2	1996 063A	C	Arabsat 2B	1433	GTO	CML	Mid East
							1996 063B	C	Measat 2	1436	GTO	CML	Malaysia
	93	1997 Jan 30	79	44L-3	V93	ELA 2	1997 002A	C	GE 2	2580	GTO	CML	USA
							1997 002B	C	Nahuel 1A	1817	GTO	CML	Argentina
	94	Mar 01	31	44P-3	V94	ELA 2	1997 009A		Intelsat 801	3239	GTO	CML	Int'l
	95	Apr 16	47	44LP-3	V95	ELA 2	1997 016A	C	Thaicom 3	2645	GTO	CML	Thailand
							1997 016B	C	BSat 1A	1248	GTO	CML	Japan
	96	Jun 03	49	44L-3	V97	ELA 2	1997 027A	C	Inmarsat 304	2064	GTO	CML	Int'l
							1997 027B	C	Insat 2D	2046	GTO	CIV	India
	97	Jun 25	23	44P-3	V96	ELA 2	1997 031A		Intelsat 802	3239	GTO	CML	Int'l
	98	Aug 08	45	44P-3	V98	ELA 2	1997 040A		PAS 6	3015	GTO	CML	USA
	99	Sep 02	26	44LP-3	V99	ELA 2	1997 049A	C	Hot Bird 3	2895	GTO	CML	Europe
							1997 049B	C	Meteosat 7	704	GTO	CIV	Europe
	100	Sep 23	22	42L-3	V100	ELA 2	1997 053A		Intelsat 803	3239	GTO	CML	Int'l
P,T	**101**	**Oct 30**	**38**	**5G**	**V101 (502)**	**ELA 3**	1997 066A	C	Maqsat H/TEAMSAT	2300	GTO	CIV	Europe
							1997 066B	C	Maqsat B	1400	GTO	CIV	Europe
							1997 066C	A	YES	160	GTO	CIV	Europe
	102	Nov 12	14	44L-3	V102	ELA 2	1997 071A	C	Sirius 2	2883	GTO	CML	Sweden
							1997 071B	C	Cakrawarta 1	1417	GTO	CML	Indonesia

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Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
									(Indostar 1)				
	103	Dec 02	21	44P-3	V103	ELA 2	1997 075A		JCSat 5	2982	GTO	CML	Japan
							1997 075B	A	Equator S	230	EEO (4)	CIV	Germany
	104	Dec 22	21	42L-3	V104	ELA 2	1997 083A		Intelsat 804	3239	GTO	CML	Int'l
	105	1998 Feb 04	45	44LP-3	V105	ELA 2	1998 006A	C	Brasilsat B3	1760	GTO	CML	Brazil
							1998 006B	C	Inmarsat 305	2000	GTO	CML	Int'l
	106	Feb 27	24	42P-3	V106	ELA 2	1998 013A		Hot Bird 4	2885	GTO	CML	Europe
	107	Mar 24	26	40-3	V107	ELA 2	1998 017A		Spot 4	2755	SSO	CIV	Europe
	108	Apr 28	36	44P-3	V108	ELA 2	1998 024A	C	Nilesat 101	840	GTO	CML	Egypt
							1998 024B	C	BSat 1B	1230	GTO	CML	Japan
	109	Aug 25	120	44P-3	V109	ELA 2	1998 049A		ST 1	3255	GTO	CML	Singapore
	110	Sep 16	23	44LP-3	V110	ELA 2	1998 052A		PAS 7	3838	GTO	CML	USA
	111	Oct 05	20	44L-3	V111	ELA 2	1998 056A	C	Eutelsat W2	2950	GTO	CML	Europe
							1998 056B	C	Sirius 3	1420	GTO	CML	Sweden
T	112	Oct 21	17	5G	V112 (503)	ELA 3	1998 059A	C	Maqsat 3	2730	GTO	CIV	Europe
							none	C	ARD 1	2750	suborbital	CIV	Europe
	113	Oct 28	8	44L-3	V113	ELA 2	1998 063A	C	Afristar 1	2793	GTO	CML	
							1998 063B	C	GE 5	1698	GTO	CML	USA
	114	Dec 06	40	42L-3	V114	ELA 2	1998 070A		Satmex 5	4135	GTO	CML	Mexico
	115	Dec 22	17	42L-3	V115	ELA 2	1998 075A		PAS 6B	3594	GTO	CML	USA
	116	1999 Feb 26	66	44L-3	V116	ELA 2	1999 009A	C	Arabsat 3A	2708	GTO	CML	mid-East
							1999 009B	C	Skynet 4E	1490	GTO	MIL	UK
	117	Apr 02	35	42P-3	V117	ELA 2	1999 016A		Insat 2E	2550	GTO	CIV	India
	118	Aug 12	132	42P-3	V118	ELA 2	1999 042A		Telkom 1	2763	GTO	CML	Indonesia
	119	Sep 04	23	42P-3	V120	ELA 2	1999 046A		Koreasat 3	2790	GTO	CML	South Korea
	120	Sep 25	21	44LP-3	V121	ELA 2	1999 052A		Telstar 7	3790	GTO+	CML	USA
	121	Oct 19	24	44LP-3	V122	ELA 2	1999 059A		Orion 2	3788	GTO	CML	USA
	122	Nov 13	25	44LP-3	V123	ELA 2	1999 060A		GE 4	3900	GTO	CML	USA
	123	Dec 03	20	40-3	V124	ELA 2	1999 064A		Helios 1B	2544	SSO	MIL	Europe
							1999 064B	A	Clementine	50	SSO	MIL	France
	124	Dec 10	7	5G	V119 (504)	ELA 3	1999 066A		XMM	3764	EEO (40)	CIV	Europe
	125	Dec 22	12	44L-3	V125	ELA 2	1999 071A		Galaxy 11	4485	GTO	CML	USA
	126	2000 Jan 25	34	42L-3	V126	ELA 2	2000 002A		Galaxy 10R	3476	GTO	CML	USA
	127	Feb 18	24	44LP-3	V127	ELA 2	2000 012A		Superbird 4	4061	GTO	CML	Japan
	128	Mar 21	32	5G	V128 (505)	ELA 3	2000 016A	C	AsiaStar 1	2777	GTO	CML	China
							2000 016B	C	Insat 3B	2070	GTO	CML	India
	129	Apr 19	29	42L-3	V129	ELA 2	2000 020A		Galaxy 4R	3668	GTO	CML	USA
	130	Aug 17	120	44LP-3	V131	ELA 2	2000 046A	C	Brasilsat B4	1757	GTO	CML	Brazil
							2000 046B	C	Nilesat 102	1827	GTO	CML	Egypt
	131	Sep 06	20	44P-3	V132	ELA 2	2000 052A		Eutelsat W1	3250	GTO	CML	Europe
	132	Sep 14	8	5G	V130 (506)	ELA 3	2000 054A	C	Astra 2B	3315	GTO	CML	Luxembourg
							2000 054B	C	GE 7	1935	GTO	CML	USA
	133	Oct 06	22	42L-3	V133	ELA 2	2000 060A		NSAT 110	3531	GTO	CML	Japan
	134	Oct 29	23	44LP-3	V134	ELA 2	2000 068A		Europe*Star 1	4167	GTO	CML	France
	135	Nov 16	18	5G	V135 (507)	ELA 3	2000 072A		PAS 1R	4793	GTO	CML	USA
							2000 072B	A	STRV 1C	100	EEO (6.5)	MIL	UK
							2000 072C	A	STRV 1D	100	EEO (6.5)	MIL	UK
							2000 072D	A	AMSAT P3D (Oscar 40)	630	EEO (63)	NGO	International
	136	Nov 21	5	44LP-3	V136	ELA 2	2000 076A		Anik F-1	4711	GTO	CML	Canada
	137	Dec 20	29	5G	V138 (508)	ELA 3	2000 081A	C	Astra 2D	1414	GTO	CML	Luxembourg
							2000 081B	C	GE 8	2015	GTO	CML	USA
S							2000 081C	A	LDREX	182	EEO (2)	CIV	Japan
	138	2001 Jan 10	21	44P-3	V137	ELA 2	2001 002A		Eurasiasat 1	3535	GTO	CML	International
	139	Feb 07	28	44L-3	V139	ELA 2	2001 005A	C	Sicral 1	2596	GTO	MIL	France
							2001 005B	C	Skynet 4F	1489	GTO	MIL	UK
	140	Mar 08	29	5G	V140 (509)	ELA 3	2001 011A	C	EuroBird 1	3045	GTO	CML	France
							2001 011B	C	BSat 2A	1314	GTO	CML	Japan
	141	Jun 09	93	44L-3	V141	ELA 2	2001 024A		Intelsat 901	4723	GTO	CML	USA
F	**142**	**Jul 12**	**33**	**5G**	**V142 (510)**	**ELA 3**	2001 029A	C	Artemis	3105	EEO	CIV	Europe
							2001 029B	C	Bsat 2B	1314	EEO	CML	Japan
	143	Aug 30	49	44L-3	V143	ELA 2	2001 039A		Intelsat 902	4723	GTO	CML	USA
	144	Sep 25	26	44P-3	V144	ELA 2	2001 042A		Atlantic Bird 2	3055	GTO	CML	France
	145	Nov 27	63	44LP-3	V146	ELA 2	2001 052A		DirecTV 4S	4247	GTO	CML	USA
	146	2002 Jan 23	57	42L-3	V147	ELA 2	2002 002A		Insat 3C	2750	GTO	CML	India
	147	Feb 23	31	44L-3	V148	ELA 2	2002 007A		Intelsat 904	4672	GTO	CML	USA
	148	Mar 01	6	5G	V145 (511)	ELA 3	2002 009A		Envisat	8111	SSO	CIV	Europe
	149	Mar 29	28	44L-3	V149	ELA 2	2002 015A	C	Astra 3A	1495	GTO	CML	Luxembourg
							2002 015B	C	JCSat 8	2600	GTO	CML	Japan
	150	Apr 16	18	42P-3	V150	ELA 2	2002 019A		NSS 7	4702	GTO	CML	Netherlands
	151	May 04	18	42P-3	V151	ELA 2	2002 021A		SPOT 5	3030	SSO	CIV	France
							2002 021B	A	Idefix	6	SSO	NGO	France
	152	Jun 05	32	44L-3	V152	ELA 2	2002 027A		Intelsat 905	4723	GTO	CML	USA

ELA-1, -2, and -3 are at the Guiana Space Center near Kourou, French Guiana

T = Test Flight; F = Launch Vehicle Failure; P = Launch Vehicle Partial Failure; S = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	153	Jul 05	30	5G	V153 (512)	ELA 3	2002 035A	C	Stellat 5	4050	GTO	CML	France
							2002 035B	C	NStar C	1645	GTO	CML	Japan
	154	Aug 28	54	5G	V155 (513)	ELA 3	2002 040A	C	Atlantic Bird 1	2700	GTO	CML	France
							2002 040B	C	MSG 1	2000	GTO	CIV	Europe
	155	Sep 06	9	44L-3	V154	ELA 2	2002 041A		Intelsat 906	4723	GTO	CML	USA
F	**156**	**Dec 11**	**96**	**5ECA**	**V157 (517)**	**ELA 3**	2002 F03A	C	Hot Bird 7	3300	GTO	CML	France
							2002 F03B	C	Stentor	2210	GTO	MIL	France
	157	Dec 17	6	44L-3	V156	ELA 2	2002 057A		NSS 6	4575	GTO	CML	Netherlands
	158	2003 Feb 15	60	44L-3	V 159	ELA 2	2003 007A		Intelsat 907	4726	GTO	CML	USA
	159	Apr 09	53	5G	V160 (514G)	ELA 3	2003 013A	C	Galaxy 12	1760	GTO	CML	USA
							2003 013B	C	Insat 3A	2958	GTO	CML	India
	160	Jun 11	63	5G	V161 (515G)	ELA 3	2003 028A	C	BSat 2C	1275	GTO	CML	Japan
							2003 028B	C	Optus C1	4725	GTO	CML	Australia
	161	Sep 27	108	5G	V162 (516G)	ELA 3	2003 043B	C	e-Bird	1525	GTO	CML	France
							2003 043C	A	SMART 1	370	Moon	CIV	Europe
							2003 043E	C	Insat 3E	2750	GTO	CML	India

Failure Descriptions:

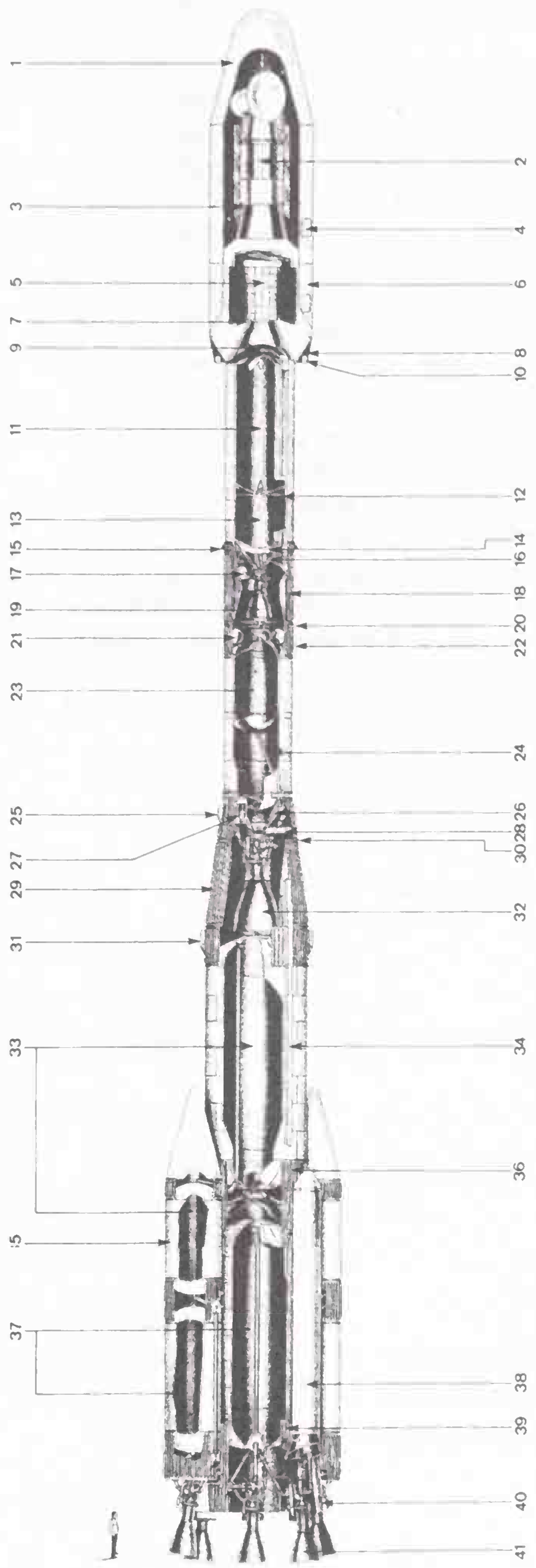
Event times are measured from main engine ignition at H+0, which occurs a few seconds before liftoff.

F	L02	1980 May 23	1980 F01	High-frequency combustion instability at H+4 and H+28 s degraded the injector of one first-stage engine. At H+64 s the chamber pressure dropped, reducing thrust, and causing the vehicle to begin to roll. This affected the propellant feed system of the other engines, causing them to shut down between H+104 and 108 s. The vehicle broke up, triggering autodestruct at H+108.
F	L5	1982 Sep 09	1982 F06	A failure of the third-stage turbopump gear caused pump speed to gradually drop, beginning at H+560 s until complete shutdown of the engine at H+610, halfway through the planned burn. The vehicle failed to reach orbit.
S	Oscar 10	1983 Jun 16	1983 058B	Spacecraft propulsive maneuver after deployment did not deliver the spacecraft to the planned Molniya orbit, possibly due to the collision of the spacecraft with the Ariane third stage after separation; amateur radio mission continued to operate.
F	V15	1985 Sep 12	1985 F03	Third stage engine failed to ignite properly and shut down at H+276 s, because of leaking hydrogen injector valve. The vehicle was destroyed by range safety officer.
F	V18	1986 May 31	1986 F05	The third-stage igniter failed, causing an explosive ignition that disrupted hydrogen flow and shutdown the third-stage engine.
S	TV-Sat 1	1987 Nov 21	1987 095A	Spacecraft solar array failed to deploy following launch.
S	GStar 3	1988 Sep 08	1988 081A	Spacecraft apogee kick motor burn did not result in correct GEO orbit due to imbalanced hydrazine loading on spacecraft. Final circularization into GEO performed by spacecraft thrusters.
S	Hiparcos	1989 Aug 08	1989 062B	Spacecraft stranded in GTO by failed apogee motor; astrometry mission plan reworked to complete most objectives.
F	V36	1990 Feb 22	1990 F01	A piece of cloth left in a water line during rework clogged the water coolant line, causing the gas-generator turbine in a first-stage main engine to overheat. The chamber pressure of the engine was reduced by half at T+6 s, causing a thrust imbalance. The flight control system gimbaled the engines to correct, but reached the 4.5 deg gimbal limit at H+90 s, after which angle of attack increased until the vehicle broke up because of aerodynamic forces at H+101 s.
S	Orbcomm X	1991 Jul 17	1991 050C	Contact with satellite lost shortly after deployment because of faulty power control software.
F	V63	1994 Jan 24	1994 F01	The third-stage oxygen turbopump bearing overheated beginning at H+408 s because of a combination of insufficient precooling and overloading. The engine shut down at H+428, 80 s after ignition, and the vehicle fell into the Atlantic.
S	Telstar 402	1994 Sep 09	1994 058A	Satellite propulsion system exploded after separation due to malfunctioning pyrovalve.
F	V70	1994 Dec 01	1994 F04	At third-stage ignition the engine achieved only 70% of normal thrust, caused by either a propellant impurity or ice clogging the oxygen line to the gas generator. The engine shut down 40 s early and the vehicle reentered.
F	V88	1996 Jun 04	1996 F03	A programming flaw carried over from Ariane 4 caused a stack overflow error in both of the redundant intertial reference systems simultaneously at H+36.7 s, causing the flight computer to erroneously command the engines to gimbal hard over. 2 s later the vehicle broke up only 4 km above the launch site, because of high aerodynamic loads caused by an angle of attack exceeding 20 deg.
P	V101	1997 Oct 30	1997 066	Exhaust swirl from the first-stage engine created a roll moment that caused the vehicle to spin at about 6 rpm after strap-on separation. This caused a vortex in the tanks, which resulted in propellant level sensors reading empty prematurely, triggering main engine shutdown 7 s early. The upper stage could not compensate, leaving the payloads with an apogee 9000 km too low.
S	LDREX	2000 Dec 20	2000 081C	The experimental antenna failed to deploy.
F	V142(510)	2001 Jul 12	2001 029	Overpressure during ignition of the EPS stage caused a brief interruption in propellant flow and high frequency instability, increasing heating of the chamber wall, which boiled the fuel flowing through the regenerative cooling passages. This caused a reduction in fuel flow and an increase in the flow of oxidizer. The reduction in fuel flow caused cooling channels to burn through, dumping fuel directly into the chamber, reducing specific impulse 20%. The engine shut down 90 s early because of oxidizer depletion, leaving the satellites with an apogee of only 17,528 km. BSAT-2b was a total loss, while Artemis used its electric propulsion system to achieve GEO more than 18 months later.
F	V157(517)	2002 Dec 11	2002 F03	Cracks in the cooling passages of the Vulcain 2 nozzle caused a loss of coolant first detected at H+96 s. The nozzle deformed from overheating causing asymmetric thrust leading to flight control difficulties starting around H+176 s and a complete loss of control when fairing separation occurred at H+187 s. Command destruct was triggered at H+455 s. Poor characterization of the flight loads experienced by the nozzle was considered a contributing factor.

VEHICLE DESIGN

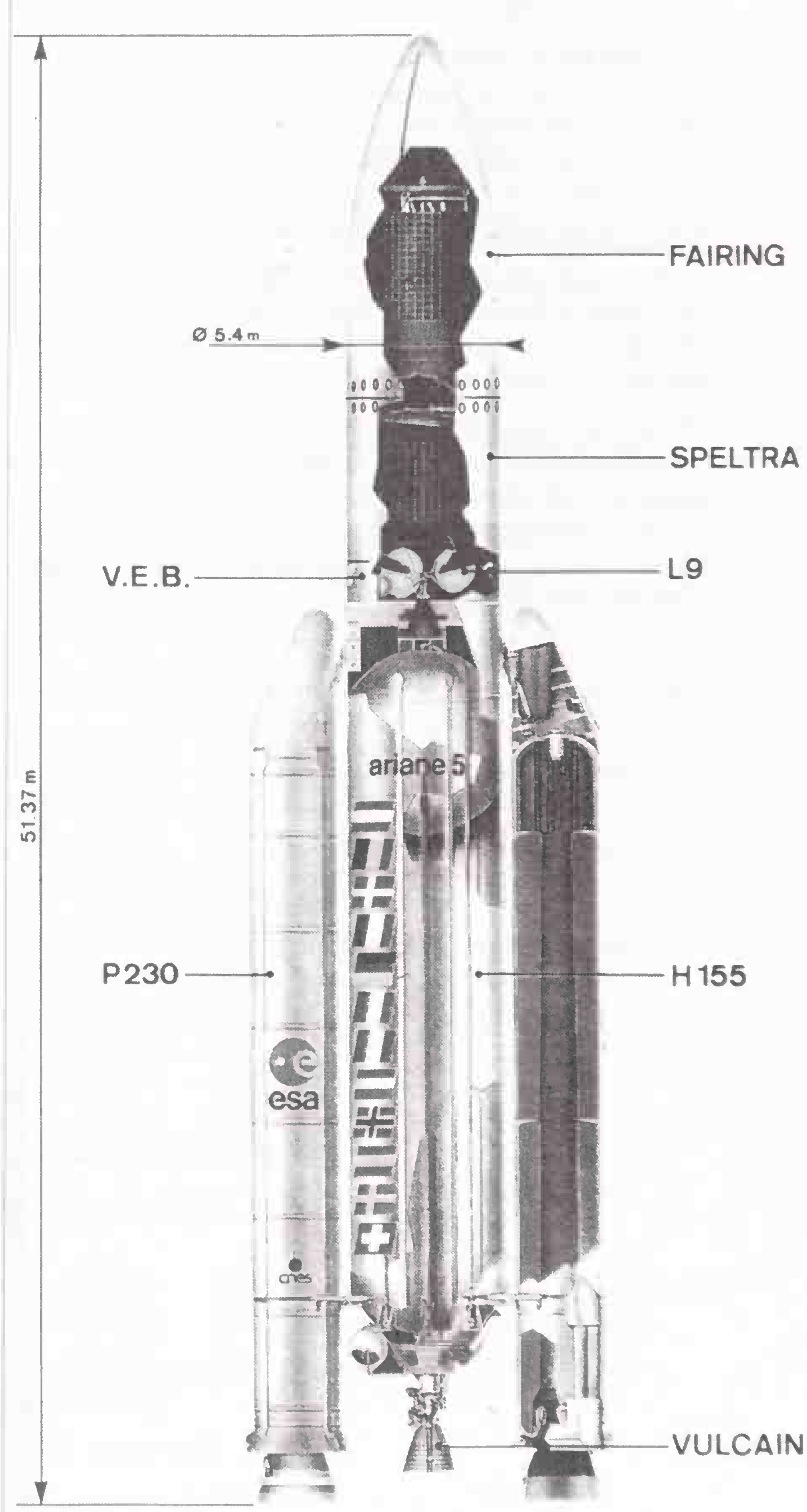
Overall Vehicle

Ariane 4



1 Fairing
2 Upper spacecraft
3 Upper spacecraft separation plane
4 Fairing door
5 Inner spacecraft
6 Spelda
7 Inner spacecraft separation plane
8 Vehicle Equipment Bay (VEB)
9 Third-stage sealing membrane
10 VEB equipment
11 Liquid-hydrogen tank
12 Anti-sloshing device
13 Liquid-oxygen tank
14 Roll and attitude control system
15 Third-stage acceleration rockets (2×2)
16 Third-stage thrust frame
17 Helium pressurization tank
18 Interstage 2–3
19 Third-stage engine (HM-7B)
20 2–3 separation plane
21 Second-stage helium pressurization tanks (3)
22 Second-stage front skirt
23 N_2O_4 tank
24 UH25 tank
25 Second-stage retro-rockets (2)
26 Second-stage thrust frame
27 Second-stage acceleration rockets (4)
28 Second-stage toroidal water tank
29 Interstage 1–2
30 1–2 separation plane
31 First-stage retro-rockets (4×2)
32 Second-stage engine (Viking IV)
33 N_2O_4 tank
34 External cable duct
35 Liquid strap-on boosters (2)
36 Intertank skirt
37 UH25 tank
38 Solid strap-on boosters (2)
39 First-stage thrust frame
40 Fairings
41 First-stage engines—Viking V (4)

Ariane 5



Height

AR4	58.7 m (192 ft)
AR5G	46.1–53.9 m (151–179 ft)
AR5E	46.1–53.9 m (151–179 ft)
AR5ECA	50.5 m (165.7 ft)

Gross Liftoff Mass

AR40	240 t (529 klbm)
AR42P	339 t (747 klbm)
AR44P	358 t (789 klbm)
AR42L	400 t (882 klbm)
AR44LP	420 t (925 klbm)
AR44L	470 t (1040 klbm)
AR5G	746 t (1645 klbm)
AR5ES	767 t (1690 klbm)
AR5ECA	780 t (1720 klbm)
AR5ECB	790 t (1740 klbm)

Thrust at Liftoff

AR40	2.71 MN (610 klbf)
AR42P	4.15 MN (930 klbf)
AR44P	5.59 MN (1260 klbf)
AR42L	4.04 MN (910 klbf)
AR44LP	5.27 MN (1180 klbf)
AR44L	5.38 MN (2560 klbf)
AR5G	11.4 MN (2560 klbf)
AR5E	13.0 MN (2900 klbf)

Vehicle Design

Ariane 4 Stages

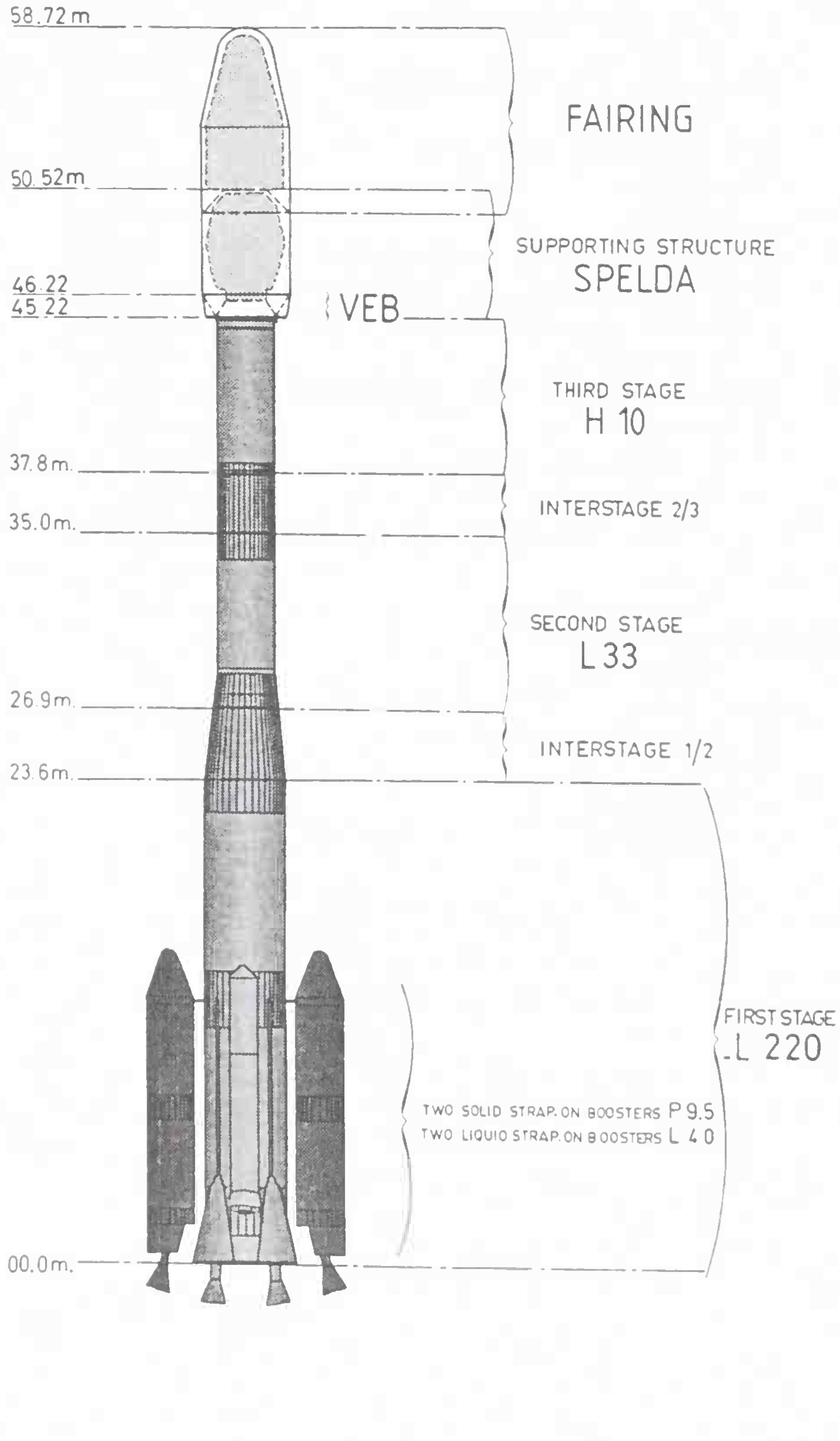
The first stage, designated L220, is powered by four Viking V engines, which burn UH25 as fuel and N_2O_4 oxidizer. UH25 is a mixture of 75% unsymmetrical dimethylhydrazine and 25% hydrazine hydrate. Each engine is an independent assembly supplied with propellant and coolant water by its own valves. Propellant is housed in two identical cylindrical steel propellant tanks connected by an intertank skirt of the same diameter. Inside the intertank skirt is the cylindrical steel water tank, containing a maximum of 6700 kg (14,700 lbm) of water. Propellant is off-loaded for the AR40 configuration. The engines are mounted in the cylindrical thrust frame, which connects to the lower end of the UH25 tank. Each Viking engine has a gas-generator turbine that drives the propellant and water pumps and also serves to pressurize the tanks. The engine throats and turbopump bearings are strengthened for longer burn time compared to earlier Ariane versions.

The second stage, known as L33, has one Viking IV engine. It carries propellants in aluminum alloy tanks, along with 560 kg (1230 lbm) of water. The tanks form a cylindrical structure with hemispherical bulkheads divided into two chambers by an intermediate bulkhead, also hemispherical, and with its concave face upward. Both tank compartments are pressurized by helium gas stored in spherical bottles at a pressure of 300 bar (4350 psi) at ambient temperature. During the waiting period on the launch pad before liftoff, the second-stage tanks are protected by a thermal shield, the temperature of which is controlled with cool air, to limit heat exchange between the fuels and the ambient environment. The shield is jettisoned at liftoff.

The third stage, designated H10, has one HM-7B engine. The two tanks holding the LOX and LH_2 propellants are made from an aluminum alloy and have a common intermediate bulkhead with a double vacuum-loaded skin. They are pressurized during flight by GH_2 for the hydrogen tank and by cold helium for the oxygen tank. Externally, the tanks are coated with thermal insulation to avoid rapid heating of the cryogenic propellants. The H10+ stage, introduced on flight V50 in 1992, was stretched 0.32 m (1 ft) to carry 340 kg more propellant. Structural changes reduced inert weight by 30 kg (66 lbm). The current H10-3 stage was introduced on V70 in 1994. The propellant margin was reduced, and the intermediate bulkhead was moved slightly to accomodate a slightly lower mixture ratio.

Ariane 4 can use a combination of liquid and/or solid strap-on boosters. The liquid strap-ons (called L40) consist of a Viking VI engine with two identical separate steel tanks, an intertank skirt, a forward skirt, and a nose cone. Water for the Viking engine is supplied from the first stage's central tank. After burnout, the liquid strap-ons are released by pyrotechnic cutting devices and jettisoned by small rockets. The solid strap-ons (called P9.5) are derived from the Ariane 3 solid boosters by increasing propellant mass 30%. Each solid strap-on is jettisoned after burnout by a system of four springs.

The Ariane 4 core stages are separated by explosive cords fitted into the rear skirts of the second and third stages. The stages are moved away from each other by retro-rockets mounted on the stage below the separation joint. Acceleration rockets fitted to the stage above the separation joint allow a small acceleration to be maintained to ensure steady propellant flow to the engine during ignition. Separation of the first and second stages is initiated by the onboard computer when the inertial guidance platform detects half-thrust decay in the first stage as a result of depletion of one of the propellants. Separation of the second and third stages is initiated by the onboard computer when the increase in velocity from second stage thrust has reached a predetermined value.



Ariane 4

Vehicle Design

	Solid Strap-On (P9.5 or PAP)	Liquid Strap-On (L40 or PAL)	Stage 1 (L220)	Stage 2 (L33)	Stage 3 (H10–3) + VEB
Dimensions					
Length	11.5 m (37.7 ft)	18.6 m (61.0 ft)	25.4 m (83.3 ft)	11.6 m (38.1 ft)	H10: 11.05 m (36.3 ft) VEB: 1.0 m (3.3 ft)
Diameter	1.07 m (3.51 ft)	2.17 m (7.12 ft)	3.8 m (12.5 ft)	2.6 m (8.5 ft)	H10: 2.6 m (8.5 ft) VEB: 4 m (13.1 ft)
Mass (each)					
Propellant Mass	9.5 t (20.9 klbm)	39.5 t (87.1 klbm)	228 t (503 klbm)	35.4 t (78.0 klbm)	11.7 t (25.8 klbm)
Inert Mass	3.1 t (6.8 klbm)	4.5 t (9.9 klbm)	17.9 t (39.5 klbm)	3.5 t (7.7 klbm)	1250 kg stage + 530 kg VEB (2750 + 1170 lbm)
Gross Mass	12.6 t (27.8 klbm)	44.0 t (97.0 klbm)	246 t (542 klbm)	38.9 t (85.7 klbm)	13.5 t (29.8 klbm)
Propellant Mass Fraction	0.75	0.90	0.93	0.91	0.87
Structure					
Type	Monocoque	Semi-monocoque	Semi-monocoque	Semi-monocoque	Semi-monocoque
Material	Steel	Tanks: stainless steel Skirts: aluminum Nose cone: aluminum	Tanks: stainless steel Skirts: aluminum	Aluminum	Aluminum
Propulsion					
Engine Designation	—	Viking VI (Snecma)	Viking VC (Snecma)	Viking IVB (Snecma)	HM7B (Snecma)
Number of Engines	0, 2, or 4 (1 segment each)	0, 2, or 4 PALs (1 engine each)	4	1	1
Propellant	CTPB	N_2O_4/UH25	N_2O_4/UH25	N_2O_4/UH25	LOX/LH_2
Average Thrust (per engine)	Sea level: 650 kN (145 klbf)	Sea level: 676.9 kN (152 klbf) Vacuum: 758.5 kN (171 klbf)	Sea level: 676.9 kN (152 klbf) Vacuum: 758.5 kN (171 klbf)	Vacuum: 785 kN (176 klbf)	Vacuum: 62.7 kN (14.1 klbf)
Isp	Sea level: 241 s	Sea level: 248 s Vacuum: 278 s	Sea level: 248.5 s Vacuum: 278.4 s	Vacuum: 293.5 s	Vacuum: 445.1 s
Chamber Pressure	?	58.5 bar (848 psi)	58.5 bar (848 psi)	58.5 bar (848 psi)	35 bar (508 psi)
Nozzle Expansion Ratio	?	10.48:1	10.48:1	30.8:1	62.5:1
Propellant Feed System	—	Gas-generator turbopump	Gas-generator turbopump	Gas-generator turbopump	Gas-generator turbopump
Mixture Ratio (O/F)	—	1.7:1	1.7:1	1.7:1	4.77:1
Throttling Capability	No	100% only	100% only	100% only	100% only
Restart Capability	No	No	No	No	No
Tank Pressurization	—	Pressurized He	Gas generator exhaust	Pressurized He	Fuel: GH_2, Oxidizer: He
Attitude Control					
Pitch, Yaw	Controlled by Stage 1, fixed 12-deg nozzle cant	Controlled by Stage 1, fixed 10-deg nozzle cant	Hydraulic gimbal of all 4 nozzles ±6 deg	Hydraulic nozzle gimbal ±3 deg	Hydraulic nozzle gimbal ±3 deg
Roll	Controlled by Stage 1	Controlled by Stage 1	Hydraulic gimbal of all 4 nozzles ±6 deg	2 hot gas thrusters	GH_2 thrusters
Staging					
Nominal Burn Time	42 s	140 s	205 s	126 s	745 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Command shutdown at predetermined velocity	Command shutdown at predetermined velocity
Stage Separation	4 springs	6 retro-rockets	8 retro-rockets	Retro-rockets	

Vehicle Design

Ariane 5 Stages

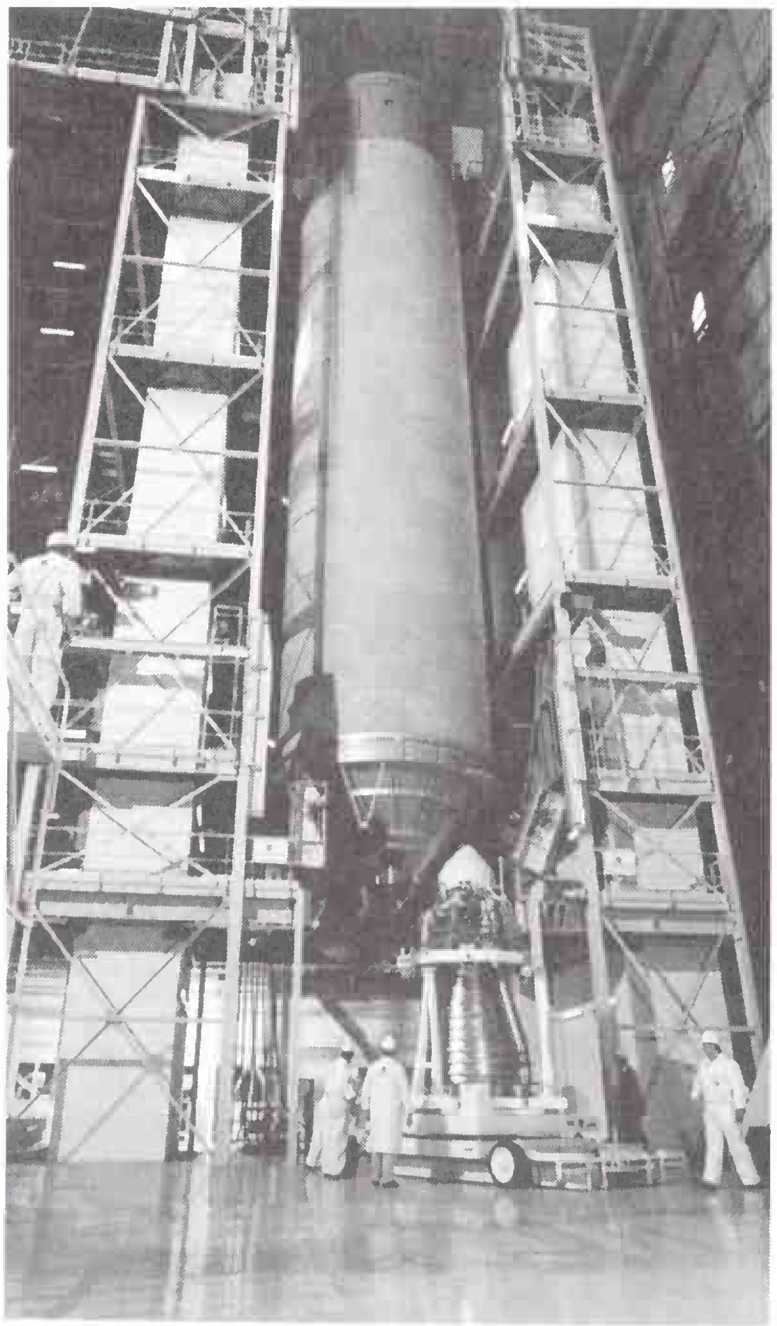
Most of Ariane 5's liftoff thrust is provided by two EAP solid boosters, which use large segmented solid motors designated MPS. The motors are a significant technical advance for ESA, being about 10 times larger than previous European solid motors. The motors are made of seven casing sections that are combined to make three propellant segments. The large central and lower segments are cast on site at the CSG Space Center, while the small forward igniter segment is cast in Italy. The three separate segments are integrated using two field joints during vehicle integration. The thrust profile of the boosters includes reduced thrust during the period of maximum dynamic pressure, so that the first stage engine does not need to be throttled. On occasional flights, a recovery system consisting of six parachutes is installed in the nose cone of each booster. The boosters are not intended to be reused, but recovery permits inspection of the motors after use. Several improvements are being made to the boosters for the Ariane 5E. An additional 2500 kg (5500 lbm) of propellant is added to the forward segment of each motor in a grain shape that primarily increases the thrust at liftoff. The nozzle design has also evolved, changing materials and manufacturing processes to reduce cost with weight reduction as a secondary benefit. In the future the case segments will be welded instead of bolted together, which will reduce weight and improve reliability. This improvement was originally planned for the 5E, but has not yet been implemented.

The EPC core stage is a cryogenic stage powered by the Vulcain main engine. The stage consists of two aluminum propellant tanks sharing a common hemispherical bulkhead. A thrust frame transmits the thrust of the main engine and solid boosters to the core stage structure. At the top of the stage, the forward skirt provides the interface to the second stage, and to the upper attachment points of the boosters. The forward skirt also includes most of the first-stage avionics.

The new Vulcain 2 engine is used on the Ariane 5E. The Vulcain 2 has a higher mixture ratio, 20% more thrust, and a larger nozzle. The LOX turbopump has been redesigned to accommodate the higher flow rate of LOX. The main combustion chamber has been redesigned with a wider throat and new injector. Whereas the Vulcain 1 dumped the exhaust from the gas generator overboard, the Vulcain 2 injects the exhaust gas into the nozzle at an area ratio of 32:1, using it to film-cool the lower extension of the nozzle. For this reason, hydrogen cooling tubes are only required above this point, and the lower nozzle section is a single wall with stiffening rings. Six or seven engines have been involved in the Vulcain 2 test program, with 53,000 s of test time accumulated by the first flight. The Vulcain 2 engine alone is responsible for 850 kg (1875 kg) of the performance improvement in the Ariane 5E.

The external dimensions and layout of the Ariane 5E EPC stage are unchanged. Modifications to the EPC stage are modest, and include lowering the common bulkhead to increase the amount of LOX and decrease the quantity of hydrogen. This allows more propellant to be carried in the same total tank volume, because LOX is much denser than LH_2. As a result of the increased thrust of the Vulcain 2, reinforcement of the thrust structure, engine gimbal system, some tank panels, and the front skirt was required. No changes to the feedlines were required, thanks to an increase in LOX turbopump suction and slightly higher tank pressure.

The EPS is a small, storable propellant upper stage that is used for orbital injection. It is a relatively simple design, using a pressure-fed, gimbaled Aestus engine. Propellant is stored in four tanks pressurized by helium bottles. While the Aestus engine itself is restartable, the EPS stage does not support engine restrats, because of limitations of thermal, power, and attitude control systems. Restart capability will be added for future flights by upgrading these systems. The EPS is encircled by the VEB. The VEB is a short, ring-shaped structure that supports the fairing at the top of its outer circumference, and the EPS on its inner circumference. The VEB contains the primary vehicle avionics as well as a hydrazine ACS system.



Courtesy Arianespace.

Ariane 5 Cryogenic Stage

Ariane 5

Vehicle Design

	EAP Booster		EPC Core Stage	
	P230 Ariane 5G	**P240 Ariane 5E**	**H158 Ariane 5G**	**H173 Ariane 5E**
Dimensions				
Length	31.6 m (103.7 ft)	31.6 m (103.7 ft)	30.5 m (100.0 ft)	30.5 m (100 ft)
Diameter	3.05 m (10.0 ft)	3.05 m (10.0 ft)	5.46 m (17.9 ft)	5.94 m (17.9 ft)
Mass (each)				
Propellant Mass	237.7 t (524.0 klbm)	240.2 t (530 klbm)	158.1 t (348.6 klbm)	172 t (378.4 klbm)
Inert Mass	39.8 t (87.7 klbm)	40.3 t (89 klbm)	12.2 t (26.7 klbm)	16.0 t (35.0 klbm)
Gross Mass	277.5 t (611.8 klbm)	280.5 t (618 klbm)	170.3 t (375.0 klbm)	188.3 t (414.6 klbm)
Propellant Mass Fraction	0.86	0.86	0.93	0.91
Structure				
Type	Monocoque	Monocoque	Semi-monocoque	Semi-monocoque
Material	Steel	Steel	Aluminum	Aluminum
Propulsion				
Engine Designation	MPS (Europropulsion)	MPS (Europropulsion)	Vulcain (Snecma)	Vulcain 2 (Snecma)
Number of Engines	2 (3 field segments each)	2 (3 field segments each)	1	1
Propellant	HTPB	HTPB	LOX/LH_2	LOX/LH_2
Thrust	Average: 4400 kN (990 klbf) Max: 6650 kN (1500 klbf)	Average: 5060 kN (1140 klbf) Max: 7080 kN (1590 klbf)	Sea level: 885 kN (199 klbf) Vacuum: 1140 kN (256 klbf)	Vacuum: 1350 kN (303 klbf)
Vacuum Isp	275.4 s	275.4 s	431.2 s	434 s
Chamber Pressure	61.3 bar (890 psi)	61.3 bar (890 psi)	108 bar (1566 psi)	115 bar (1670 psi)
Nozzle Expansion Ratio	10.36:1	10.36:1	45:1	58.5:1
Propellant Feed System	—	—	Open-cycle gas generator turbopump	Open-cycle gas generator turbopump
Mixture Ratio (O/F)	—	—	5.3:1	6.1:1
Throttling Capability	No	No	100% only	100% only
Restart Capability	No	No	No	No
Tank Pressurization	—	—	Fuel: Cold GH_2 Oxidizer: Hot He	Fuel: Cold GH_2 Oxidizer: Hot He
Attitude Control				
Pitch, Yaw	Hydraulic nozzle gimbal, ±6 deg	Hydraulic nozzle gimbal, ±6 deg	Blowdown hydraulic gimbal, ±6 deg	Blowdown hydraulic gimbal, ±6 deg
Roll	Hydraulic nozzle gimbal, ±6 deg	Hydraulic nozzle gimbal, ±6 deg	ACS on VEB	ACS on VEB or upper stage
Staging				
Nominal Burn Time	129 s	129 s	589 s	540 s
Shutdown Process	Burn to depletion	Burn to depletion	Command shutdown	Command shutdown
Stage Separation	8 separation rockets	8 separation rockets	Pyrotechnic tube and actuators	Pyrotechnic tube and actuators

Vehicle Design

Ariane 5 Upper Stages

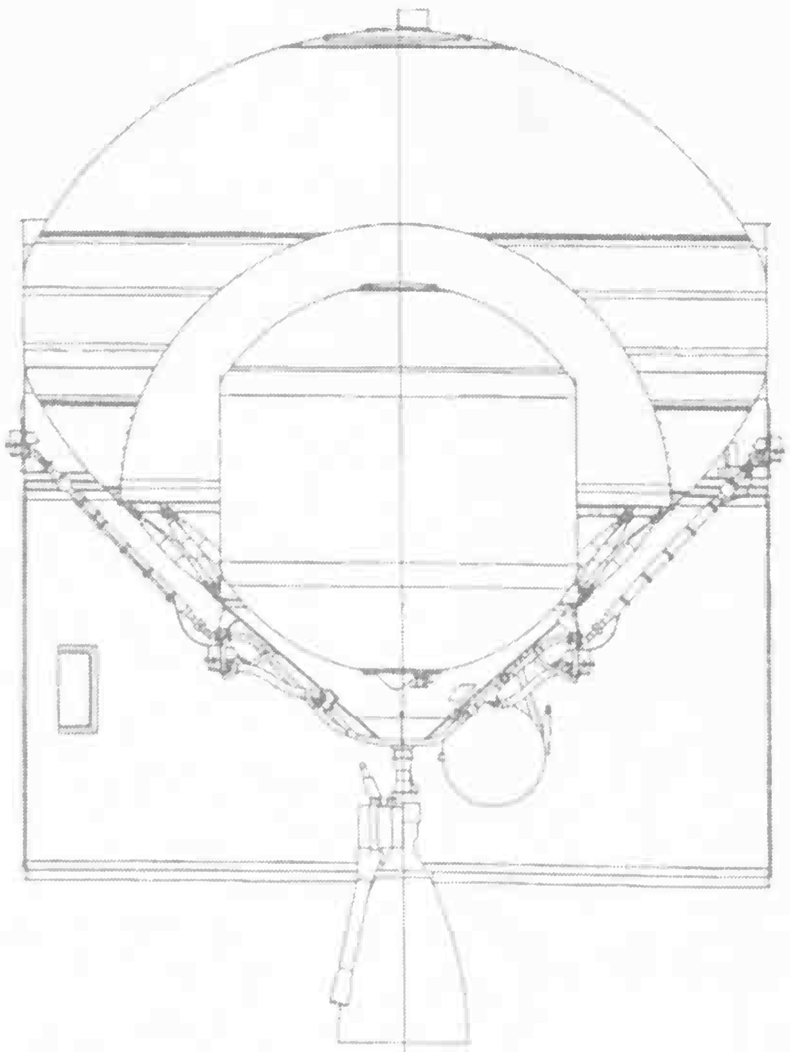
The EPS is a small, storable propellant upper stage that is used for orbital injection. It is a relatively simple design, using a pressure-fed, gimbaled Aestus engine. Propellant is stored in four tanks pressurized by helium bottles. While the Aestus engine itself is restartable, the original EPS stage did not support engine restarts, because of limitations of thermal, power, and attitude control systems. Restart capability will be added for the "Versatile" version of the stage by upgrading these systems. The stage has also been upgraded by slightly increasing the amount of fuel that can be carried, and reducing the structural weight.

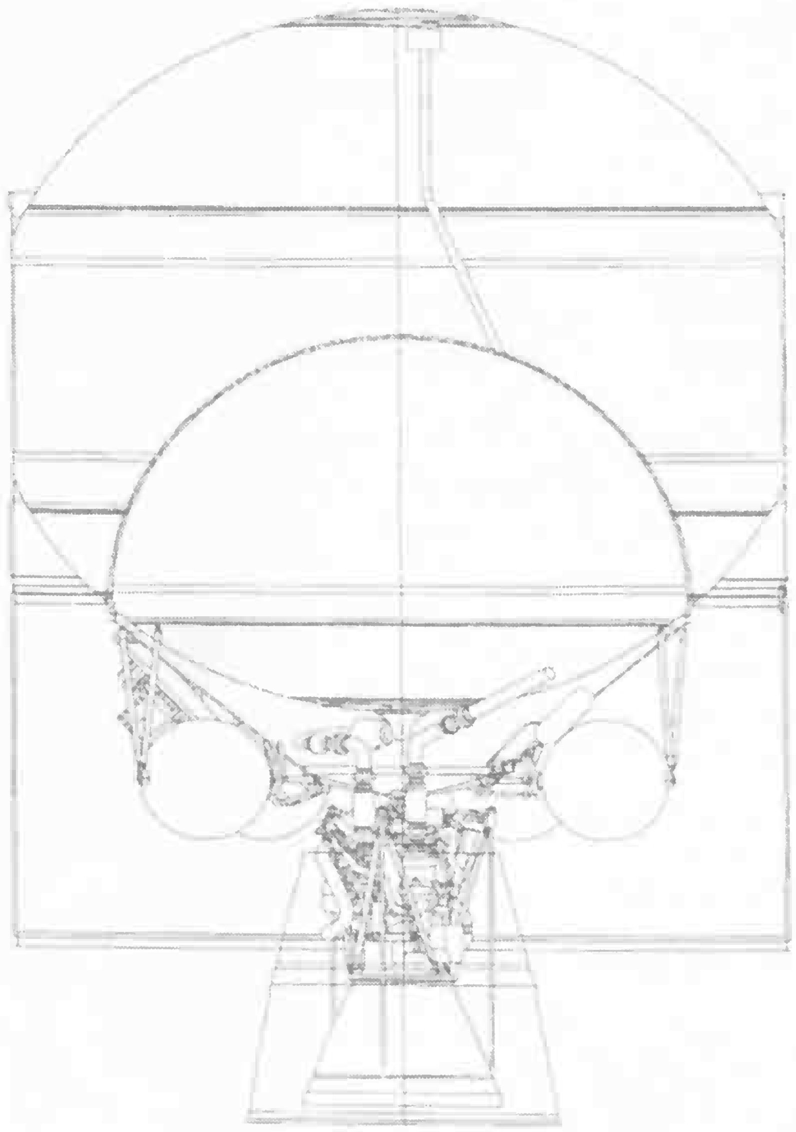
The EPS stage is encircled by the Vehicle Equipment Bay (VEB). The VEB is a short, ring-shaped structure that supports the fairing at the top of its outer circumference and the EPS on its inner circumference. The VEB contains the primary vehicle avionics as well as a hydrazine ACS. As part of a weight reduction program, the metal structure of the VEB has been replaced by carbon composite.

In response to growth in satellite mass, Arianespace and ESA chose to develop two new cryogenic upper stages for the Ariane 5E. The first is the ESC-A (Etage Supérieur Cryotechnique, cryogenic upper stage, Type A). To field a new cryogenic stage quickly, the ESC-A is derived from the successful Ariane 4 H10-3 stage, using the same HM-7B engine, a derivative 2.5-m diameter LOX tank, and a number of modified components, such as the thrust structure and the enlarged helium tank. While many elements could be reused, requalification was required to adjust to the longer burn duration, colder interstage environment, and different launch pad operations. A new 5.4-m diameter, 11.4-m^3 (403-ft^3) hydrogen tank was developed, using bulkheads similar to those on the EPC stage. The lower bulkhead is concave, allowing the LOX tank to nest within the hydrogen tank, though they do not share a common bulkhead. The two tanks are connected by 36 struts and 4 dampers. The upper stage is mounted on a new carbon composite interstage structure built by CASA.

Although the ESC-A could be developed quickly, it is considered only an intermediate step toward a more capable upper stage. The ESC-B will increase GTO performance from 10 t to 12 t, and also provide the capability for engine restarts. This requires the development of a new engine, which could not have been accomplished in time for the ESC-A. A key design decision during development was how to meet the needs of both GTO and so-called "versatile" missions, which can required different helium supplies, thermal insulation, and venting, depending on mission duration and the required number of engine starts. One option was to develop two slightly different versions of the stage—one optimized for GTO missions, the other for other missions. The second option was to design and build one flexible stage that could do both types of missions. The third option was to design the stage for GTO missions, with additional kits available to add capabilities as needed. The first option provides the best capability, but is the most expensive to develop. The second sacrifices some performance and is the most expensive to build. Considering the challenging cost and capability requirements and the prevalence of GTO missions, the third approach was selected.

The first ESC-B design, designated B0, had 23 t of propellant in two nested, but separate, propellant tanks. This design was reworked to achieve the performance and cost goals of the program. Keeping the propellant tanks separate would have simplified pressurization, but is heavier than common bulkhead tanks. A concept dubbed "stage under fairing" was considered, in which the entire stage would fit inside the payload fairing. This has weight advantages because the tanks do not need to support the aerodynamic loads of the fairing during flight, but loading propellant on the ground is difficult. The selected design, called B5-A H28, has 28 t of propellant in a common bulkhead tank, with the payload fairing above the stage. Essentially, there is a single propellant tank divided into two compartments. The hydrogen compartment is the same shape as the ESC-A tank, with a longer cylinder section. The oxygen is carried in a compartment formed by enclosing the lower concave cavity of the hydrogen tank. The engine for the ESC-B is the new Vinci engine developed by Snecma. It is an expander cycle engine capable of multiple restarts. For increased specific impulse, the engine has a large carbon–carbon nozzle extension skirt which is deployed in flight.





The ESC-A, left, and ESC-B, right, upper stages

Vehicle Design

Ariane 5 Upper Stages

A VEB is mounted above each stage. The Ariane 5G VEB has a height of 1.56 m (5.1 ft), diameter of 5.46 m (17.9 ft), and mass of 1500 kg (3300 lbm). For Ariane 5E an upgraded VEB is used with a lightweight composite structure and the ACS thrust package may be deleted in favor of using the thrusters on the ESC-A upper stage. This VEB has a mass of 950 kg (2100 lbm) and a height of 1.13 m (3.7 ft).

	EPS, L9	ESC-A	ESC-B
Dimensions			
Length	3.36 m (11.0 ft)	4.71 m (15.5 ft)	5.62 m (18.4 ft)
Diameter	3.96 m (13.0 ft)	5.43 m (17.8 ft)	5.43 m (17.8 ft)
Mass (each)			
Propellant Mass	5G: 9.7 t (21.4 klbm) 5E: 10 t (22 klbm)	2590 kg LH_2 + 11,950 kg LOX (5710 + 26,435 lbm)	4320 kg LH_2 + 23,922 kg LOX (9524 + 52,738 lbm)
Inert Mass	1200 kg (2600 lbm)	3418 kg (7535 lbm)	3908 kg (8616 lbm)
Gross Mass	5G: 10.9 t (24,000 lbm) 5E: 11.2 t (24,700 lbm)	17,958 kg (39,590 lbm)	32,150 kg (70,878 lbm)
Propellant Mass Fraction	0.89	0.81	0.88
Structure			
Type	Tanks: Monocoque Structures: Monocoque and honeycomb	Monocoque tanks, struts for intertank structure	Monocoque tanks Thrust structure skin-stiffened or sandwich
Material	Tanks: Aluminum Structures: Aluminum or aluminum–composite sandwich	Aluminum	Tanks: Aluminum Thrust structure to be either metallic or GFRP–Aluminum sandwich
Propulsion			
Engine Designation	Aestus (DaimlerChrysler)	HM7B (Snecma)	Vinci (Snecma)
Number of Engines	1	1	1
Propellant	MMH/N_2O_4	LH_2/ LOX	LH_2/ LOX
Average Thrust	Vacuum: 29 kN (6.5 klbf)	64.8 kN (14.6 klbf)	180 kN (40.5 klbf)
Isp	324 s	445.5 s	465 s
Chamber Pressure	10 bar (145 psi)	35 bar (508 psi)	60 bar (870 psi)
Nozzle Expansion Ratio	83:1	83:1	240:1
Propellant Feed System	Pressure fed	Gas-generator cycle turbopump	Expander cycle turbopump
Mixture Ratio (O/F)	2.05:1	4.86:1	5.8:1
Throttling Capability	100% only	100% only	100% only
Restart Capability	5G: No; 5E: yes	No	4 restarts
Tank Pressurization	He	Fuel: GH_2 Oxidizer: He with heat exchanger	He
Attitude Control			
Pitch, Yaw	Electromechanical nozzle gimbal ±9.5 deg	Nozzle gimbal, Cold-gas hydrogen thrusters	Nozzle gimbal during burn, cold-gas hydrogen thrusters during coast
Roll	VEB blowdown hydrazine ACS	Cold-gas hydrogen thrusters	Hot-gas hydrogen thrusters during burn, cold-gas during coast
Staging			
Nominal Burn Time	1100 s	970 s	?
Shutdown Process	Command shutdown	Command shutdown	Command shutdown
Stage Separation	—	4 acceleration rockets	?

Vehicle Design

Attitude Control System

Ariane 4

The four first-stage engine nozzles pivot in the plane tangent to the thrust frame, providing pitch, yaw, and roll control. In the AR42L and AR44L configurations, aerodynamic stability is improved by four fins attached to the thrust section, with an area of 2 m^2 (21 ft^2) each. The second stage Viking IV engine can be swiveled about two axes to allow yaw and pitch control. Roll control is provided by two tangential thrusters that use hot gas from the second stage gas generator to provide 50 N (11 lbf) of thrust. The third-stage engine, designated HM7B, is linked to the conical thrust frame through a gimbal joint, which allows swiveling of the engine for pitch and yaw attitude control. GH_2 thrusters provide roll control. These thrusters, together with additional hydrogen thrusters, provide three-axis attitude control of the stage and the attached payload after engine cutoff.

Ariane 5

The solid boosters are equipped with a blowdown hydraulic nozzle gimbal system (GAT), which enables pitch, yaw, and roll control. The core stage is equipped with a hydraulic engine activation system (GAM), which enables swiveling of the Vulcain engine for pitch and yaw control. The 180-liter (47-gal) hydraulic system is an open-cycle system pressurized by helium. An electromechanical gimbal on the Aestus engine provides pitch and yaw control during flight of the EPS second stage. After separation of the solid boosters, roll is controlled by a hydrazine attitude control system (SCA) installed in the VEB. This system also provides three-axis control during coast periods and deployment maneuvers. The SCA is a blowdown hydrazine system pressurized by nitrogen, with thrust levels decaying from 200 to 100 N (45 to 22 lbf). The cryogenic upper stages use nozzle gimbaling to control pitch and yaw during thrusting phases. Roll control during burns and three axis control during coast phases are provided using thrusters powered by hydrogen tapped from the main fuel tank.

Avionics

Ariane 4

Ariane 4 avionics are carried in the VEB. This configuration has two considerable advantages compared with Ariane 2 and 3. First, it allows separation of the equipment-carrying platform from the load-bearing structure, thus allowing independent integration of the payloads with the Spelda and the fairing in the clean rooms of the payload preparation buildings, after the equipment platform has already been installed on the launch vehicle. Second, it allows easy access to the launch vehicle equipment for checkout or any necessary intervention after the Spelda–fairing–payload cluster has been mounted on the launch vehicle. The VEB structure is fabricated in four parts from a honeycomb material with carbon-fiber facing.

The Ariane 4 guidance system originally consisted of two separate navigational units: a classical inertial platform (like that on Ariane 2 and 3) was the primary unit, with a laser gyro system as a backup. The inertial platform has been retired, and Ariane 4 now uses two redundant ring laser gyro systems similar to Ariane 5. A digital flight-control system on Ariane 4 replaced the analog control system used on Ariane 2 and 3. Other systems are practically unchanged, including the tracking system with two redundant radar transponders, the fully redundant destruct system that can receive a destruct command from the ground, and the telemetry system.

Ariane 5

Navigation data are provided by two redundant strap-down Inertial Reference Systems (IRS). Each IRS contains three ring laser gyros and four accelerometers. The IRS units are very similar to the Ariane 4 system to maintain reliability and reduce development cost. However, the failure of the maiden flight of Ariane 5 was caused by a data type conversion error in the IRS software, which occurred because the Ariane 5 trajectory profile causes some navigation parameters to exceed values experienced on Ariane 4. IRS data are provided to the onboard computer (OBC), which calculates guidance commands and event timing. The Ariane 5 VEB also contains the telemetry systems, redundant batteries and power distribution systems, and a flight termination system.

Payload Fairing

Ariane 4

The fairing is made of carbon-fiber reinforced plastics to keep its mass low. At separation, which is normally initiated at an altitude of about 110 km (60 nmi), the clamp band that holds the fairing to the vehicle is released. A pyrotechnic cord then cuts the fairing into two vertical halves and pushes them apart laterally—the fairing halves do not rotate on hinges. The Type 03 extra-long fairing is a nonstandard option available only by special request. For dual payload launches, the Ariane 4 Spelda supports the fairing, encloses the bottom spacecraft, and supports the upper spacecraft. Separation is achieved by cutting the structure along a horizontal plane and jettisoning the Spelda forward with springs. An egg-shaped Sylda support structure can be placed inside a fairing to encapsulate smaller payloads. Doors are available in both the Spelda and fairing.

Ariane 5

Ariane 5 is designed to launch two comanifested payloads per flight. This requires a significant amount of flexibility to ensure that the volumetric requirements of any combination of satellites can be met. Four different elements, each available in several lengths, can be combined to provide a large number of possible combinations.

The primary element, the payload fairing, has a 5.4-m (17.7-ft) outer diameter. It is available in short, medium and long lengths. The fairing structure is made up of sandwiched panels with an aluminum honeycomb core between CFRP skins. It is a two half-shell structure with a longitudinal separation system similar to that used on Ariane 4. The horizontal separation system is located in the lower flange and is based on a pyrotechnic cord. The long fairing is made of two parts, the upper one being similar to the short fairing, while the lower part is an extension. Acoustic absorbers are standard.

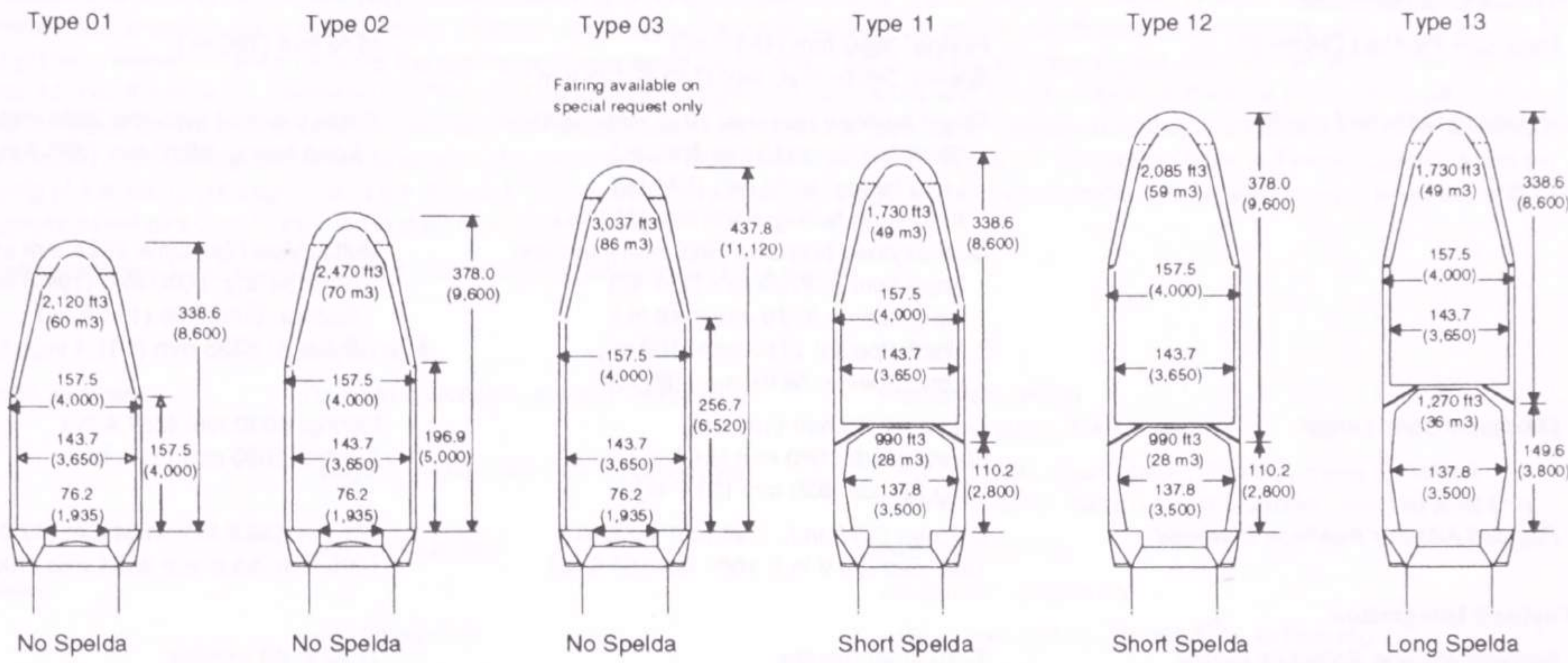
The Speltra is an external support structure providing an extra compartment for dual launches. It is made up of a 5.4-m (17.7-ft) diameter cylinder, which sits between the VEB and the fairing, and a conical section, which supports the upper payload. The lower payload is carried inside the Speltra. The Speltra is available in two versions, Speltra 4160 and Speltra 5660, indicating the length of the cylindrical compartment in millimeters. The structure is made of sandwiched panels with an aluminum honeycomb core between CFRP skins. The lower flange of the Speltra contains a pyrotechnic separation system. Doors are available in both the Speltra and fairing.

The Sylda 5 is also a dual-payload compartment, but unlike the Speltra it is a smaller 4.5-m (15.0-ft) diameter structure that fits inside the payload fairing. While it can not carry large diameter satellites internally, it weighs much less than a Speltra because it does not have to support fairing aerodynamic loads during ascent. The basic Sylda 5 can also be extended up to 1.5 m (4.9 ft) in 0.3-m (1-ft) increments.

A 5.4-m diameter spacer element, the ACY 5400, is available in lengths from 0.5 to 2.0 m (1.6–6.6 ft) in 0.5-m (1.6-ft) increments. It is simply a cylindrical ring spacer that can be used to vary the payload compartment height of any of the elements described above.

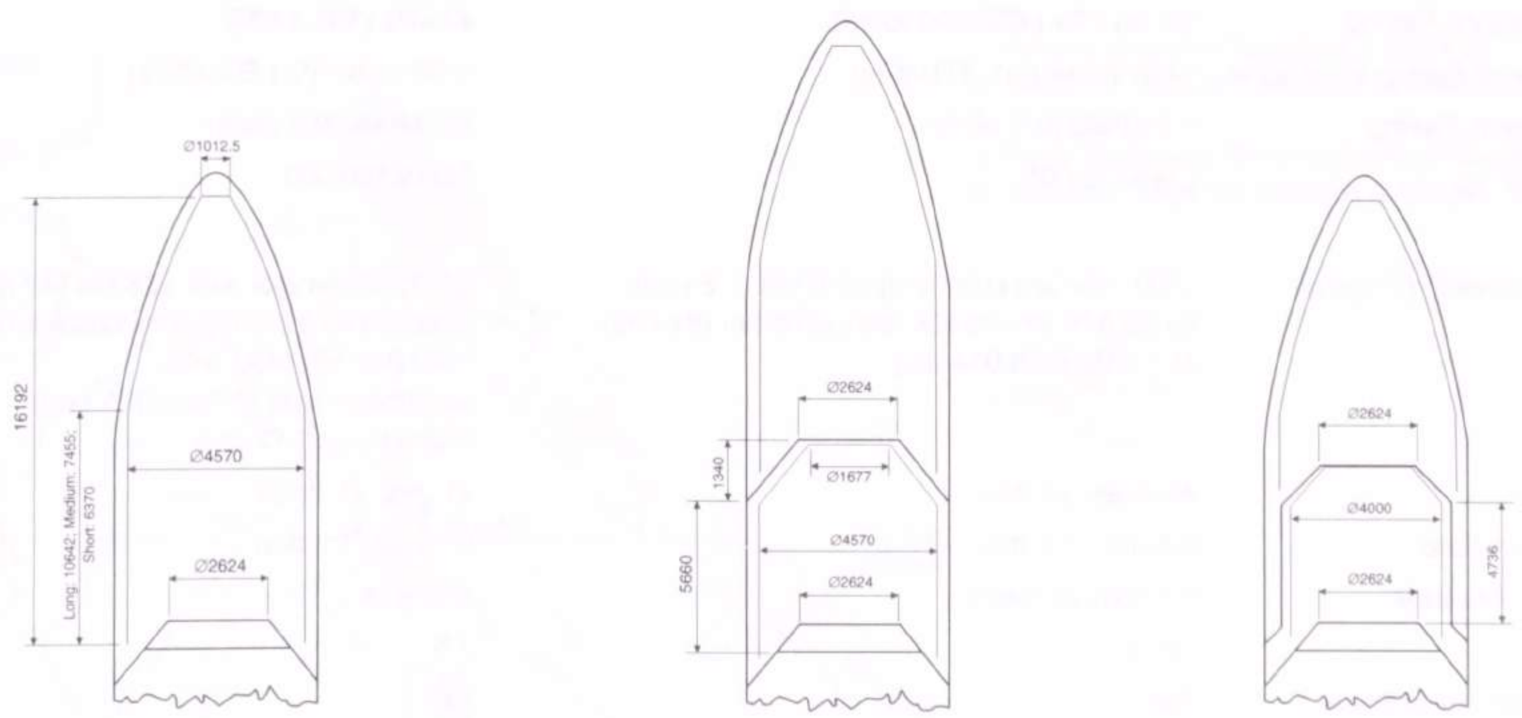
Vehicle Design

Ariane 4



	Type 01	Type 02	Type 03	Short Spelda	Long Spelda	Sylda
Length	8.6 m (28.2 ft)	9.6 m (31.6 ft)	11.1 m (36.4 ft)	2.8 m (9.2 ft)	3.8 m (12.5 ft)	4.4 m (14.4 ft)
Primary Diameter	4.0 m (13.1 ft)	4.0 m (13.1 ft)	4.0 m (13.1 ft)	4.0 m (13.1 ft)	4.0 m (13.1 ft)	2.8 m (9.2 ft)
Mass	760 kg (1675 lbm)	815 kg (1800 lbm)	890 kg (1960 lbm)	330 kg (730 lbm)	410 kg (900 lbm)	200 kg (440 lbm)
Sections	2	2	2	1	1	2
Structure	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich
Material	Graphite–epoxy/ aluminum honeycomb sandwich	Graphite–epoxy/ aluminum honeycomb sandwich	Graphite–epoxy/ aluminum honeycomb sandwich	Graphite–epoxy/ aluminum honeycomb sandwich	Graphite–epoxy/ aluminum honeycomb sandwich	Graphite–epoxy/ aluminum honeycomb sandwich

Ariane 5



	Short Fairing*	Long Fairing	SPELTRA	SYLDA 5	ACY 5400
Length	12.7 m (41.7 ft)	17.0 m (55.8 ft)	5.5 m (18.0 ft) or 7.0 m (23 ft)	4.9 m (16.1 ft)–6.4 m (21 ft) in 0.3-m (1-ft) increments	0.5 m (1.6 ft)–2 m (6.6 ft) in 0.3-m (1-ft) increments
Primary Diameter	5.4 m (17.7 ft)	5.4 m (17.7 ft)	5.4 m (17.7 ft)	4.56 m (15.0 ft)	5.4 m (17.7 ft)
Mass	2025 kg (4465 lbm)	2900 kg (6400 lbm)	716 kg (1578 lbm) or 822 kg (1812 lbm)	425–500 kg (940–1100 lbm)	134–250 kg (295–550 lbm)
Sections	2	2	1	1	1
Structure	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich
Material	Aluminum frames, graphite–epoxy/ aluminum honeycomb sandwich	Aluminum frames, graphite–epoxy/ aluminum honeycomb sandwich	Aluminum frames, graphite–epoxy/ aluminum honeycomb sandwich	Aluminum frames, graphite–epoxy/ aluminum honeycomb sandwich	Aluminum frames, graphite–epoxy/ aluminum honeycomb sandwich

* Medium fairing length is 13.8 m (41.7 ft)

Payload Accommodations

	Ariane 4	Ariane 5
Payload Compartment		
Maximum Payload Diameter	Fairing: 3650 mm (143.7 in.) Spelda: 3650–3190 mm (143.7–125.6 in.)	4570 mm (180 in.)
Maximum Cylinder Length	Single payload (assume 1920 mm interface) Short fairing: 3940 mm (155 in.) Long fairing: 4940 mm (194 in.) Extra-long fairing: 6460 mm (254 in.) Dual payload (assume 1920 mm interface) Short fairing: 2720 mm (107 in.) Long fairing: 3720 mm (146 in.) Short Spelda: 2760 mm (109 in.) Long Spelda: 3810 mm (150 in.)	Single payload (assume 2624 mm interface) Long fairing: 9822 mm (386.7 in.) Multipayload (assume 2624 mm interface) Short fairing: 5030 mm (198.0 in.) Speltra: 5000 mm (196.9 in.) Sylda 5: 3338 mm (131.4 in.)
Maximum Cone Length	Fairing: 4255 mm (168 in.) Short Spelda: 880 mm (34.6 in.) Long Spelda: 830 mm (32.7 in.)	Fairing: 6030 mm (237.4 in.) Speltra: 1050 mm (41.4 in.)
Payload Adapter Interface Diameter	937 mm (36.9 in.); 1194 mm (47.0 in.); 1497 mm (58.9 in.); 1666 mm (65.6 in.)	937 mm (36.9 in.); 1194 mm (47.0 in.); 1666 mm (65.6 in.); 2624 mm (105.3 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–10 to 40 months	T–10 to 40 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–6 min (launch postponed after T–5 sec requires minimum 10 days until next launch attempt)	T–5 min, 40 s
On Pad Storage Capability	Fueled stage 2: 2 days, Fueled stage 3: 5 h	60 days after fueling EPS
Last Access to Payload	T–2 days through access doors	T–2 days through access doors
Environment		
Maximum Axial Load	+4.5 *g*	+4.25 *g*
Maximum Lateral Load	±0.2 *g*	±0.25 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz/31 Hz	9 Hz/payload dependent
Maximum Acoustic Level	139 dB at 2000–4000 Hz	139 dB at 2000–4500 Hz
Overall Sound Pressure Level	142 dB (one third octave)	142 dB (one third octave)
Maximum Flight Shock	2000 *g* at 1500–4000 Hz at lower adapter interface	5000 *g* at 2–10 kHz on spacecraft interface
Maximum Dynamic Pressure on Fairing	20–30 kPa (420–630 lbf/ft^2)	40 kPa (835 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	3.2 kPa/s (0.5 psi/s)	3.2 kPa/s (0.5 psi/s)
Cleanliness Level in Fairing	Class 10,000	Class 100,000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	GTO: 200 km (108 nmi) ±3.0 km (1.6 nmi) by 35,975 km (19424 nmi) ±156 km (84 nmi) at 7.0 deg ± 0.054 deg	GTO: Semimajor axis ±78 km (42 nmi); Eccentricity ±0.0009; Inclination ± 0.06 deg SSO 800 km (430 nmi): Semimajor axis ±12 km (6.5 nmi) Inclination ±0.12 deg
Attitude Accuracy (99%)	All axes: ±3 deg	±1 deg; ±1 deg/s
Nominal Payload Separation Rate	0.5 m/s (1.6 ft/s)	0.5 m/s (1.6 ft/s)
Deployment Rotation Rate Available	0-5 rpm ±2 deg/s	0–5 rpm
Loiter Duration in Orbit	400 s	? h
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes
Multiple/Auxiliary Payloads		
Dual or Multiple Manifest	Ariane 4 frequently carries dual payloads for separate customers.	Ariane 5 is designed to carry up to three independent primary payloads or multiple satellites for constellations.
Auxiliary Payloads	Several microsatellites can be carried on the Ariane Structure for Auxiliary Payloads (ASAP), which circles the payload adapter. Each must fit in the volume of a cube 450 mm (17.7 in.) on each side and weigh no more than 50 kg (110 lbm) with a combined mass of 200 kg (440 lbm). Generally, auxiliary payloads are only carried on occasional LEO missions, as GTO payloads use all available performance. Launch price is typically < $1 million.	The ASAP5 structure carries auxiliary payloads on the Ariane 5 adapter. Up to eight 100-kg microsatellites or four 300-kg minisatellites, or a combination thereof, can be carried. Microsatellites must be no more than 600×600 mm square and 800 mm high (23.6×23.6×31.5 in.). Minisatellites must fit in the volume of a cylinder 1500 mm high and 1500 mm in diameter (59×59 in.). Auxiliary payloads may be carried on missions to either LEO or GTO.

Production and Launch Operations

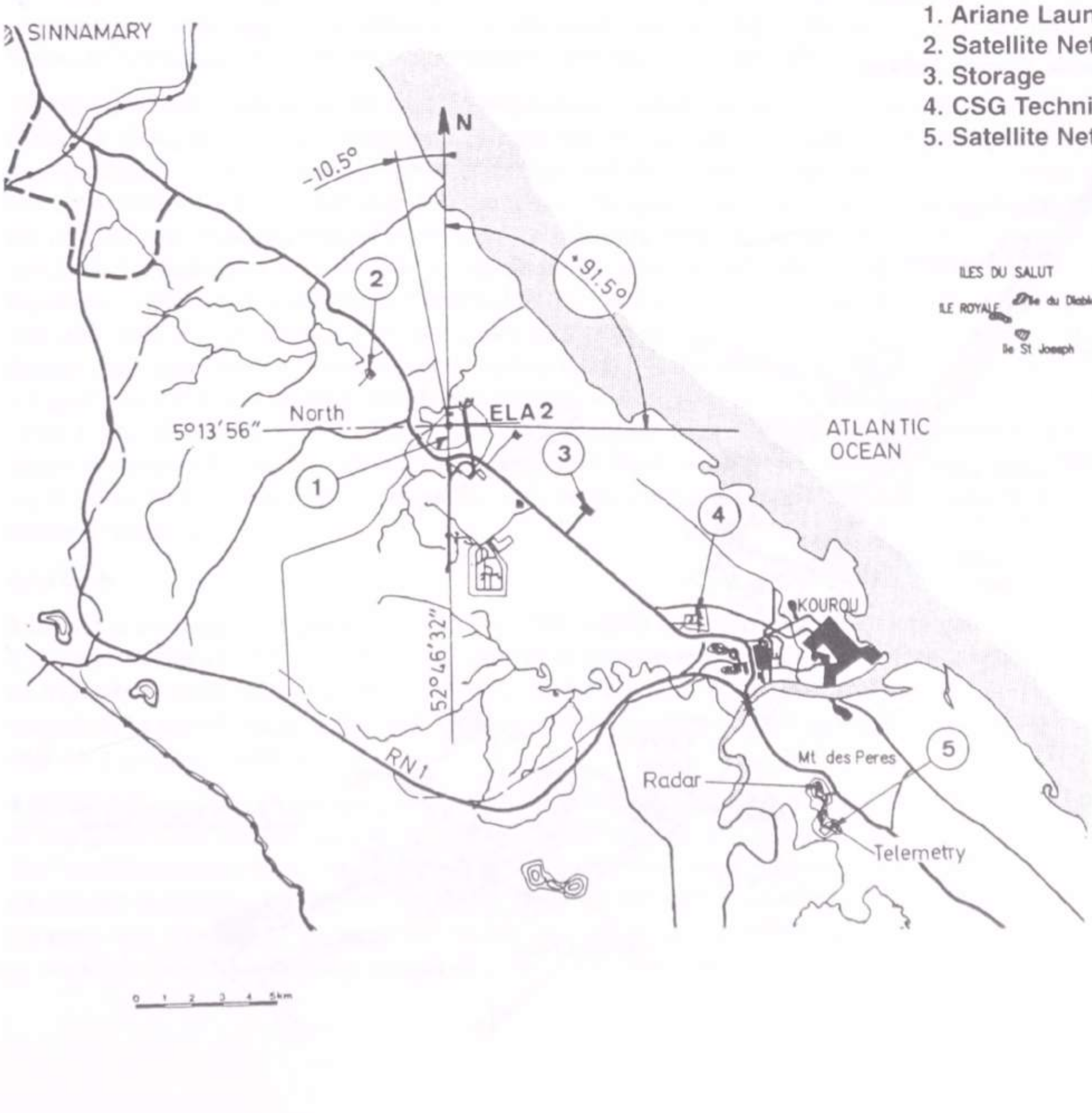
Production

ESA rules dictate that work on the Ariane program be divided up among member nations in proportion to the amount each country contributed to the vehicle development program. As a result, Ariane vehicles are produced for Arianespace by roughly 50 European companies, with responsibility for stages and primary elements spread among the members' major aerospace firms. The countries with the largest work shares are France (50.6% of Ariane 4 and 42.1% of Ariane 5), Germany (22.7% and 21.5% respectively), and Italy (6.0% and 14.1%). Much of the final assembly is performed at EADS's Les Mureaux facility outside of Paris (formerly an Aerospatiale factory), and at Astrium in Bremen (formerly a DASA facility). The assembled stages are shipped from Europe to French Guiana in the cargo vessel *MN Toucan*. The only significant production done in French Guiana is the propellant casting of the two large segments of the Ariane 5 solid booster, which are too large to ship from Europe. The smaller forward segment with the igniter is manufactured and cast in Italy at FiatAvio's Colleferro facility.

The following companies have significant production responsibilities for Ariane vehicles.

Contractors	Ariane 4	Ariane 5
EADS Launch Vehicles	industrial architect, stage 1, stage 3	industrial architect EPC core stage, EAP boosters,
Astrium	stage 2, liquid-strap-ons, VEB, Spelda	EPS and ESC upper stages, Speltra, Sylda 5, Aestus engine for EPS, Vulcain thrust chamber, payload adapters
CASA	payload adapters	payload adapters
Europropulsion		solid motor propulsion
FiatAvio	solid strap-ons	solid rocket motors, Vulcain LOX turbopump
MAN Technologie	Viking turbopumps	solid booster cases, EPS tanks
Contraves Space	payload fairing	payload fairing
Regulus		solid booster propellant
SAAB Ericsson		payload adapters
SFENA	ring laser gyros	ring laser gyros
Snecma	Viking engingеs for stage 1, 2, and HM7B engine for stage 3	Vulcain 1 and 2 engines, HM7B for ESC-A
Volvo Aero		Vulcain 2 nozzle

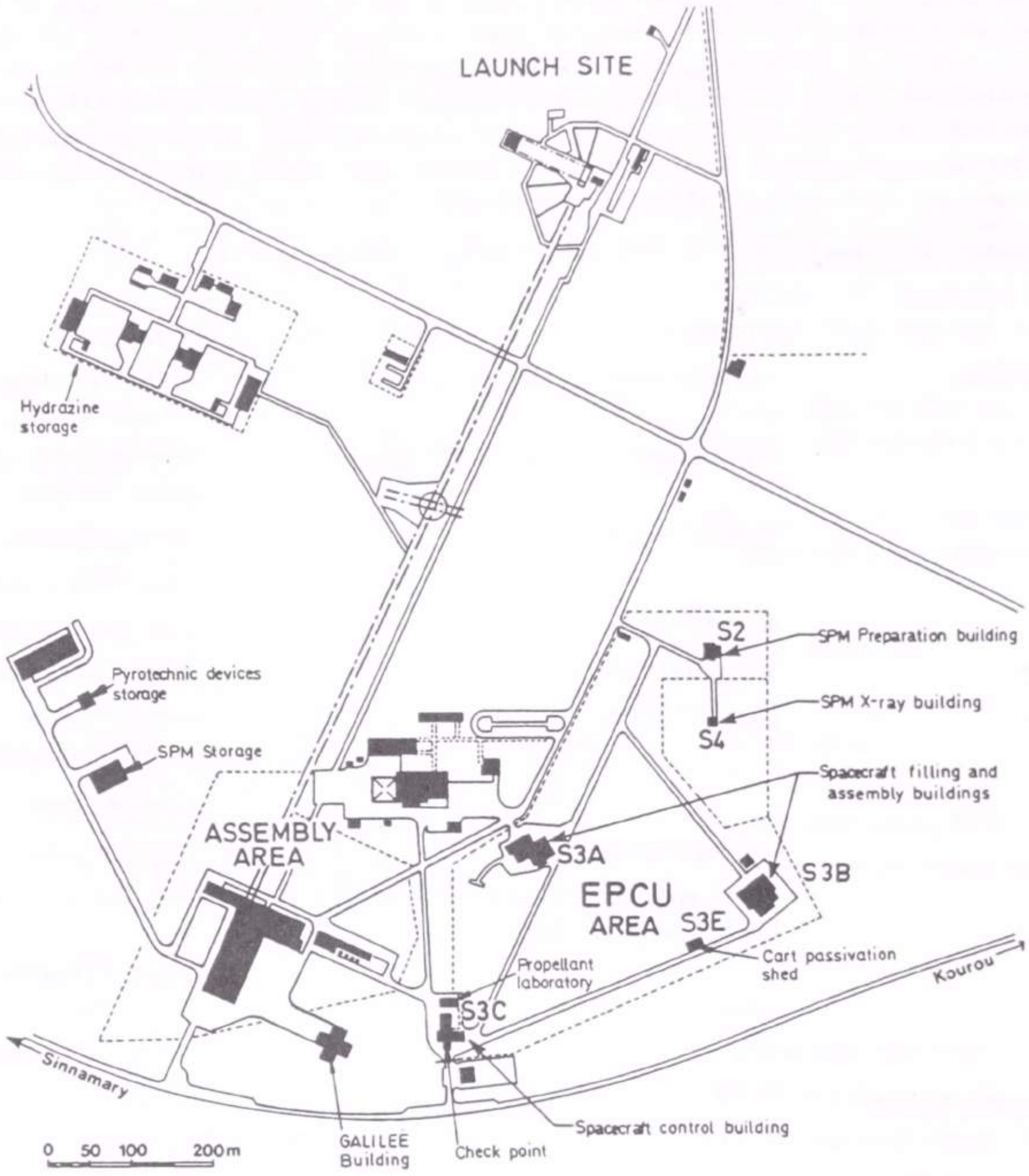
Launch Operations



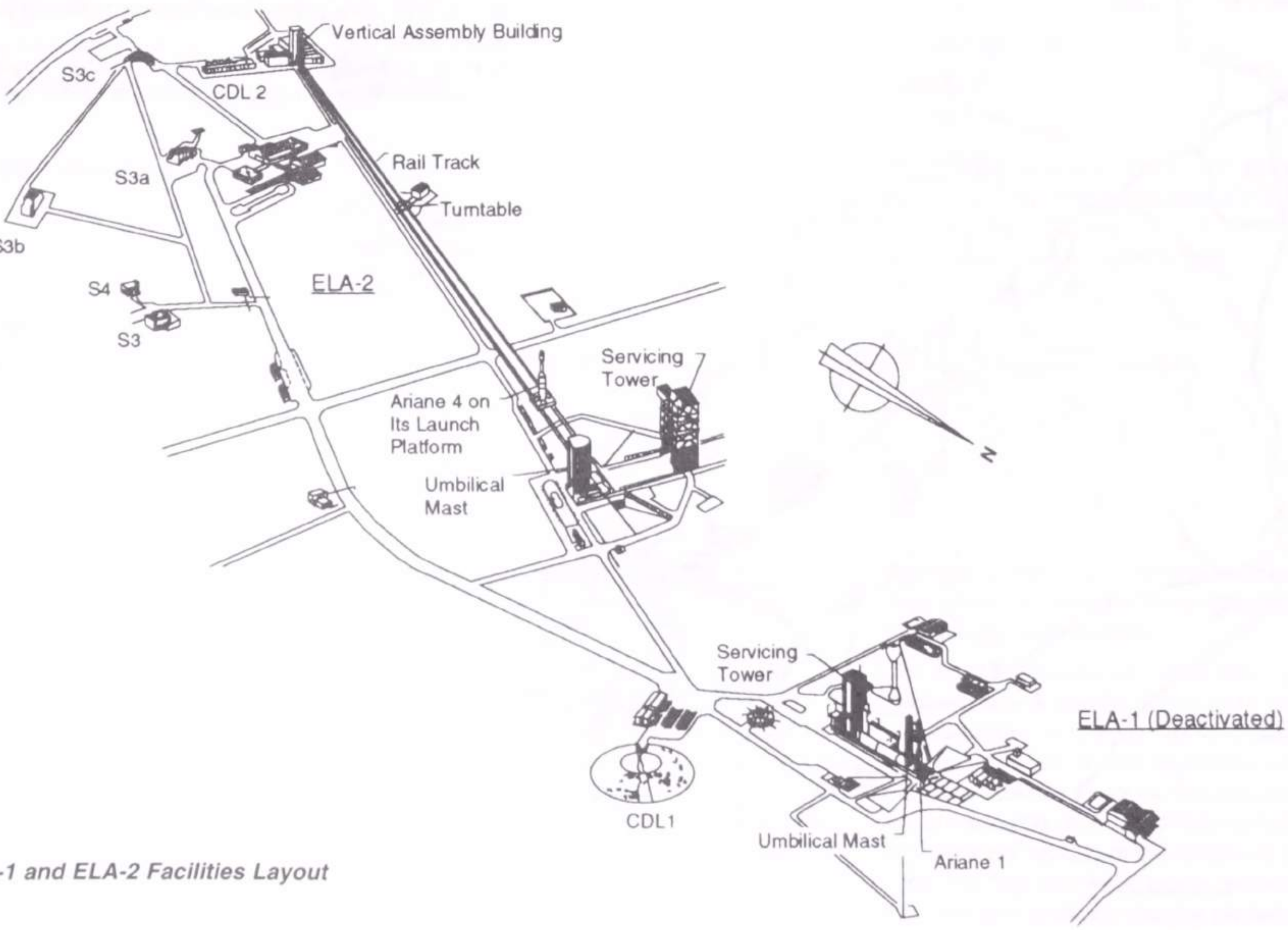
Key:
1. Ariane Launch Complex 2 (ELA 2)
2. Satellite Network Control Station (KRU93) S-band
3. Storage
4. CSG Technical Center (Control Center, Payload Processing)
5. Satellite Network Station (KRU 98, KRU 100)–S band

Production and Launch Operations

Launch Facilities

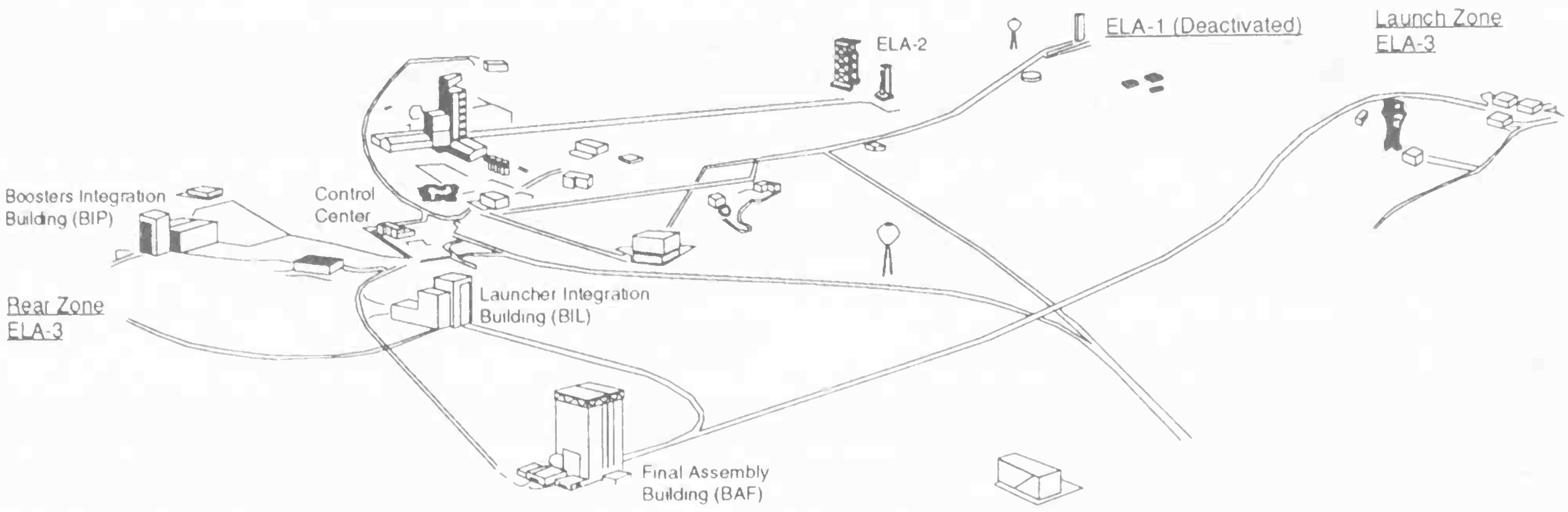


Ariane Launch Complex (ELA-2)



ELA-1 and ELA-2 Facilities Layout

Production and Launch Operations



ELA-3 Facilities Layout

Launch Operations—Guiana Space Center

Ariane launches are conducted from the ELA (Ensemble de Lancement Ariane) launch sites at the Guiana Space Center (Centre Spatial Guyanais, CSG) near Kourou, French Guiana, in South America. The space center includes all facilities used for a launch campaign sequence, including Ariane launch complexes, payload preparation complex, and downrange stations.

The CSG facilities are spread over a 18-km (11-mi) strip of the Atlantic coast, between Kourou and Sinnamary. The CSG is located close to the equator (latitude 5.23° N), which makes it well located for launching satellites into GEO. In addition, a broad launch sector over the Atlantic Ocean extends from north to east (−10.5 to +91.5 deg launch azimuth), making CSG one of the few launch sites in the world that can efficiently launch both GTO and polar satellites. The CSG is responsible not only for supplying the overall logistic support during launch activities but also operating the tracking and telemetry networks (including the processing of all launch vehicle telemetry data) and for the safety of personnel and the protection of the facilities. Owned by ESA, the launch sites are operated by Arianespace, which also maintains the launch facilities. The fixed costs of operating CSG are paid by European governments. The budget allotment for the five year period 2002–2006 was 617 million, ($606 million), or about $120 million per year.

ELA-1 was originally used for launching Europa 2, and later for launching Ariane 1, 2, and 3 before being deactivated. ELA-2 was used for Ariane 3 and currently is used for Ariane 4. ELA-3 was constructed for Ariane 5. For the eastward launches, the CSG radar, telemetry, and telecommand stations are supported by three downrange stations to continuously receive data on the launch vehicle's trajectory and behavior in flight. These are located at Natal, Brazil, on Ascension Island in the South Atlantic, and near Libreville, Gabon.

Payload processing occurs at the payload preparation complex (EPCU), which is owned by ESA and operated by CSG for Arianespace customers. The EPCU is designed for the preparation of satellites in a single- or dual-launch configuration, and includes all facilities placed at the disposal of Arianespace customers for the preparation of their satellites, from their arrival in French Guiana up to the actual mounting of the payload on the Ariane launch vehicle. The EPCU consists of a number of geographically dispersed buildings. Buildings S1A and S1B are located in the CSG technical center, and provide clean-room facilities for satellite receiving, final integration, and testing of electrical, mechanical, optical, and pneumatic systems. Solid kick motor preparation is performed in buildings S2 and S4, near the ELAs. These facilities are designed for x-ray inspection, and installation of pyrotechnic devices on the motors. Satellite fueling operations are conducted in building S3A and S3B near the two ELAs. Building S3C is located near S3A and S3B, and is used for monitoring and control of hazardous operations. For Ariane 4, final integration, assembly of the satellites on a Sylda or Spelda dual launch system, and satellite encapsulation into the payload fairing are performed in buildings S3A and S3B. For Ariane 5, these operations are performed in the Final Assembly Building (BAF) at ELA-3. In 2001 Arianespace opened the new S5 facility, built at a cost of 75 million francs ($11.3 million). The S5 facility has two preparation halls designed to handle the new generation of large satellites, including not only large GEO satellites but also even larger LEO spacecraft such as Envisat and the ATV, which will weigh up to 20 t (44,000 lbm). All satellite processing tasks, including the loading of up to 10 t (22,000 lbm) of propellant, can be accomplished in the S5 building avoiding the need to move the spacecraft to multiple facilities for different functions.

Ariane 4

Ariane 4 is prepared and launched at ELA-2. This facility allows the minimum interval between two launches to be reduced to less than three weeks, compared to two months on ELA-1, thus providing considerably greater flexibility in launch scheduling. While the concept of the ELA-1 complex called for the vehicle erection and assembly directly on the pad, the ELA-2 vehicle assembly and checkout takes place in a remote VAB. The vehicle is then moved on its launch table to the launchpad. While this vehicle is undergoing final preparation on the pad, the next one can already be erected in the VAB on a second mobile launch table.

A typical Ariane 4 launch campaign starts about nine weeks before the launch, with the transport of all the vehicle hardware in special containers, first on the Seine from Les Mureaux to Le Havre and then by ship to Cayenne, French Guiana. Propellants (except LOX, which is produced in Kourou) are also loaded onboard at Le Havre. Some 10 days later the vessel arrives at the port of Cayenne, from where the launch vehicle and propellants are transferred by truck to the launch site, some 15 km (9 mi) west of Kourou. In the VAB, the launch vehicle is erected on the mobile launch table and liquid strap-ons, if required, are attached. Mating and checkout in the VAB take three to four weeks. The vehicle, on one of its two mobile tables, is pulled by truck for about 50 min along a rail track 1 km (0.6 mi) long to the pad. At the pad, solid strap-ons, if required, are attached.

Production and Launch Operations

In parallel, the payload, which has been flown to Cayenne, is prepared at the EPCU. At building S3A/B, payloads are encapsulated in the payload fairing, making a transport container superfluous and reducing the time needed for payload preparation in the launch tower. The fully integrated upper component consisting of the VEB structure, the Spelda, if needed, the payload(s), and the fairing is transported to the launch pad and installed on top of the vehicle five working days before launch.

The launch countdown, covering mainly the filling of the stages with propellants, takes about 38 h distributed over three days. At about 10 and 5 h before launch, balloons are released for winds-aloft measurement. The keepout zone for lightning, which occurs mostly during July and August, is 8 km (5 mi). During the last 6 min before launch initiation, the ground checkout system verifies the proper functioning of the vehicle. It also separates the propellant transfer arms from the third stage 5 s before the end of the sequence, and finally commands ignition of the four first-stage engines and of the liquid-propellant boosters.

After ignition, the ground checkout system monitors the functional parameters of the ignited engines. If their status is correct, the command to open the jaws holding the launcher is given at the same time as that to ignite the solid propellant boosters. The typical flight time for the Ariane 4 launch vehicle, up to third-stage engine shutdown, is 17–18 min. The ELA-2 launch complex is capable of supporting 10–12 launches per year.

Ariane 5

Ariane 5 is launched from the ELA-3 complex near ELA-2. The ELA-3 launch site was required to maintain a high launch rate of 10 Ariane 5 vehicles per year, while reducing costs for investment, operations, and maintenance. To fulfill these requirements, ELA-3 was designed for a launch process different from that used at ELA-2. Preparation and checkout of the launch vehicle components are done in parallel, in separate buildings designed for each task, thus reducing the total time needed for the launch campaign. The payload is checked and encapsulated in the fairing before being installed on the launch vehicle. Final vehicle assembly, checkout, and payload integration are done in a fixed building, after which the launch vehicle is transported to a simple pad on a mobile launch table. Only cryogenic propellant loading and the launch take place at the pad, eliminating the need for complex mobile service structures, and reducing the amount of infrastructure that must be replaced in the event of an accident. The parallel operations allow the typical launch campaign to take only three weeks.

ELA-3 is divided into two zones, the preparation zone and the launch zone. The preparation zone consists of four main buildings: Booster Integration Building (BIP), Launcher Integration Building (BIL), Final Assembly Building (BAF), and Preparation and Launch Control Center (CDL 3). These buildings are widely separated because of safety constraints imposed by the use of solid-propellant boosters.

The BIP is where the solid boosters are assembled and checked out. The center and aft booster segments are assembled and cast onsite in a plant located near the BIP, because it would be difficult to safely transport such large motors from Europe. The smaller forward segment is produced in Italy. In the BIP, the segments are mated in two integration rooms on two special devices that are used to transfer the booster. The boosters are fully prepared and checked before leaving.

In the BIL, the EPC cryogenic stage, shipped from Europe, is erected and mated on the launch table. The VEB and EPS stage are integrated onto the core stage. Both solid boosters are then mounted on the launch table. Electrical, mechanical, and fluid systems are checked, and the vehicle is prepared for mating with the payload.

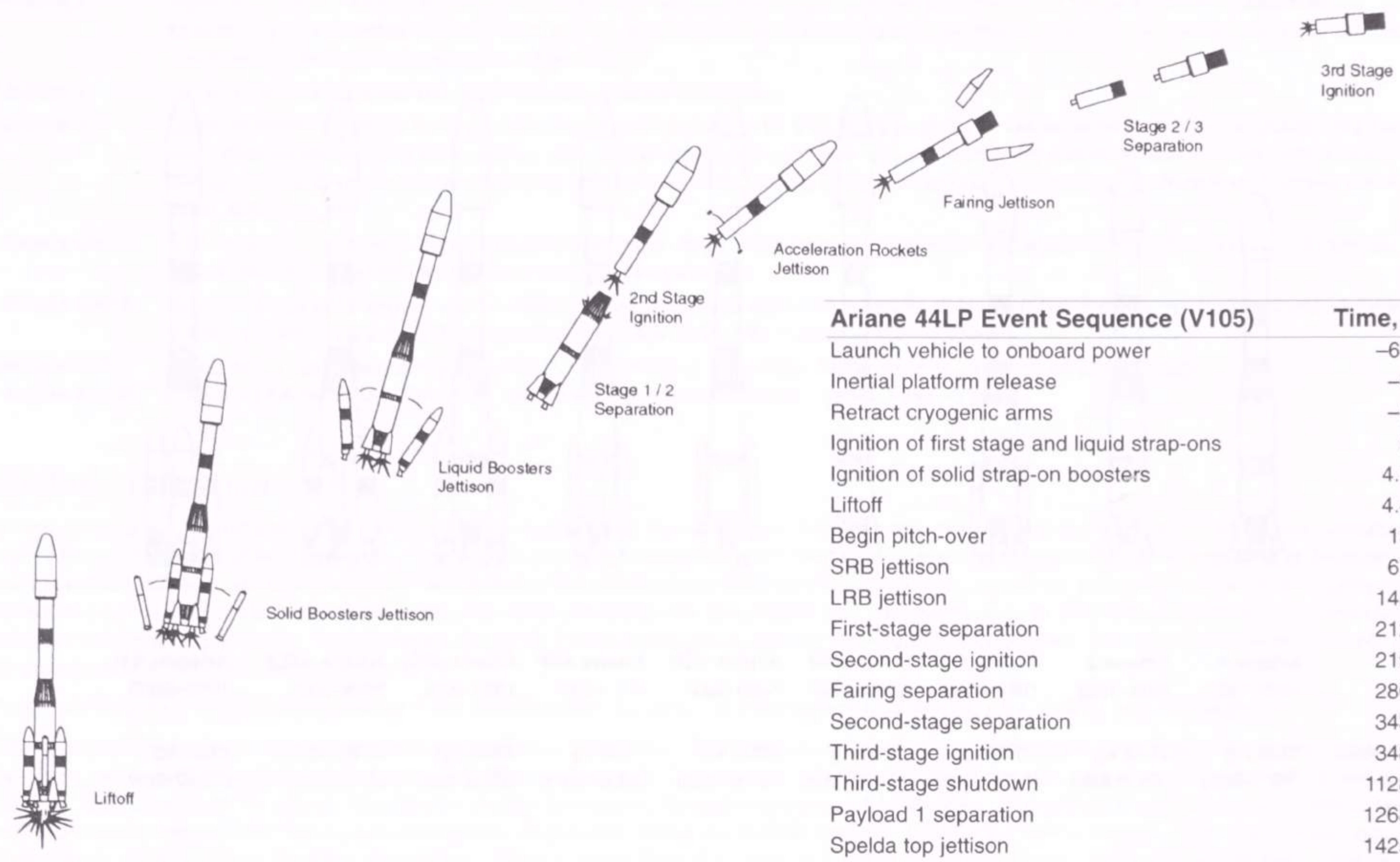
In the BAF, the payload component is assembled and erected, the fairing is assembled, the EPS tanks are filled, and the final electrical checkout is done. The upper payload is attached to the upper Speltra interface and encapsulated in the fairing in the ground-level encapsulation hall, which has a 30 t (66 klbm) crane. The lower payload is integrated directly on the Ariane 5, and then the encapsulated assembly containing the upper payload is lifted over the Ariane 5 and mated to it, enclosing the lower payload. In effect, the BAF replaces the mobile gantry used at ELA-2. After integration and testing, the launch vehicle is transferred on the launch table along a twin railway track from the preparation zone to the launch zone in a fully integrated state. Roll out to the pad can occur less than a day before launch.

The launchpad is in the launch zone. This area includes deflectors and exhaust ducts, a small building protecting the electrical and fluid connections, and a water tower to supply noise suppressing water. A few days before launch, the pad's cryogenic storage tanks are filled with propellants trucked from the cryogenic propellant plant. Four tall towers serve as lightning rods. This simple design permits a sustained launch rate and, in case of a pad accident, minimal stand-down. The launch facility was modified to support the ESC-A cryogenic upper stage, by adding a pair of articulated retractable arms that load LOX and LH_2 into the upper stage from the umbilical tower.

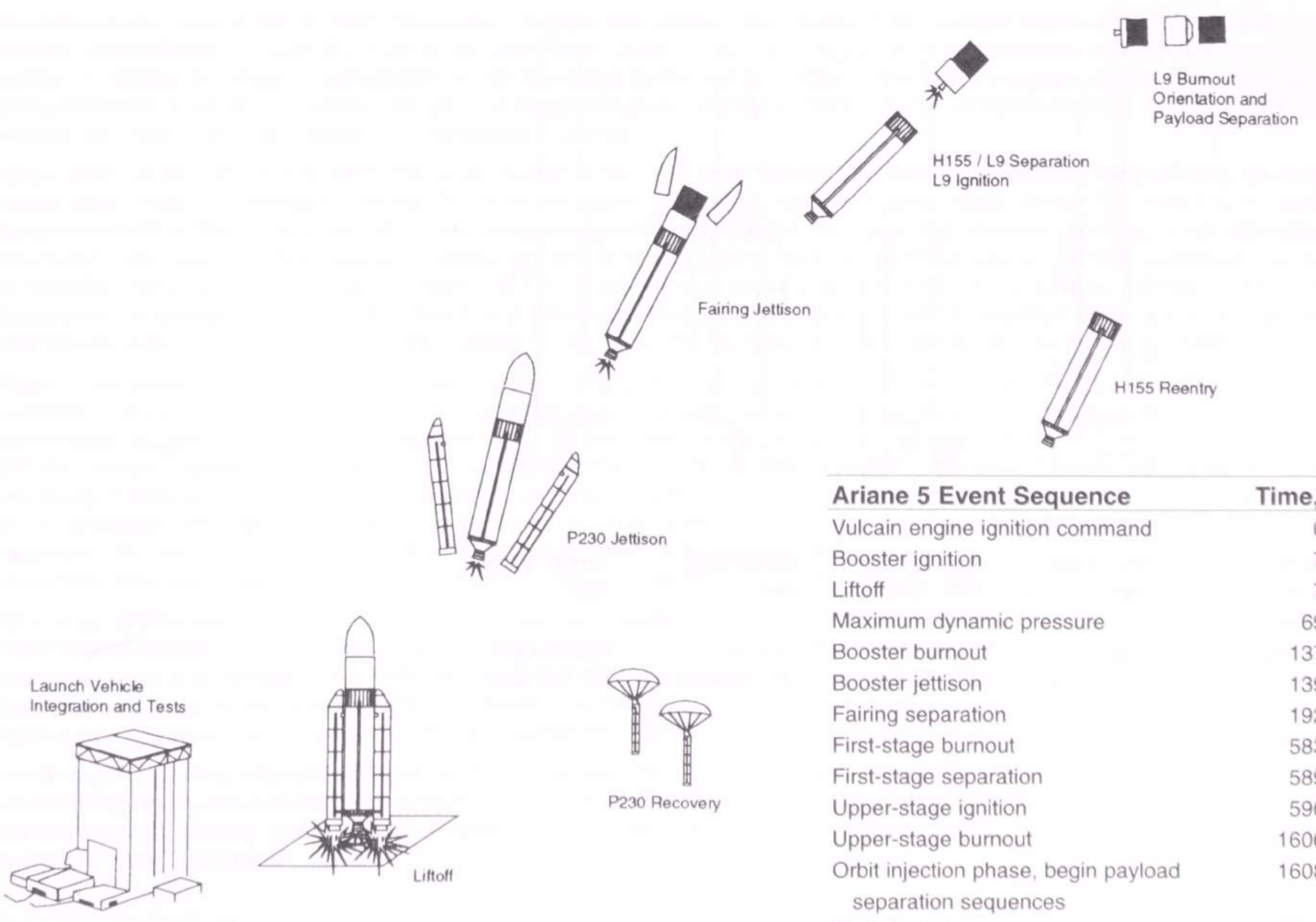
At the CDL3, operations are controlled and monitored down to the moment of launch. Checkout and command equipment allows the activation of two launchers at the same time, so that a second booster can be assembled while the first is on the pad.

PRODUCTION AND LAUNCH OPERATIONS

Flight Sequence



Ariane 44LP Event Sequence (V105)	Time, s
Launch vehicle to onboard power	–60
Inertial platform release	–9
Retract cryogenic arms	–5
Ignition of first stage and liquid strap-ons	0
Ignition of solid strap-on boosters	4.1
Liftoff	4.4
Begin pitch-over	13
SRB jettison	67
LRB jettison	148
First-stage separation	212
Second-stage ignition	219
Fairing separation	286
Second-stage separation	343
Third-stage ignition	348
Third-stage shutdown	1128
Payload 1 separation	1268
Spelda top jettison	1425
Payload 2 separation	1564
Start of third-stage avoidance maneuver	1566
End of mission	1821

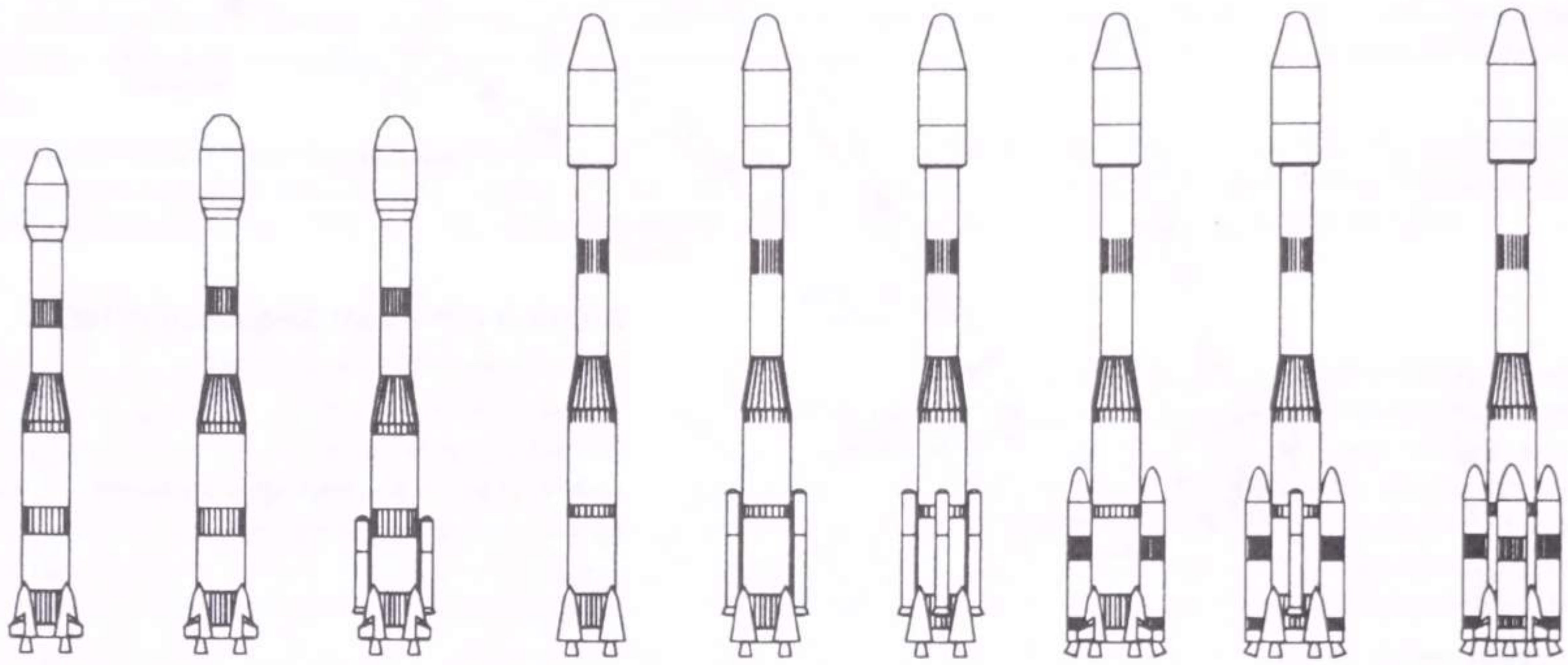


Ariane 5 Event Sequence	Time, s
Vulcain engine ignition command	0
Booster ignition	7
Liftoff	7
Maximum dynamic pressure	69
Booster burnout	137
Booster jettison	139
Fairing separation	192
First-stage burnout	583
First-stage separation	589
Upper-stage ignition	596
Upper-stage burnout	1606
Orbit injection phase, begin payload separation sequences	1608
End of mission	3396

Vehicle History

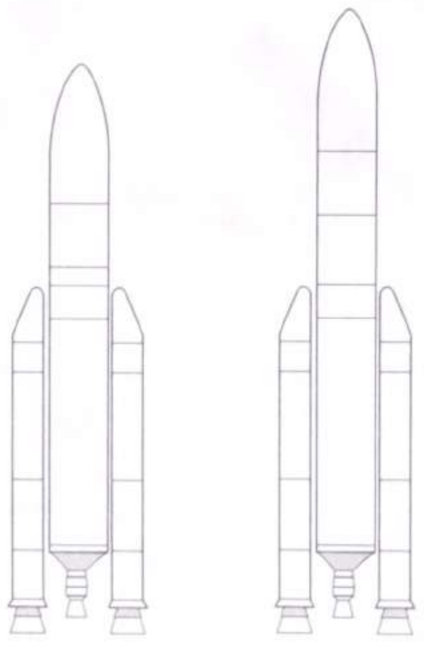
Vehicle Evolution

Retired

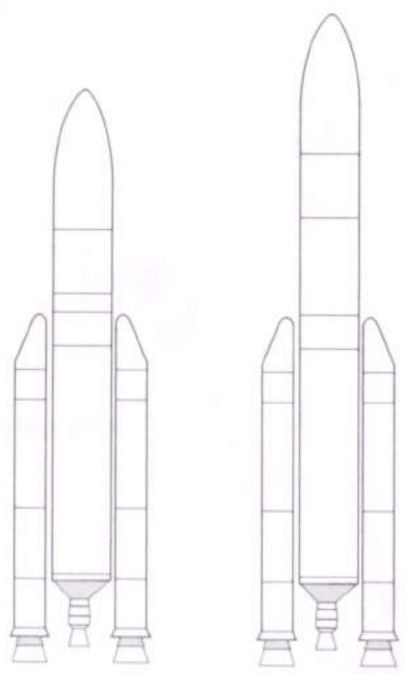


Vehicle	**Ariane 1**	**Ariane 2**	**Ariane 3**	**Ariane 40**	**Ariane 42P**	**Ariane 44P**	**Ariane 42L**	**Ariane 44LP**	**Arianne 44L**
Period of Service	1979–1987	1986–1989	1984–1989	1990–1999	1990–2002	1991–2001	1993–2002	1988–2001	1989–2003
GTO Payload	1850 kg (4070 lbm)	2175 kg (4800 lbm)	2580 kg (5690 lbm)	1900 kg (4190 lbm)	2600 kg (5730 lbm)	3000 kg (6610 lbm)	3200 kg (7050 lbm)	3700 kg (8160 lbm)	4200 kg (9260 lbm)

Operational



In Development



Vehicle	**Ariane 5G**	**Ariane 5ECA**	**Ariane 5ES**	**Ariane 5ECB**
Period of Service	1996–Present	2002–Present	2005	TBD
GTO Payload	6700 kg (14,770 lbm)	10,050 kg (23,150 lbm)	7575 kg (16,700 lbm)	12,000 kg (26,500 lbm)

VEHICLE HISTORY

Vehicle Description

•Ariane 1	First- and second-stage storable propellant, third-stage cryogenic propellant, 3.2-m diam (10.5-ft) fairing.
•Ariane 2	Same as Ariane 1 except increased thrust for first- and second-stage engines, stretched third stage for 25% more propellant, 4 s specific impulse increase in third-stage engine, increased volume in fairing by a modified forward conic section (bicone), and Sylda structure available for two payloads under fairing.
•Ariane 3	Same as Ariane 2 except two solid strap-on boosters are added.
•Ariane 4	Same as Ariane 3 except stretched and strengthened first stage for 61% more propellant, new water tank and new propulsion bay layout: strengthened second and third stages; new vehicle equipment bay; new onboard computer and new laser gyro backup; 4 m (13 ft) fairing; Spelda structure for two payloads; and a mix of boosters either solid strap-ons (30% more propellant than Ariane-3 solids) or liquid strap-ons.
•Ariane 5G	Two large solid strap-ons, cryogenic propellant core stage, small storable propellant final stage, VEB, 5.4-m diam (17-ft) fairing and, if required, a Speltra structure for two or three payloads.
•Ariane 5ECA	Upgraded Vulcain 2 engine with increased thrust, enhanced solid boosters with 2500 kg (5500 lbm) more propellant and reduced structural weight; new ESC-A cryogenic upper stage using HM7B engine from Ariane 4.
•Ariane 5ES	Lower stage upgrades from Ariane 5ECA but enhanced EPS stage from Ariane 5G in place of ESC-A stage.
•Ariane 5ECB	Similar to Ariane 5ECA with larger cryogenic upper stage powered by new Vinci engine.

Historical Summary

Ariane is Europe's second attempt to develop its own multinational launch vehicle, following the unsuccessful Europa project, which was canceled in April 1973 after a series of launch failures. Even before the official cancellation of Europa, the French space agency, CNES, proposed its replacement launch vehicle, code-named L3S (the French acronym for third-generation substitution launch vehicle), as well as a detailed financial plan. France guaranteed it would finance more than half of the program itself and would take responsibility for cost overruns up to 15% of program cost. When the Europa program was finally abandoned, France began to rally its European neighbors behind the Ariane launch system. The substitution launch vehicle was given a go-ahead for development in July 1973.

There are three key organizations involved in the development, operation, and management of Ariane: ESA, CNES, and Arianespace.

ESA, a multinational space organization, was established in December 1973. It was formed out of, and took over the rights and obligations of, two earlier European space organizations: the European Space Research Organization (ESRO) and the European Launcher Development Organization (ELDO). The purpose of the agency is to provide for and to promote, for exclusively peaceful purposes, cooperation among European states in space research and technology and their space applications. Its member states are Austria, Belgium, Denmark, Finland, France, Germany, Ireland, Italy, the Netherlands, Norway, Spain, Sweden, Switzerland, and the United Kingdom. Canada is also a cooperating state. ESA's role with Ariane is direction of the development program and financing of the facilities construction (launchpads and payload preparation facilities).

CNES was created in 1962 to promote the development of French space activities. It developed the Diamant series of launch vehicles. The first launch took place in 1965 from Hammaguir, Algeria, and the last in 1975 from Kourou. CNES is responsible for management of Ariane development programs, construction of the space center facilities, coordination of operations, and operation of the launch complex and payload preparation facility.

Arianespace was set up in March 1980. Fifty founding partners participated in the creation of Arianespace, including 36 of Europe's key aerospace and avionics manufacturers, 13 major European banks, and CNES. Under terms of an intergovernmental agreement, ESA member states transferred production, marketing, and launch responsibilities for the operational Ariane and its uprated versions to Arianespace. ESA is responsible for development and qualification of new Ariane vehicles, and the organization delegates authority to CNES to carry out the development. Once declared operational, the vehicles are turned over to Arianespace for commerical operations.

Ariane launches are conducted at the Guiana Space Center in Kourou, French Guiana. Early French space launch tests were conducted at the French missile testing facility at Hammaguir, Algeria. The base was abandoned in 1967, following Algerian independence. The CSG was set up by the French government in April 1965 and built by CNES. It became operational in April 1968 with the launch of a Veronique sounding rocket. Following the Diamant and Europa programs, CSG was selected for Ariane launch operations based on its proximity to the equator, the wide opening on the ocean allowing all inclination missions, the low population density, and the absence of hurricanes or earthquakes. CSG is operated for ESA by CNES. CNES is also responsible for expanding the launch facilities to meet growing mission demands. Operation and maintenance of the launch facilities are the duty of Arianespace, and CNES is reimbursed by Arianespace for the personnel, facilities, and materials used to support launch operations.

Ariane 1 was defined in 1973 and was intended to achieve a GTO lift capacity for satellite masses of up to 1850 kg (4070 lbm). The development and qualification of Ariane covered a period of eight and a half years, from mid 1973 through 1981. The first three years were devoted to qualifying stage components (engines, structures, and equipment). Later, systems trials were conducted with dynamic and electric mock-ups of the launch vehicle and with the propulsion systems at the stage level. Construction of the SIL (or VAB) at Aerospatiale's Les Mureaux Center was completed in 1976. The second phase of development consisted of qualification at stage level and the construction of the launch facilities in French Guiana. The first launch attempt on 15 December 1979 resulted in a launch abort—all four engines ignited, but were shut down before liftoff. Technicians were flown from France to reconfigure the rocket, and after another scrubbed attempt, it was finally launched on Christmas Eve, 1979. The development cost for Ariane 1 was about 2000 MAU [using year 1986 ECU, equivalent to 2 billion($1.96 billion) at 2002 exchange rates].

As soon as development of Ariane 1 was complete, it became apparent that if Ariane was to remain competitive it would have to be able to launch two PAM-D class payloads simultaneously. So in July 1980, the Ariane 2 and 3 program was started. The launch vehicle was derived from Ariane 1, incorporating a series of modifications that increased lift capacity to GTO by 50%. To allow dual-payload launches, a new structure was developed, named Sylda (systeme de lancement double Ariane). The Ariane 3 version was identical to Ariane 2, with the addition of two solid strap-on boosters. The development cost for Ariane 2 and 3 was about 144 MAU [using year 1986 ECU, equivalent to 144 million ($141 million) at 2002 exchange rates].

The decision to develop Ariane 4 was made by ESA in January 1982, when the need became evident for a more powerful and more flexible launch vehicle (compared to Ariane 2 and 3) to match the trend in the payload market. The major program objectives were to achieve a significant increase in payload mass and volume; to maintain the capability of multiple launches; to create a range of mission-adaptable configurations; and to improve the flexibility of launch operations.

Vehicle History

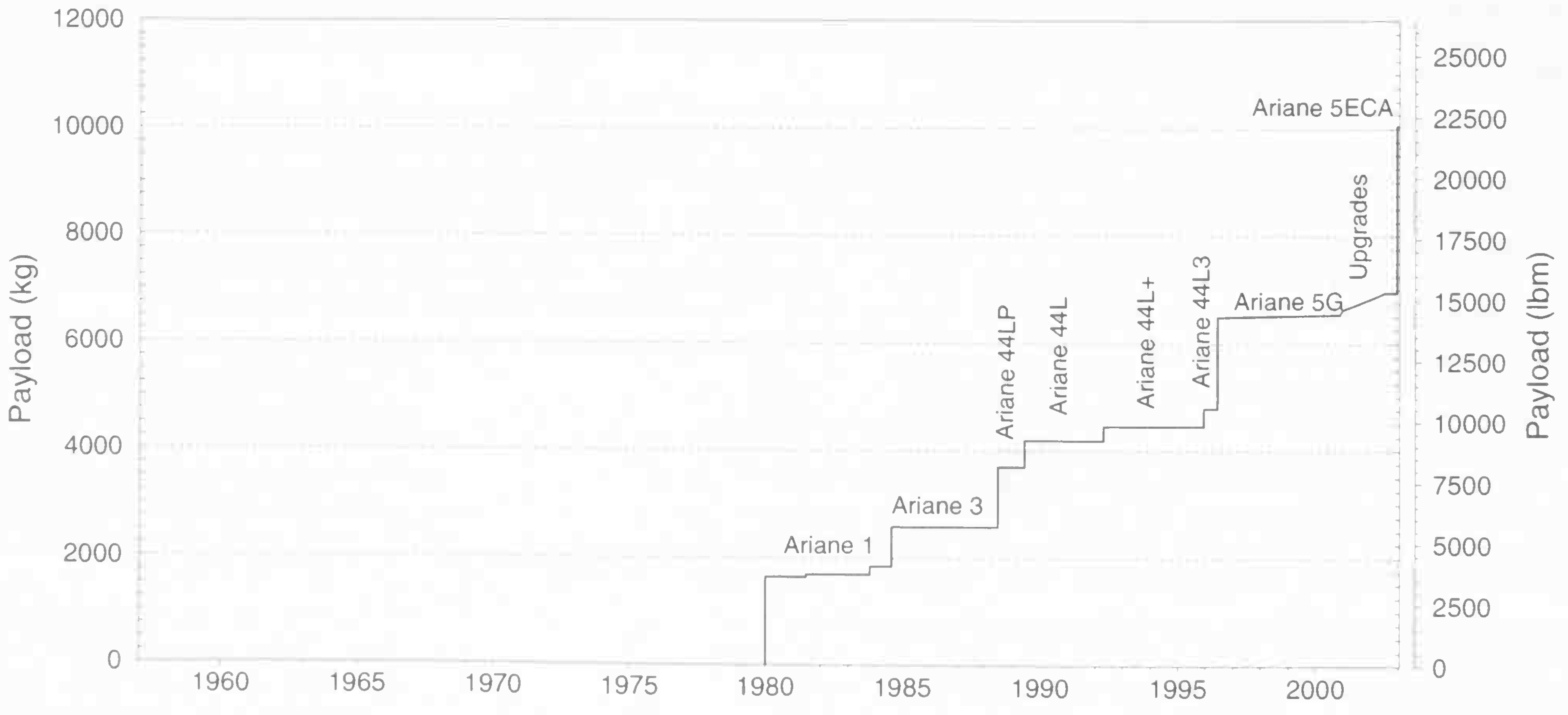
The increase in performance, by up to 90%, over Ariane 3 was achieved by increasing the first-stage propellant mass and by attaching newly developed, more powerful solid or liquid boosters. The diameter of the payload fairing was increased to accommodate bigger payloads. A new supporting structure, the Spelda (structure porteuse externe pour lancements double Ariane) allows multiple payload launches. Fairings of various heights and different sizes of Spelda allow adaptation of the launch vehicle to the volume of the payloads to be launched. Six different versions of Ariane 4 are offered by varying the number of liquid and/or solid strap-ons. A new launch complex, ELA-2, was built to allow both a higher launch rate and greater flexibility in launch scheduling.

The majority of the Ariane 4 development work was completed by 1986. Manufacture of the flight hardware for the demonstration flight started in 1985. All launch vehicle hardware except the third stage was ready by November 1987. The demonstration flight had to be postponed because of development difficulties on the third-stage engine. The launch campaign finally started in December 1987, before being interrupted and then resumed, leading up to the flawless first launch on 15 June 1988. The development cost for Ariane 4 was about 476 MAU [using year 1986 ECU equivalent to 476 million ($467 million) at 2002 exchange rates]. The highest contributors to Ariane 4 development were France, 62%, Germany, 18%, and Italy, 5%.

Once operational, Ariane 4 proved to be the most successful of the Ariane versions, flying more than 80 times by 1998. Flexible performance capability and a focus on commercial customers make it one of the most popular commercial launch vehicles for clients around the world. Arianespace captured 50–60% of the launch market in the early 1990s, though its market share has begun to decrease with the arrival of Russian boosters and competitors serving LEO constellations. Launches of Ariane 4 are expected to continue until at least 2003, though they will gradually be replaced by the newer, more capable Ariane 5.

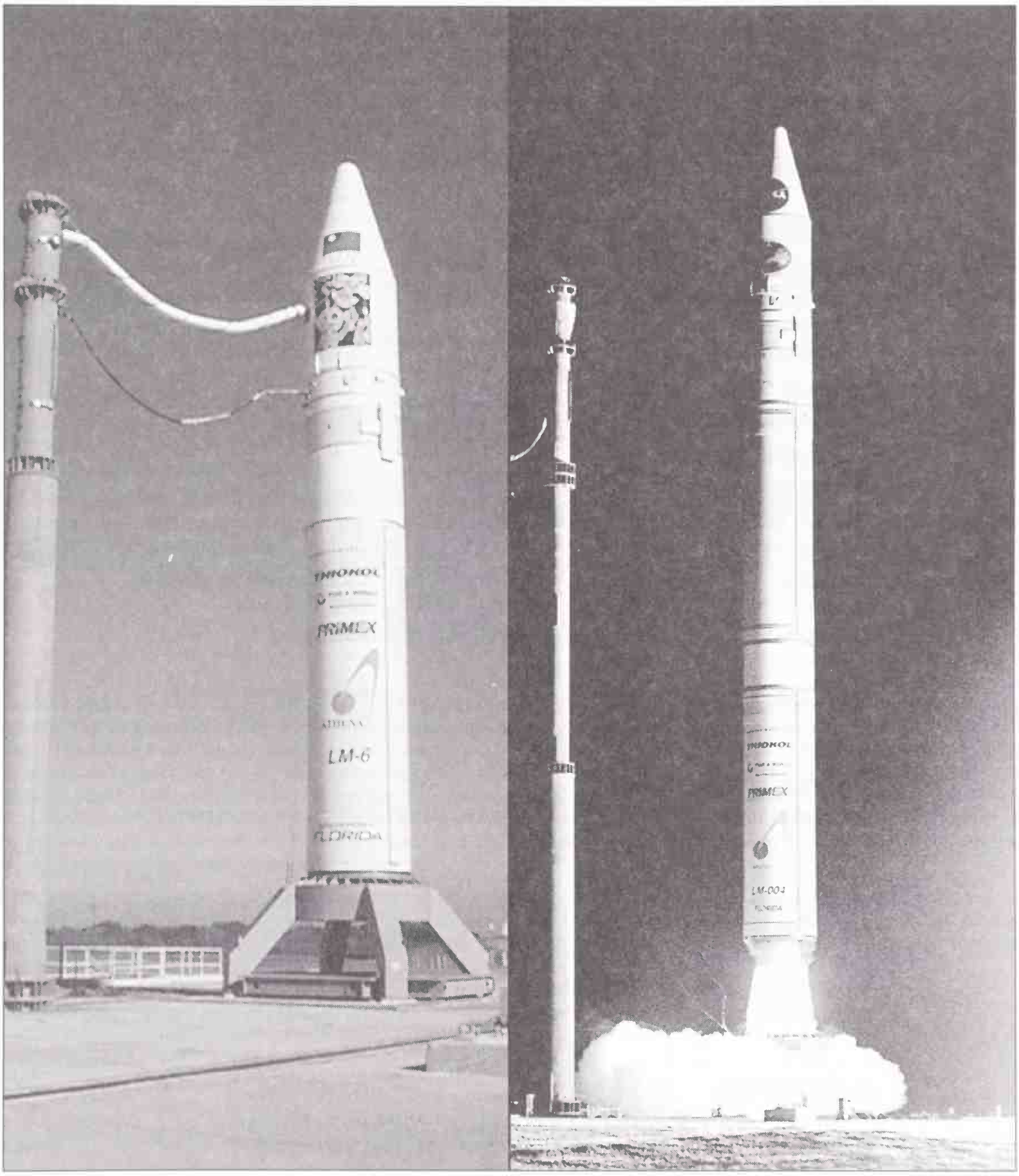
Ariane 5, the successor to Ariane 4, is a completely new design. The Ariane 5 development was approved in 1988 after a preparatory program initiated in 1984. The purpose of the new vehicle would be to carry larger commercial payloads to GTO and to launch the Hermes crewed spaceplane for ESA. Objectives of the Ariane 5 development program were to develop a launch vehicle that could deliver into GTO one or more satellites with a total mass of 6800 kg (15,000 lbm) for a single-launch configuration or 5900 kg (13,000 lbm) in the dual-launch configuration; carry an automated cargo transfer vehicle or a crew transfer vehicle for rendezvous with the International Space Station; provide a 4.57-m diam (15-ft) payload volume; meet reliability target of 0.98 for the launcher's total mission; comply with safety target of 0.999 with respect to crewed flights; and satisfy a cost goal of using Ariane 5 for a dual launch into GTO that would be at least 10% lower than that of using an Ariane 44L, assuming eight launches a year. This would correspond to a 45% reduction in the cost-per-pound compared with Ariane 44L.

The proposed crewed missions for Ariane 5 were eliminated when ESA canceled the proposed Hermes spaceplane, but development of Ariane 5 continued. The first demonstration launch occurred in June 1996 carrying the four Cluster science satellites. However, approximately 36 s after liftoff, the redundant IRS failed, causing the launch vehicle to veer off course and break up from high aerodynamic forces. The inquiry board found that the IRS alignment software had been duplicated from the Ariane 4 design without proper testing for difference in the Ariane 5 trajectory. A software routine within the IRS converts the horizontal bias parameter from a 64-bit floating point variable to a 16-bit integer. This parameter is much larger for an Ariane 5 trajectory early in flight than for an Ariane 4, and the larger value could not be represented as a 16-bit variable. Because both IRS units had the same software, both failed simultaneously and sent bad data to the onboard computer. As a result of the failure, Ariane 5 operations were delayed significantly. The second test flight was postponed until October 1997, and a third test flight was added to the schedule to demonstrate reliability before beginning commercial operations. As a result, Ariane 5 began commercial operations in 1999, roughly three years behind schedule. Ariane 5 has had some significant successes, notably the launches of the XMM x-ray astronomy observatory and Envisat, Europe's largest Earth observation satellite, as well as the launches of some of the largest commercial communications satellites ever. However, the program continues to be troubled by failures, the most significant of which was the failure of the first Ariane 5ECA in 2002. The impending phase-out of Ariane 4 in 2003 increased the pressure on Ariane 5 to prove its reliability, as Europe will no longer have a proven back-up vehicle.



Growth of Ariane Performance to GTO

ATHENA



Courtesy Lockheed Martin Corporation.

The Athena I and II are small, solid-propellant launch vehicles. They were developed commercially by Lockheed Martin beginning in the early 1990s. An Athena I carried ROCSAT, the first satellite for Taiwan, in January 1999 (left). The first flight of the Athena II in January 1998 carried NASA's Lunar Prospector spacecraft to the moon (right).

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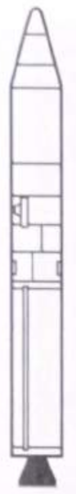
Web site: www.ast.lmco.com/programs/athena.shtml

Athena I and II

General Description



Athena I

Athena II

Summary

Athena was developed commercially by Lockheed Martin to address the market for delivering small spacecraft to LEO. The Athena family includes two small launch vehicles, the three-stage Athena I and the four-stage Athena II. Both vehicles are made up of solid motors and a small liquid injection stage, called the Orbit Adjust Module (OAM). The vehicles are modular, and the Athena II shares all the components of the Athena I with the addition of an extra Castor® 120 solid motor.

Status

Operational. Athena I first launch in 1995. Athena II first launch in 1998.

Origin

United States

Key Organizations

Marketing Organization	Lockheed Martin Commercial Launch Services
Launch Service Provider	Lockheed Martin Space Systems
Prime Contractor	Lockheed Martin Space Systems

Primary Missions

Small payloads to LEO

Estimated Launch Price

Athena I: $40–45 million (Lockheed Martin, 2002)

Athena II: $45–50 million (Lockheed Martin, 2002)

Spaceports

Launch Site	Cape Canaveral AFS, SLC-46
Location	28.5° N, 81.0° W
Available Inclinations	28.5–50 deg

Launch Site	Kodiak Launch Complex
Location	57.6° N, 152.2°W
Available Inclinations	64–116 deg

Performance Summary

The mass of the payload adapter must be subtracted from the performance shown below.

	Athena I	Athena II
185 km (100 nmi), 28.5 deg	820 kg (1805 lbm)	2065 kg (4520 lbm)
185 km (100 nmi), 90 deg	545 kg (1200 lbm)	1575 kg (3470 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	665 kg (1465 lbm)	1735 kg (3825 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	360 kg (790 lbm)	1165 kg (1565 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 28.5 deg	No capability	590 kg (1290 lbm) with optional Star 37FM kick stage
Geostationary Orbit	No capability	

Flight Record (through 31 December 2003)

Total Orbital Flights	7
Launch Vehicle Successes	5
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	2

Flight Rate

0–2 per year

Nomenclature

The Athena launch vehicle family was originally developed at Lockheed Missiles and Space Company. Because they were the only space launch vehicles built by Lockheed, they were referred to as the Lockheed Launch Vehicles (LLV 1 and 2). After Lockheed merged with Martin Marietta in 1995, the vehicles were renamed Lockheed Martin Launch Vehicles (LMLV 1 and 2). Because Lockheed Martin also builds the Atlas and Titan launch vehicles, the name was not fully appropriate. Following the first successful launch in 1997, the vehicles were renamed Athena I and Athena II. The name changes do not reflect any differences in the launch system.

Cost

In 2002, Lockheed Martin reported the following typical price ranges for Athena launch services:

Athena I: $40–45 million

Athena II: $45–50 million

Athena launch prices have increased significantly in order to cover fixed costs at a flight rate of less than one launch per year.

Availability

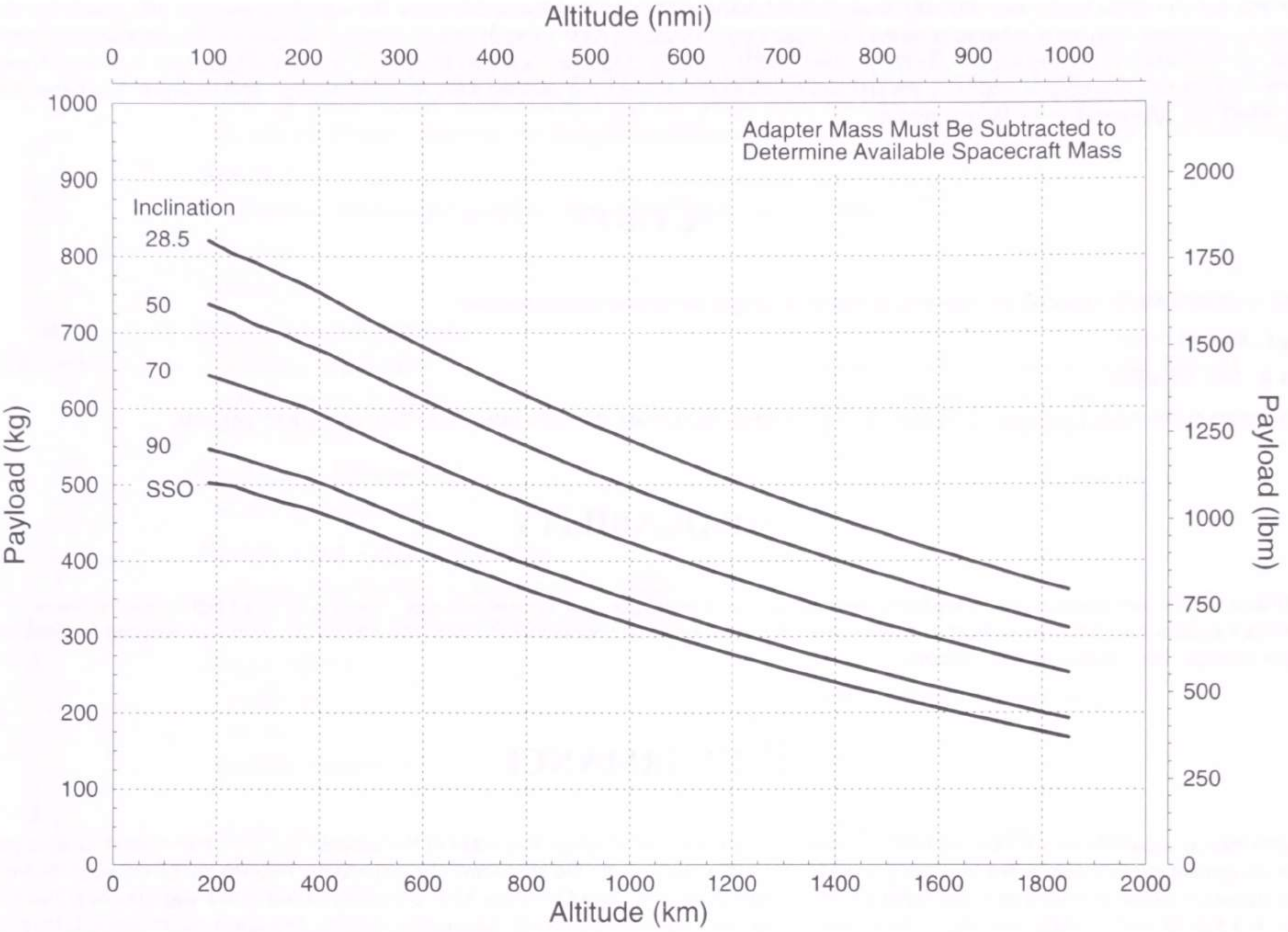
Both Athena I and II are operational and available commercially for domestic or international payloads. The first Athena I flight ended in failure in 1995, but Athena I successfully returned to flight in August 1997. Athena II became operational in 1998. The Athena program was reduced to a low level of activity in October 2001 from a lack of demand.

Performance

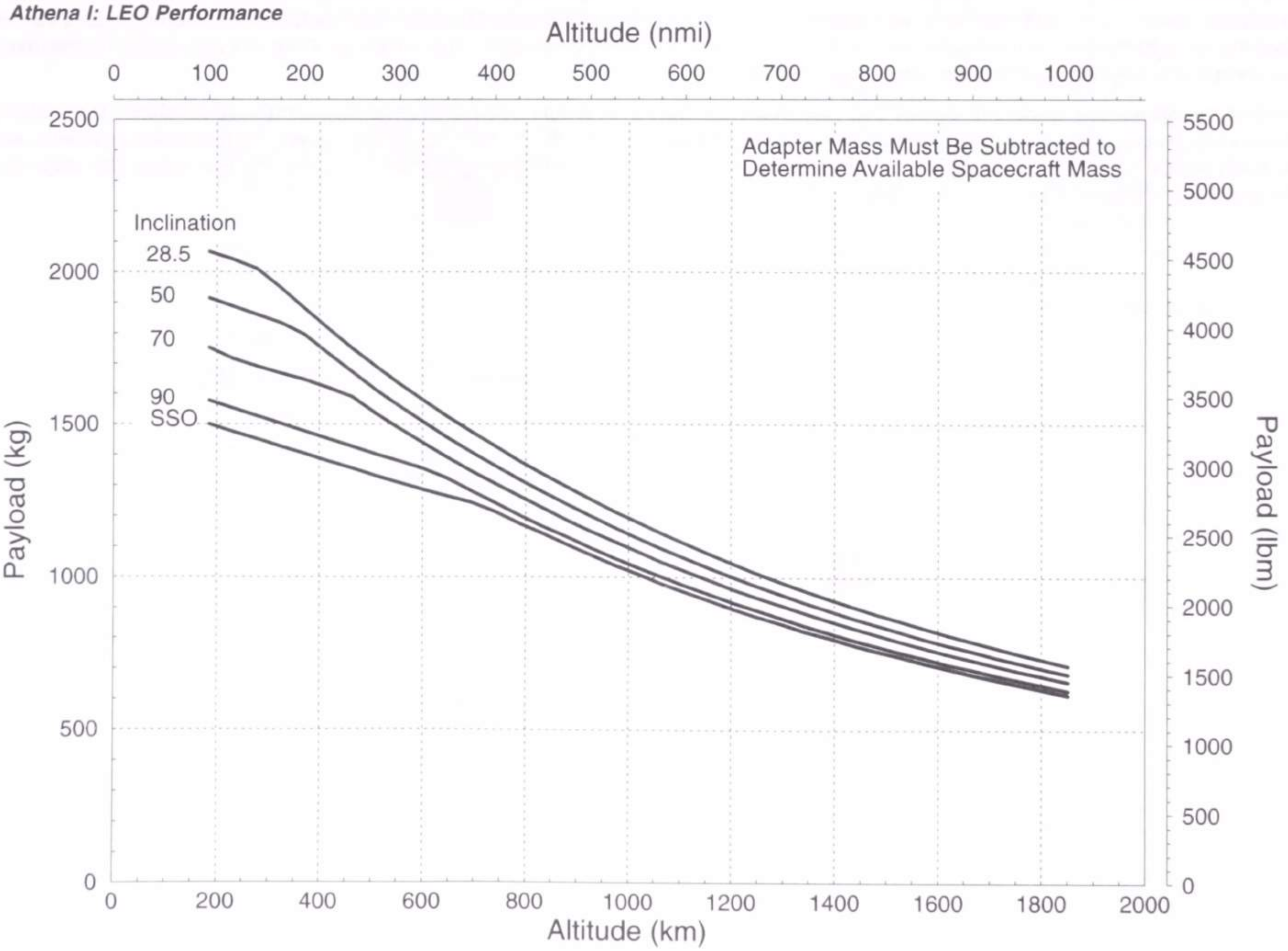
The performance capability shown includes both the spacecraft and payload adapter. The mass of the adapter and any other mission-unique payload support equipment must be subtracted to determine the performance available for the spacecraft. Typical adapter masses are 50–75 kg (110–165 lbm). Athena can reach inclinations from 28.5 deg to roughly 50 deg launching from Cape Canaveral AFS, or roughly 64 deg to 115 deg launching from Kodiak, Alaska. In a typical launch profile, the two or three solid stages fire in sequence to reach near-orbital velocity. The small, liquid-fueled OAM provides orbit circularization, velocity trim, and precision orbit injection. For missions to high-altitude LEO orbits, the OAM performs an initial burn for injection into an elliptical transfer orbit, and then performs a separate circularization burn at apogee. The OAM flight performance reserve ensures that Athena can meet the required injection requirements with 99.7% probability of sufficient performance. The restartable OAM can also deploy multiple payloads and can deorbit after payload separation to reduce orbital debris.

Lockheed Martin will replace the current Orbus® 21D solid motor with the lighter weight Orbus 21G beginning with the ninth vehicle. The performance shown assumes the use of the Orbus 21G. Performance of older vehicles is roughly 50–75 kg (110–165 lbm) lower. For high-energy missions, such as GTO or Earth escape trajectories, a small solid kick stage based on the Thiokol Star 37FM motor can be attached to the spacecraft. This stage was first used to send Lunar Prospector to the moon in 1998.

Performance

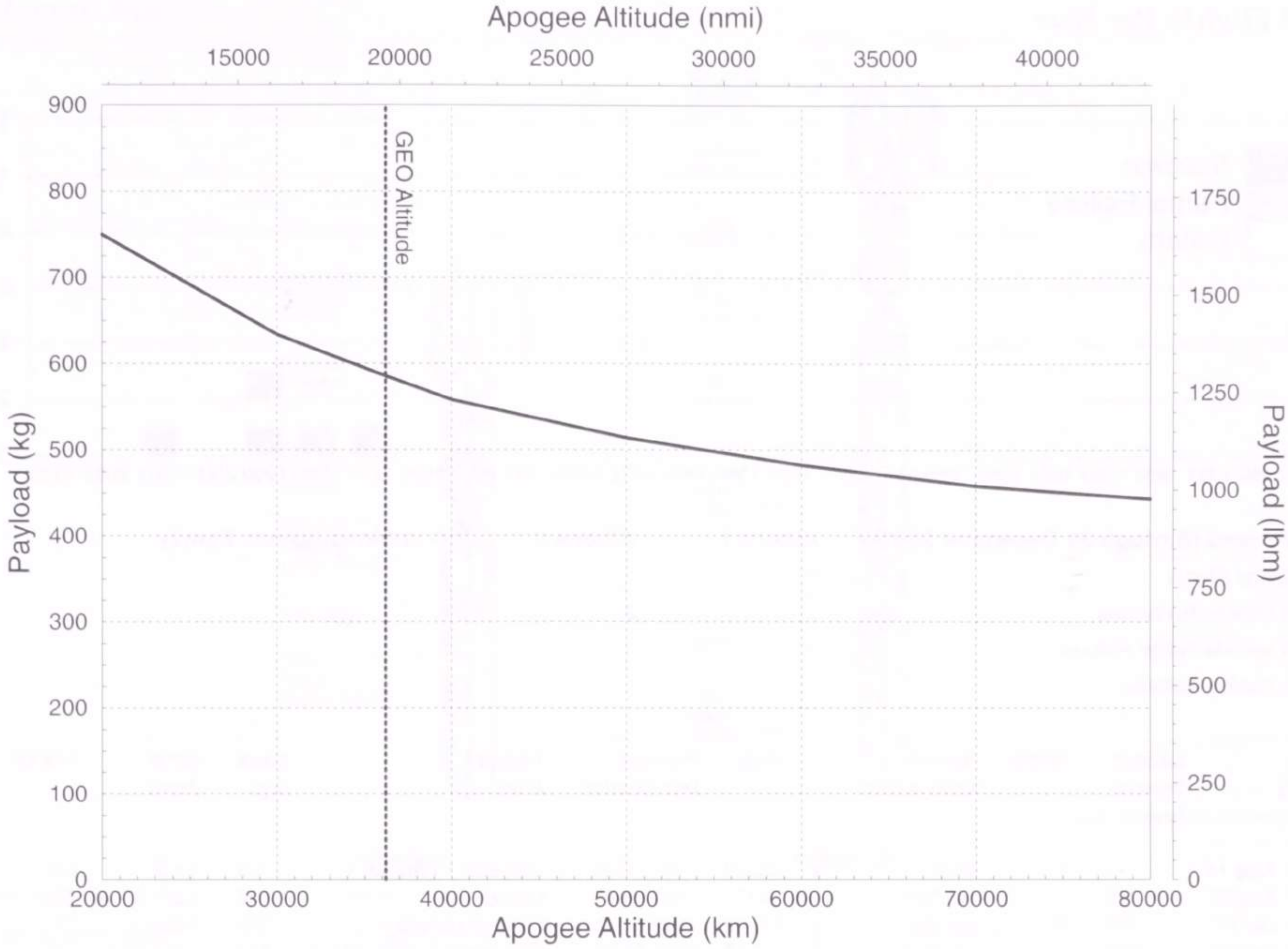


Athena I: LEO Performance

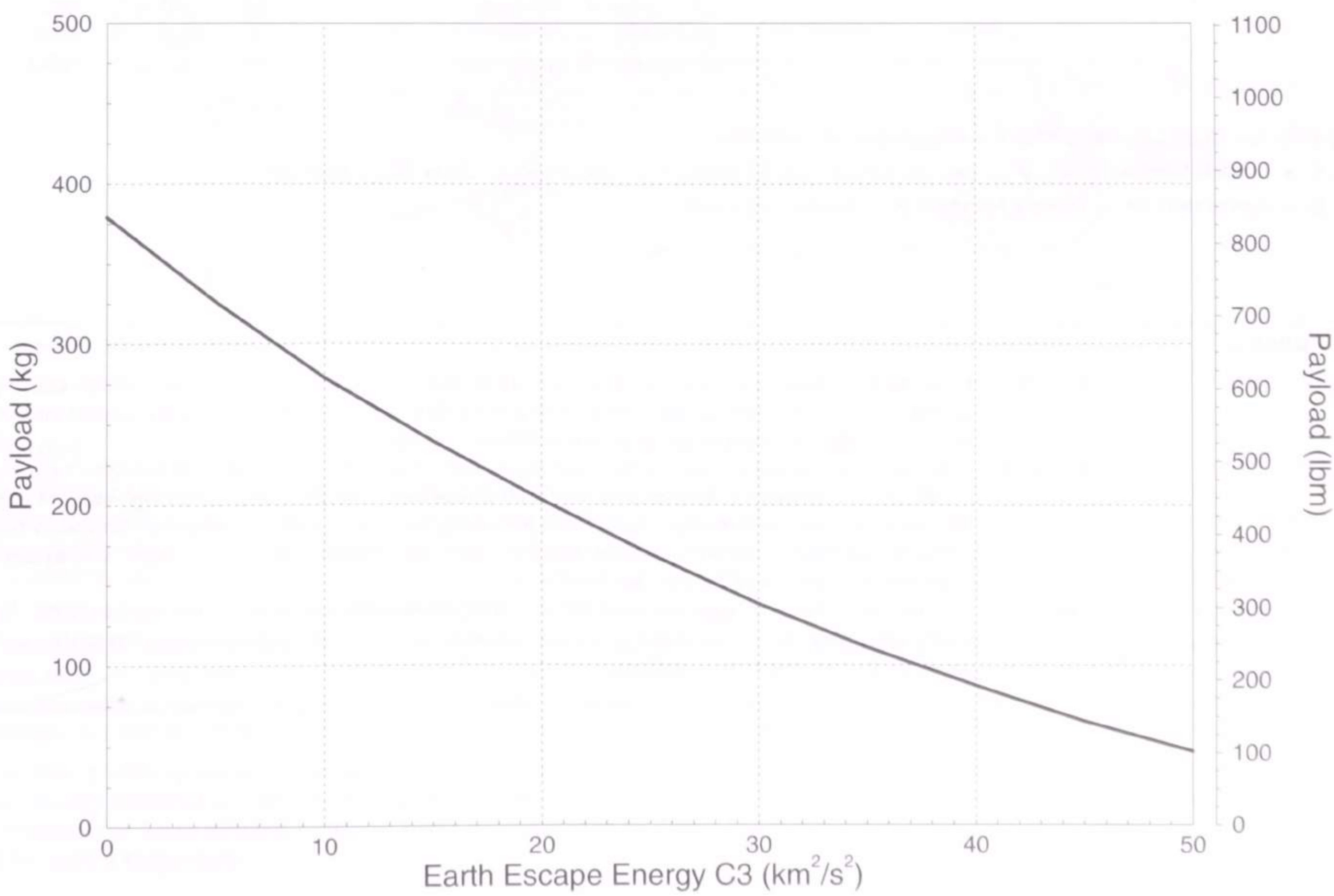


Athena II: LEO Performance

PERFORMANCE



Athena II: GTO Capability



Athena II: Earth Escape Capability

Flight History

Orbital Flights Per Year

Flight Record (through 31 December 2003)	Athena I	Athena II	Combined Athena Family
Total Orbital Flights	4	3	7
Launch Vehicle Successes	3	2	5
Launch Vehicle Partial Failures	0	0	0
Launch Vehicle Failures	1	1	2

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T,F	**1**	**1995**	**Aug 15**	**—**	**1**	**DLV**	**V**	**SLC 6**	1995 F04A		Gemstar 1 (VitaSat 1)	113	LEO	CML	USA
S	2	1997	Aug 23	739	1	LM-002	V	SLC 6	1997 044A		Lewis	404	LEO (97.5)	CIV	USA
	3	1998	Jan 07	137	2	LM-004	C	LC 46	1998 001A		Lunar Prospector	295	Moon	CIV	USA
	4	1999	Jan 27	385	1	LM-006	C	LC 46	1999 002A		Chunghua 1 (ROCSAT 1)	410	LEO (35)	CIV	Taiwan
F	**5**		**Apr 27**	**90**	**2**	**LM-005**	**V**	**SLC 6**	1999 F01A		Ikonos	726	SSO	CML	USA
	6		Sep 24	150	2	LM 007	V	SLC 6	1999 051A		Ikonos	726	SSO	CML	USA
	7	2001	Sep 30	737	1	LM-001	K	LP1	2001 043A	A	Starshine 3	90	LEO (67)	CIV	USA
									2001 043B	A	PICOsat	68	LEO (67)	MIL	USA
									2001 043C	A	PCSat	10	LEO (67)	NGO	USA
									2001 043D	A	SAPPHIRE	18	LEO (67)	NGO	USA

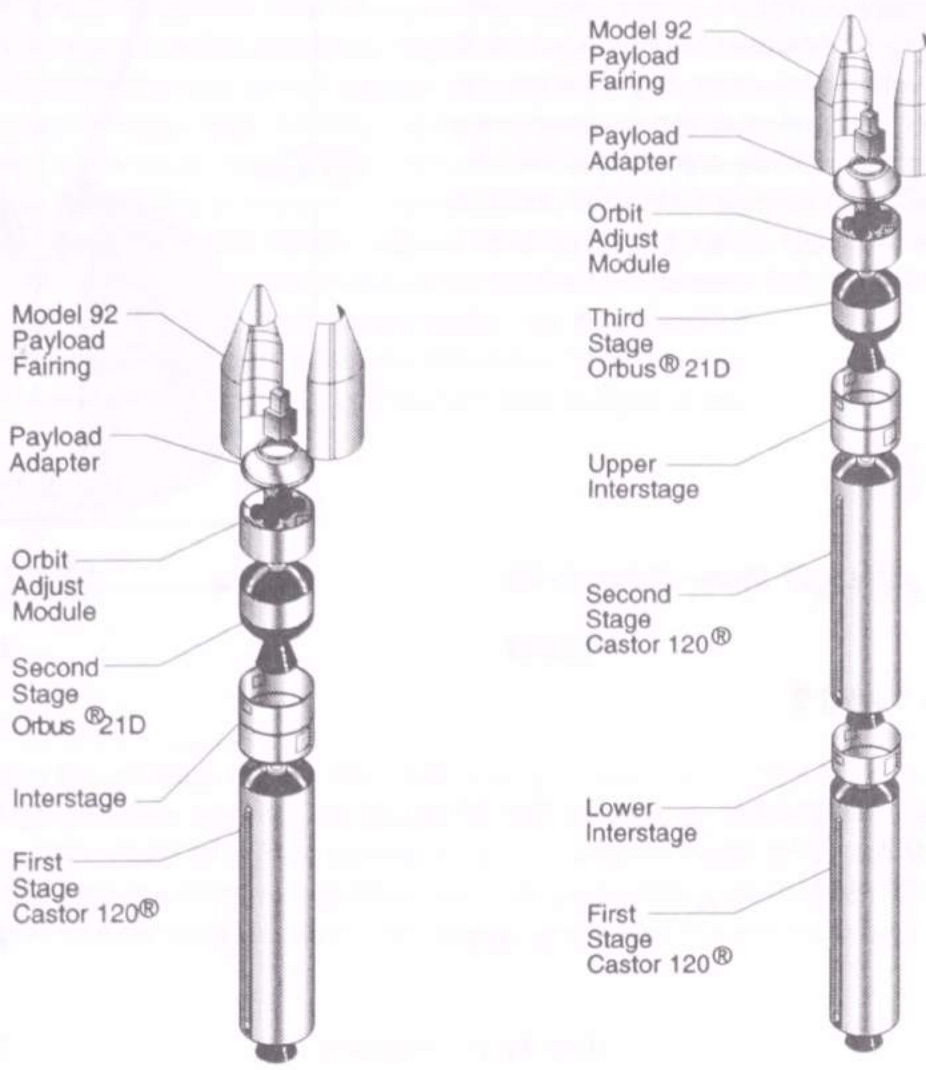
V = Vandenberg AFB; C = Cape Canaveral AFS; K = Kodiak Launch Complex
T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly
Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Failure Descriptions:

F	1995 Aug 15	DLV	1995 F04	Expended hydraulic fluid burned in first stage aft section, damaging nozzle feedback cables causing loss of gimbal control and tumbling. Separately, arcing in the IMU high-voltage power supply caused loss of attitude reference. Flight terminated by range safety officer at T+160 s.
S	1997 Aug 23	LM-002	1997 044A	The spacecraft started an unexpected spinning motion of about two rpm in its orbit only four days after being launched. As a result, the spacecraft's twin solar arrays were unable to capture enough sunlight to properly recharge the onboard batteries. Contact with the spacecraft was lost shortly thereafter. Engineers believe the spinning was started with an accidental firing by one of the satellite's steering thrusters. The satellite reentered the atmosphere 28 September 1997.
F	1999 Apr 27	LM-005	1999 F01	Payload fairing failed to separate, and the extra weight prevented the vehicle from reaching orbit. During the fairing separation event, the shock of the circumfrential ordnance firing disconnected the cable carrying the signal to fire the longitudinal ordnance.

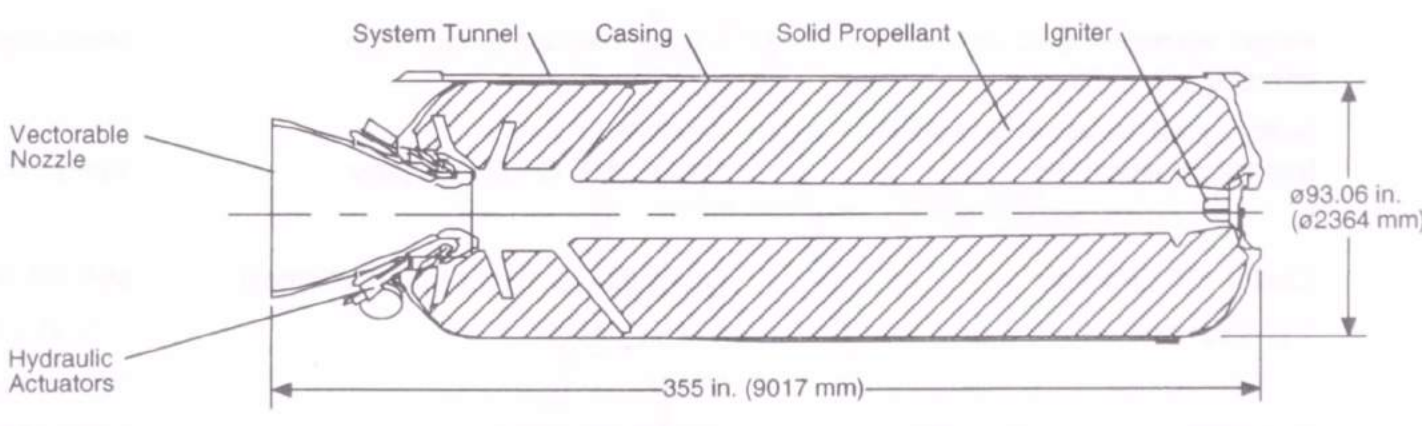
Vehicle Design

Overall Vehicle



Courtesy Lockheed Martin Corporation.

	Athena I	**Athena II**
Height	18.9 m (61.9 ft)	28.2 m (93.2 ft)
Gross Liftoff Mass	66,300 kg (146,100 lbm)	120,700 kg (266,100 lbm)
Thrust at Liftoff	1450 kN (325.9 klbf)	1450 kN (325.9 klbf)



Castor 120 Solid Motor

Stages

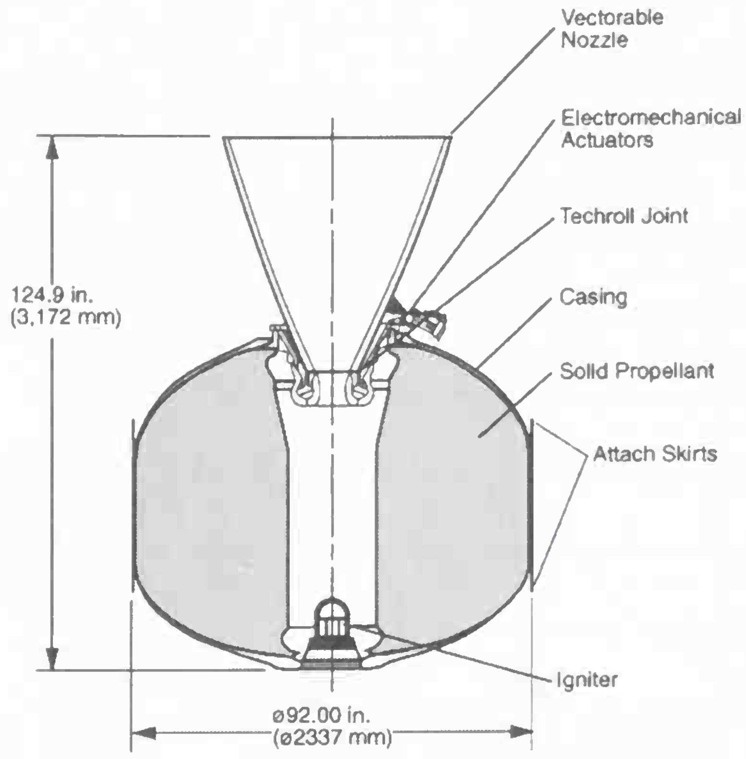
The primary propulsion for the Athena vehicles is provided by two or three solid stages. The first stage is based on Thiokol's Castor 120 motor. The Athena II includes a second Castor 120 motor as a second stage. An Orbus 21D motor provides the basis for the second stage on the Athena I, and the third stage on Athena II. To avoid confusion over stage numbering, this stage is generally referred to as the equipment section boost motor (ESBM). The ESBM is topped by the OAM, also called the equipment section, which houses the vehicle avionics and has a small monopropellant liquid-propulsion system. The OAM supports the payload adapter and fairing.

The Castor 120 is a commercial SRM developed by Thiokol. It is a derivative of the Peacekeeper ICBM first-stage motor, with modifications for space launch applications. It has the same diameter as the Peacekeeper, but is about 0.5 m (1.7 ft) longer to include more propellant. A slight change to the propellant formulation results in a longer burn time that is more suitable for space launch vehicles. The casing is filament-wound graphite–epoxy composite, and TVC is provided by a new blowdown hydraulic nozzle gimbal system similar to those developed for the Castor IVB and SSLV Taurus first stage.

The Pratt & Whitney Orbus 21D motor is derived from the Orbus 21 motors used in the IUS and TOS upper stages for the Space Shuttle and Titan. The primary difference is a carbon–phenolic nozzle exit cone, which is stronger and less expensive but also heavier than the carbon–carbon exit cone it replaces. For future missions Lockheed Martin will upgrade to the Orbus 21G motor, which has a lightweight graphite–epoxy motor casing in place of the current Kevlar case.

VEHICLE DESIGN

The OAM is a small, monopropellant hydrazine stage that provides precise orbit injection as well as attitude control throughout flight. It also contains the avionics bays. The combined propulsion–ACS system is a monopropellant blowdown system, and therefore the thrust decays during flight as the tank pressure decreases. Axial propulsion is provided by four small 220 N (50 lbf) hydrazine thrusters spaced 90 deg apart around the circumference of the stage. The thrusters can be restarted almost indefinitely because they were originally developed for spacecraft attitude control applications. The axial propulsion thrusters and lateral-radial ACS thrusters are identical and are fed from common tankage. The OAM is available with four or six propellant tanks, depending on mission requirements. The hydrazine tanks are mounted in pairs in an egg-crate frame structure inside the stage.



Orbus 21 Upper Stage Motor

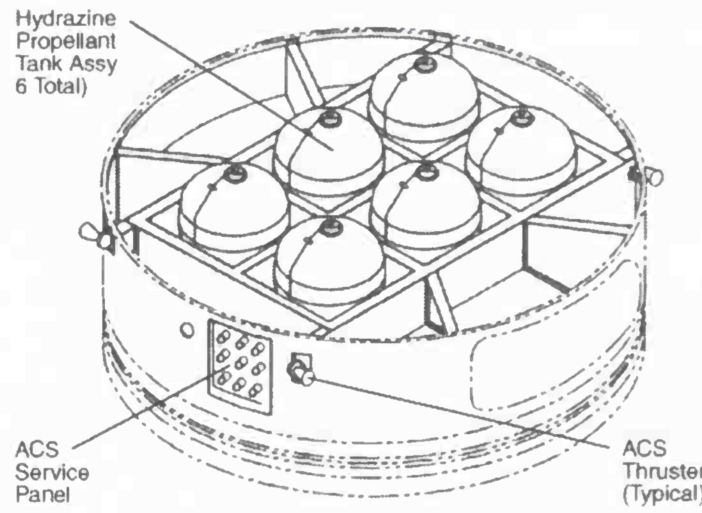
Courtesy Lockheed Martin Corporation.

	Athena I Stage 1 Athena II Stages 1 and 2	ESBM	OAM
Dimensions			
Length (with interstage)	Athena I Stage 1: 10.7 m (35.2 ft) Athena II Stage 1: 10.2 m (33.3 ft) Athena II Stage 2: 12.5 m (41.0 ft)	3.0 m (10.0 ft)	1.0 m (3.3 ft)
Diameter	2.3 m (7.7 ft)	2.3 m (7.7 ft)	2.3 m (7.7 ft)
Mass			
Propellant Mass	48.7 t (107.4 klbm)	9780 kg (21,560 lbm)	236 or 354 kg (520 or 780 lbm)
Inert Mass	Athena I Stage 1: 4375 kg (9650 lbm) Athena II Stage 1: 4450 kg (9800 lbm) Athena II Stage 2: 4510 kg (9950 lbm)	Orbus 21D: 1030 kg (2280 lbm) Orbus 21G: 935 kg (2060 lbm)	360 kg (790 lbm)
Gross Mass	53.1–53.2 t (117.1–117.4 klbm)	Orbus 21D: 10,810 kg (23,840 lbm) Orbus 21G: 10,715 kg (23,620 lbm)	596 or 714 kg (1310 or 1570 lbm)
Propellant Mass Fraction	0.92	Orbus 21D: 0.90 Orbus 21G: 0.91	0.40 or 0.50
Structure			
Type	Motor: filament-wound monocoque Interstage: Monocoque	Filament-wound monocoque	Monocoque
Material	Motor: graphite–epoxy composite Interstage: aluminum	Orbus 21D: Kevlar Orbus 21G: Graphite–epoxy	Structure: Aluminum–lithium Tanks: Titanium
Propulsion			
Motor Designation	Castor 120 (Thiokol)	Orbus 21D or 21G (Pratt & Whitney)	MR-107 (General Dynamics)
Number of Motors	1 (1 segment)	1 (1 segment)	4 Axial + 6 ACS
Propellant	HTPB	HTPB	Monopropellant hydrazine
Average Thrust	Sea level: 1450 kN (325.9 klbf) Vacuum: 1604 kN (360.5 klbf)	187 kN (42.4 klbf)	Initially 890 N (200 lbf) total axial, decreases with time
Isp	Sea level: 253 s Vacuum: 280 s	293 s	220–225 s
Chamber Pressure	Peak: 100 bar (1450 psi)	55 bar (800 psi)	8.8–2.2 bar (125–32 psi)
Nozzle Expansion Ratio	17:1	63.8	21.5:1
Propellant Feed System	—	—	Blowdown pressure fed
Throttling Capability	None	None	Thrusters can be pulsed, mimicking throttle capability
Restart Capability	None	None	Effectively unlimited
Tank Pressurization	—	—	Helium
Attitude Control			
Pitch, Yaw,	Hydraulic nozzle gimbaling ±5 deg	Electromechanical nozzle gimbaling ±3 deg	Off-modulation of axial thrusters
Roll	Hydrazine ACS on OAM stage	Hydrazine ACS on OAM stage	Lateral thrusters
Staging			
Nominal Burn Time	83.4 s	150 s	Mission specific, maximum is roughly 1500 s
Shutdown Process	Burn to depletion	Burn to depletion	Command shutdown
Stage Separation	Athena I: spring separation of ESBM Athena II: Stage 2 fire-in-the-hole hot separation	OAM hot separation	

Vehicle Design

Attitude Control System

Each solid stage has a gimbaled nozzle to control pitch and yaw. The Castor 120 has a blowdown hydraulic system to drive actuators that swivel the nozzle ±5 deg on a flexseal joint. The Orbus 21D has an electromechanical gimbal system that rotates the nozzle ±3 deg. The OAM ACS thrusters are initiated after clearing the launch tower to control roll rate during the boost phase. Unlike most other launch vehicles, Athena does not require a fixed roll orientation, thus the vehicle is allowed to roll slowly during flight. The flight control system simply converts pitch and yaw nozzle commands to account for the current vehicle alignment. The OAM includes four axial thrusters that provide propulsion for orbit injection. During these propulsive burns, each axial thruster can be pulsed off to create pitch and yaw control moments. Roll is controlled with the lateral thrusters. The OAM thrusters also provide full three-axis attitude control during all nonpropulsive phases, including payload separation.



Courtesy Lockheed Martin Corporation.

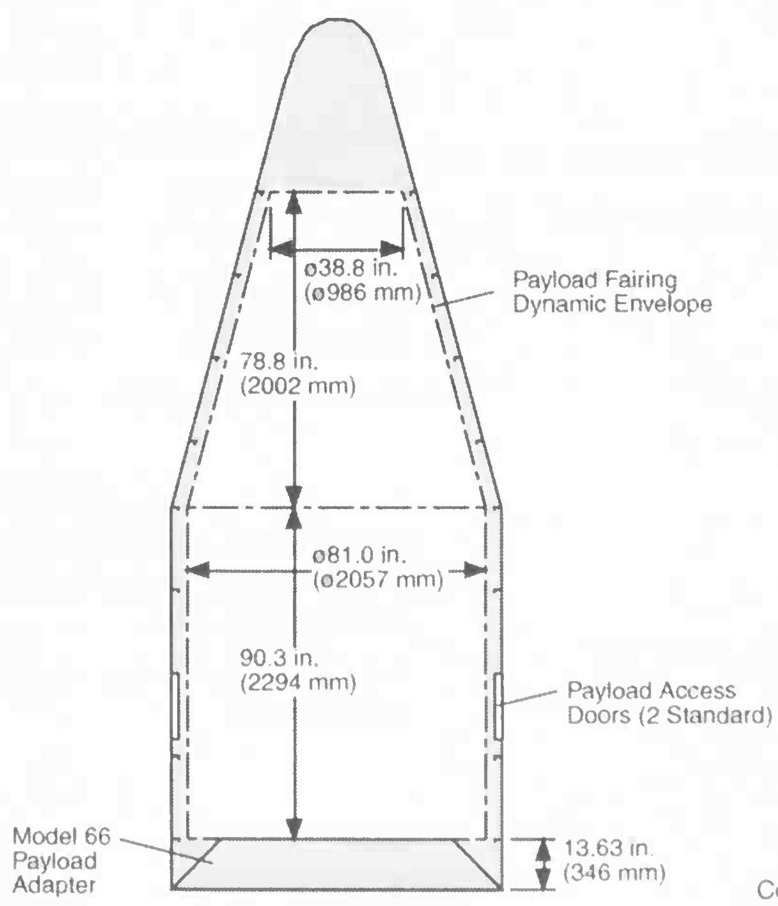
OAM Attitude Control System

Avionics

The Athena avionics are housed in two avionics bays in the OAM. A single Litton LN100 IMU feeds navigational data to the digital flight computer, which controls the vehicle based on flight software output. Commands for attitude control, stage ignition, or separation events are distributed throughout the vehicle by the interlocks package (IP). Data on vehicle performance and environments are collected through a distributed network of instruments and encoders and transmitted over two separate S-band telemetry channels (one analog, one digital). The flight termination system (FTS) has independent batteries and antennas and can terminate thrust on each solid stage by command from the ground or automatically in the event of unplanned stage separation.

Payload Fairing

The standard Athena fairing is the Model 92, so named because its cylindrical section is 92 in. in diameter. The fairing splits into two sections using a zip-tube noncontaminating separation system. The halves are pushed apart with springs and rotate 45 deg on hinges before being released from the rocket. Payload access doors are standard, and RF windows and acoustic or thermal blankets are available options. To speed up launch operations, the payload is mounted onto the launch vehicle adapter and encapsulated into the fairing in an offline processing facility.



Courtesy Lockheed Martin Corporation.

Length	6.1 m (20 ft)
Primary Diameter	2.3 m (7.6 ft)
Mass	535 kg (1180 lbm)
Sections	2
Structure	Ring-stiffened monocoque
Material	Aluminum–lithium

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	2057 mm (81 in.)
Maximum Cylinder Length	2294 mm (90.3 in.) for Model 66 payload adapter
Maximum Cone Length	2002 mm (78.8 in.)
Payload Adapter Interface Diameter	Model 24: 591 mm (23.25 in.) Model 37: 944 mm (37.15 in.) Model 38: 986 mm (38.81 in.) Model 47: 1215 mm (47.84 in.) Model 66: 1676 mm (66.0 in.)
Payload Integration	
Nominal Mission Schedule Begins	6–24 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T−15 s
On-Pad Storage Capability	> 1 year
Last Access to Payload	Depends on launch site, typically T−6 to 11 h
Environment	
Maximum Axial Load	Athena I: 8.1 *g* Athena II: 8.0 *g*
Maximum Lateral Load	Athena I: 1.8 *g* Athena II: 1.8 *g*
Minimum Lateral/Longitudinal Payload Frequency	Athena I: 15 Hz lateral, 30 Hz longitudinal Athena II: 12 Hz lateral, 30 Hz longitudinal
Maximum Acoustic Level	133.2 dB
Overall Sound Pressure Level	Without acoustic blankets: 139.2 dB
Maximum Flight Shock	Dependent on separation system, worst case is 3000 *g* from 2 kHz to 10 kHz
Maximum Dynamic Pressure on Fairing	Athena I: 140 kPa (3000 lbf/ft^2) Athena II: 38 kPa (800 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1BTU/ft^2/s)
Maximum Pressure Change in Fairing	7 kPa/s (1.0 psi/s)
Cleanliness Level in Fairing	Class 5,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±20 km (11 nmi) altitude, ±0.2 deg inclination
Attitude Accuracy (3 sigma)	<1 deg, 2.4 deg/s
Nominal Payload Separation Rate	Mission specific
Deployment Rotation Rate Available	0–3 rpm
Loiter Duration in Orbit	Approx. 1 h
Maneuvers (Thermal/Collision Avoidance)	Yes, including deorbit
Multiple/Auxiliary Payloads	
Multiple Manifest	Athena can carry multiple spacecraft on mission-specific dispensers for LEO constellations. Comanifested payloads will also be considered under certain circumstances. Contact Lockheed Martin Space Systems for details on mission-specific dispensers.
Auxiliary Payloads	Auxiliary payloads may be accommodated depending on available volume and performance.

Production and Launch Operations

Athena is produced by Lockheed Martin, in cooperation with three partner companies, which invested in development of their own products to create Athena. The partners are ATK Thiokol, which produces the Castor 120; the Chemical Systems Division (CSD) of Pratt & Whitney Space Propulsion Operations, which produces the Orbus 21D motors; and Aerojet Redmond Rocket Center, which builds the liquid propulsion system for the OAM.

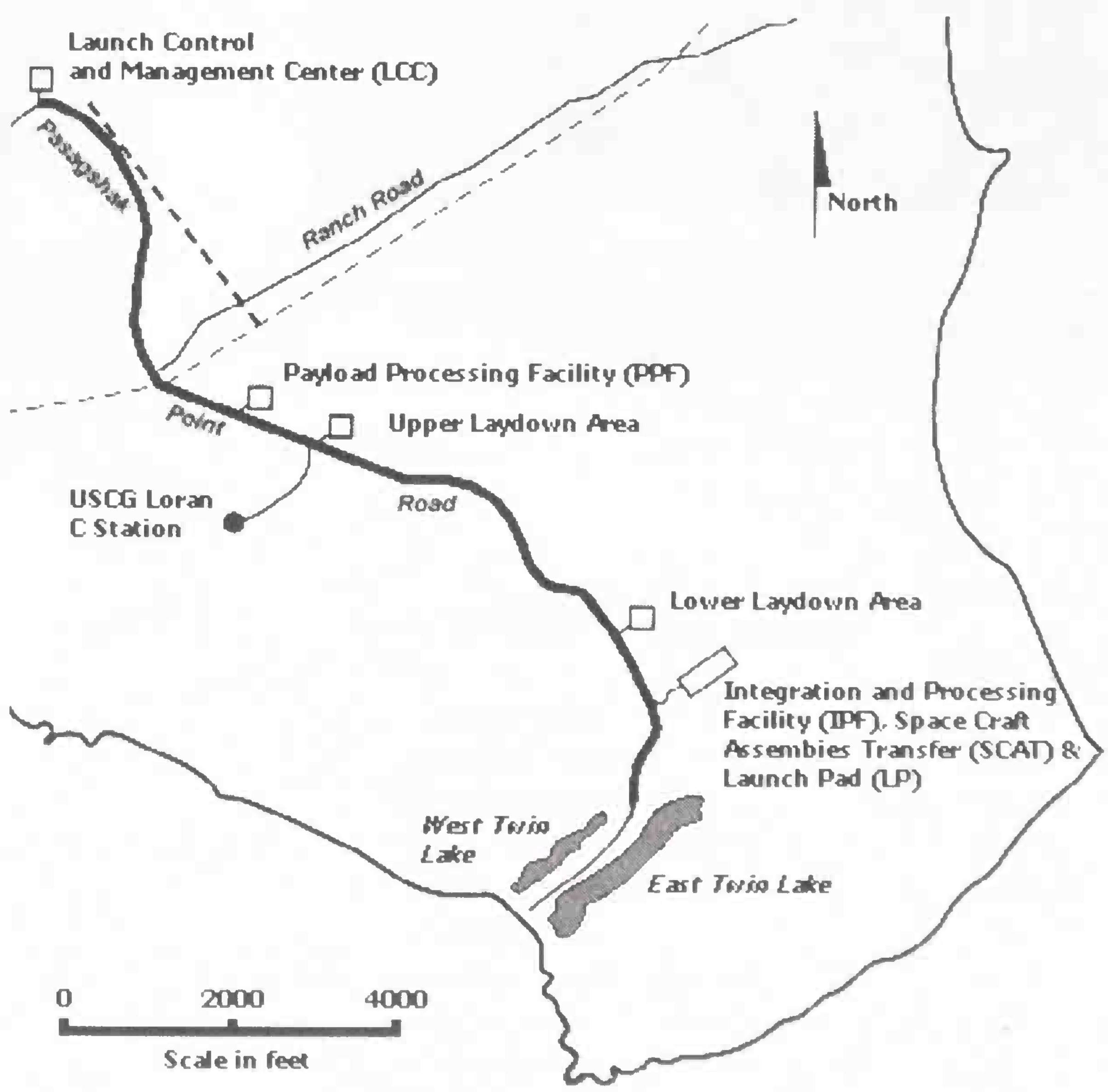
Structural elements such as interstage assemblies, the fairing, and OAM structure are produced at the Lockheed Martin factory in Harlingen, Texas. The OAM structure is shipped to Aerojet for installation of the propulsion system, including the tanks, propellant lines, and thrusters. The OAM is then shipped to Lockheed Martin Space Systems facilities outside Denver, Colorado, where the avionics and electrical systems are installed in the OAM, and the flight software is developed and checked out. The OAM is shipped to the launch site, and all other components are shipped directly from their respective factories to the launch site to build up the launch vehicle on the launch stand.

Organization	Responsibility
Lockheed Martin Space Systems	Prime contractor, mission integration, and production of structural elements
ATK Thiokol	Castor 120 motors
Pratt & Whitney	Orbus 21D and 21G motors
Aerojet	OAM propulsion
Litton	IMU
Lockheed Martin Electronics	Avionics packages

Launch Operations—Kodiak Launch Complex

Polar orbit launches of Athena were initially conducted from SLC-6 at Vandenberg AFB. The massive facility had originally been built in the 1960s for Titan launches of the Manned Orbiting Laboratory project, and later modified for military Space Shuttle flights, but both programs were cancelled before launches took place. Athena's four launches there were the first in the facility's history. At the end of 1999, Lockheed Martin's lease on SLC-6 expired, and the U.S. Air Force transferred the complex to Boeing for the Delta IV program. Lockheed Martin searched for a new launch facility for Athena, and selected the new Kodiak Launch Complex (KLC), on Kodiak Island, Alaska.

Kodiak Launch Complex is operated by the Alaska Aerospace Development Corporation (AADC), a state-owned corporation whose mission is to develop and promote Alaska's aerospace industries. AADC built the launch complex using approximately $40 million in funding from the state of Alaska, NASA, and the U.S. Air Force. The facility is located on the eastern side of Kodiak Island approximately 40km (25 miles) south of the city of Kodiak on 12.5 km² (4.8 mi²) of land at Narrow Cape. The launch facility is nicknamed "The Other Cape," in reference to the more famous Cape Canaveral. While Kodiak Island is remote, it has the largest U.S. Coast Guard station in the United States and is a popular destination for outdoor recreation. It therefore has extensive transportation infrastructure, including seaports and an airport capable of handling large transport aircraft. Commercial airline flights connect Kodiak to Anchorage several times per day.



Courtesy Alaska Aerospace Development Corporation.

Kodiak Launch Complex

KLC includes a payload processing facility (PPF), an integration and processing facility (IPF), a mobile spacecraft assemblies transfer (SCAT) facility, a launch pad, and a launch control center (LCC), each of which were designed to support year-round operations. The PPF has a single high-bay clean room and an airlock to bring spacecraft and equipment in and out. The PPF high bay is equipped for hazardous operations, including spacecraft fueling, and has a 13.6 t (15 U.S. ton) bridge crane. The launch pad consists of a flat concrete apron, a built-in flame duct sized to handle vehicles with 4.9–5.4 MN (1.1–1.2 Mlbf) of thrust, and a rotating service structure. The service structure is a clamshell design that rotates around a fixed tower to enclose the launch vehicle during assembly. The service structure is an all-steel building that is temperature controlled when fully enclosed. It has adjustable work platforms to access the launch vehicle, and a 68 t (75 U.S. ton) bridge crane for stacking the vehicle. Athena is assembled on the launch pad, but other launch vehicles have the option of horizontal assembly and payload integration in the IPF. To move the integrated launch vehicle to the pad, particularly in bad weather, the vehicle is transferred to the SCAT, an enclosed mobile facility on rails that moves between the IPF and the launch pad. Launches are controlled from the LCC, located 3 km (2 mi) from the launch pad.

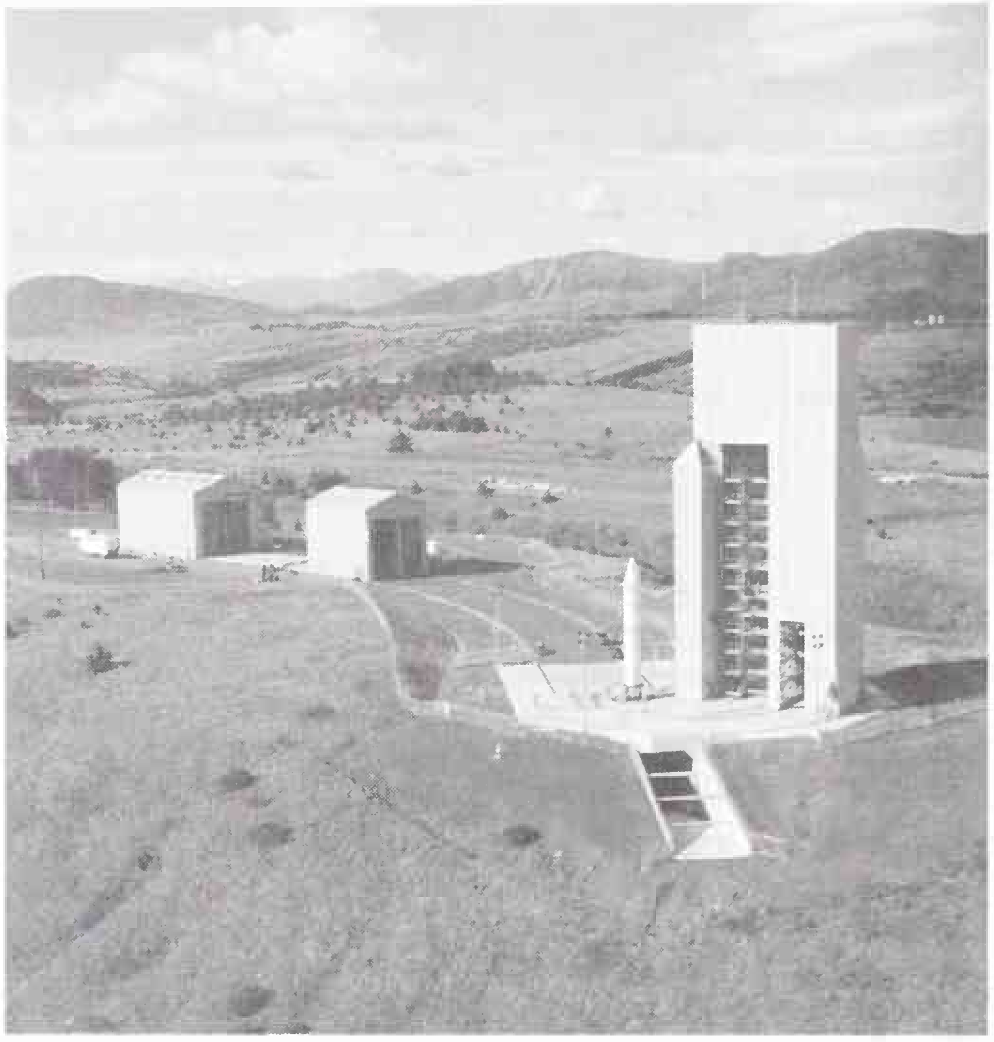
Production and Launch Operations

Launch operations are relatively simple because of the nature of the Athena design. The solid stages and their interstages are stacked on the launch mount using an overhead crane in the launch pad service structure. The OAM is fueled off line and then stacked on top of the assembled stages and electrically connected. Vehicle electrical, hydraulic, and systems are then tested. The spacecraft is mounted onto its payload adapter and encapsulated in the payload fairing at the PPF, then transported to the pad to be stacked on the launch vehicle.

On launch day, the two clamshell halves of the pad service structure are rolled away from the launch vehicle. The countdown for launch begins about 2 h before flight as the vehicle status is checked, the vehicle is powered up, and IMU alignment begins. The flight software is loaded into the onboard computer. The vehicle is monitored and controlled from the LCC. Roughly 15 min before launch, the vehicle switches to internal power. No hold down clamps or bolts are used to keep the vehicle on its launch mount. At T 0 the first-stage solid motor ignites and the vehicle lifts off.

Courtesy Lockheed Martin.

Narrow Cape. The Launch Control Center, with tracking radars and weather station is in the foreground. In the middle distance is the Payload Processing Facility. The launch pad is in the far distance.



Courtesy Lockheed Martin.

Launch Pad 1 at the Kodiak Launch Complex. Left to right: Integration and Processing Facility (IPF), Spacecraft Assemblies Transfer (SCAT) building, launch mount with Athena I launch vehicle, rotating service structure.

Production and Launch Operations

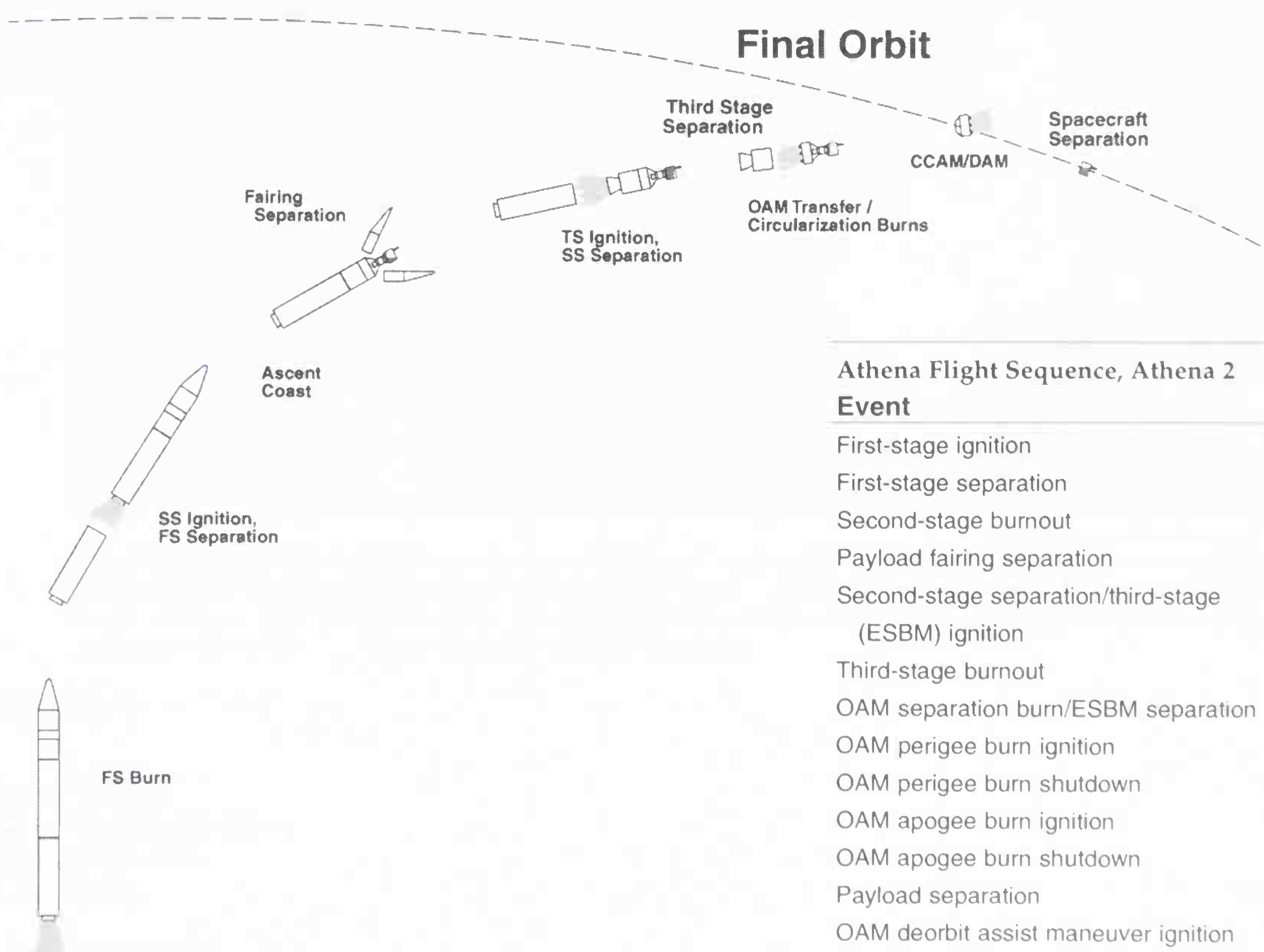
Launch Operations—Cape Canaveral Air Force Station

East coast Athena launches are conducted from SLC-46. SLC-46 was built in 1987 for the flight testing of the U.S. Navy Trident missile that ended in 1989. The facility is no longer used for Navy launches, but it is required to be available to the Navy should a need arise. SLC-46 is now operated by the Florida Space Authority (FSA) as a commercial space launch complex. FSA spent about $8.1 million to make the pad operational, including building a new mobile access structure (MAS) and launch mounts. Funding came from the State of Florida, NASA, the U.S. Air Force, U.S. Navy, U.S. Department of Transportation, and aerospace companies interested in using the facility. SLC-46 is designed to support several different types of small solid launch vehicles (it does not have liquid fueling capabilities), and is available to any commercial launch vehicle for a fee of about $250,000. The MAS is a mobile servicing tower with adjustable work platforms to access the launch vehicle. However, it lacks a crane, so Athena must be stacked using a rented mobile crane. Spacecraft processing can be conducted at Astrotech's commercial payload processing facilities.

Courtesy Spaceport Florida.

SLC-46 Launch Site

Flight Sequence



Athena Mission Profile

Athena Flight Sequence, Athena 2

Event	Time, s
First-stage ignition	0
First-stage separation	84.88
Second-stage burnout	173.66
Payload fairing separation	267.19
Second-stage separation/third-stage (ESBM) ignition	272.31
Third-stage burnout	423.61
OAM separation burn/ESBM separation	429.11
OAM perigee burn ignition	429.11
OAM perigee burn shutdown	591.23
OAM apogee burn ignition	3087.47
OAM apogee burn shutdown	3473.57
Payload separation	3487.69
OAM deorbit assist maneuver ignition	3788.19
OAM deorbit assist maneuver shutdown	4046.60

Courtesy Lockheed Martin Corporation.

Vehicle History

Historical Summary

Development of the Lockheed Launch Vehicle (LLV) family was announced by Lockheed Missiles and Space Company in May 1993. Lockheed had designed and produced fleet ballistics missiles for the U.S. Navy for decades. It wanted to use its experience with solid rockets in the commercial space launch services market. LLV was to be developed entirely with commercial funding, making it the first space launch vehicle to be developed without government funding or up front government launch contracts. Lockheed was supported in part by its teammates for the project: Thiokol, Pratt & Whitney, and Primex. No customers had committed to buy launch services, but Lockheed hoped to find a customer and fly the demonstration launch vehicle (DLV) by late 1994.

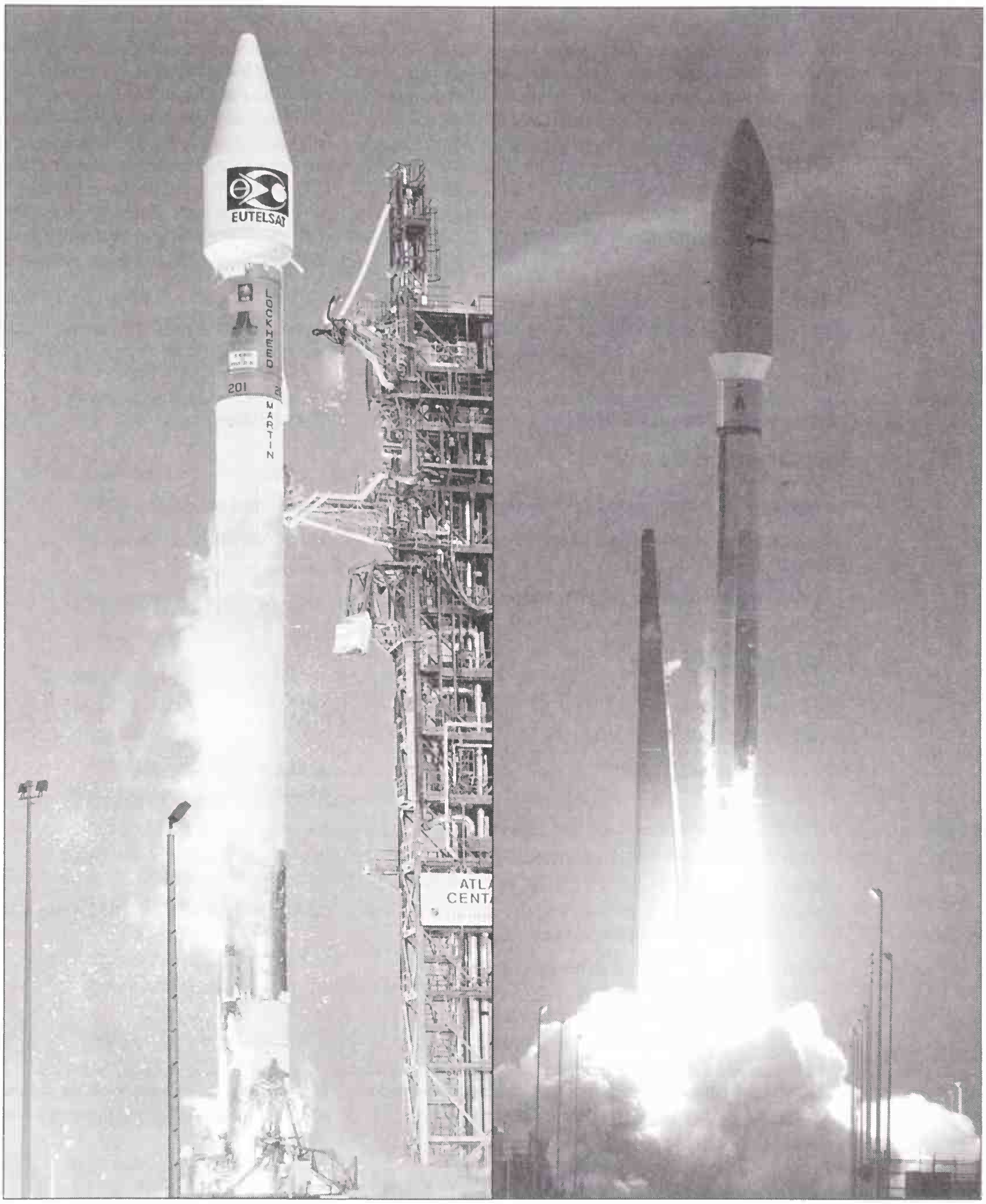
The launch of DLV, an Athena I model, occurred in August 1995. It carried Gemstar, a small commercial satellite to demonstrate LEO communications. Shortly after launch, flammable hydraulic fluid expended from the nozzle actuators was caught up in the airstream and began to recirculate around the base of the launch vehicle. The fluid burned, overheating the control cables and causing the first-stage TVC system to fail. Shortly thereafter, the IMU power supply arced, causing the IMU to lose its alignment. As a result, DLV was destroyed by the range safety officer. The summer and fall of 1995 was a bad time for the new small commercial launch vehicles. The second flight of the Pegasus XL had failed in June, and the Conestoga, a small launch vehicle built by EER Systems, failed in its maiden flight on 23 October.

Fortunately, Athena had accumulated a small backlog of launch orders by the time of the DLV failure, and this gave the program enough momentum to overcome the failure. The backlog included the Lewis and Clark satellites for NASA, the Lunar Prospector spacecraft, a pair of Ikonos commercial remote sensing satellites, and the ROCSAT satellite for Taiwan. Lockheed Martin spent two years investigating the failure, examining the rest of the launch vehicle for other failure modes, and modifying the design. As a result of the merger of Lockheed and Martin Marietta, the program also moved from Lockheed facilities in Sunnyvale, California, to Denver, Colorado, where the rest of Lockheed Martin's space launch vehicle business is based. Athena I successfully returned to flight in August 1997 carrying NASA's Lewis satellite. This was followed in January 1998 by the first launch of an Athena II carrying the Lunar Prospector spacecraft.

In September 2001, Athena performed the first orbital launch from Kodiak, Alaska. The mission, designated Kodiak Star, was conducted on behalf of NASA and deployed four small satellites belonging to NASA and the U.S. Air Force into two different orbits.

Originally planned to launch up to 12 flights per year, the Athena program has averaged only about one launch annually because the boom in small satellites anticipated during the 1990s did not materialize. The difficulties that Athena and other small launch systems have had in finding customers are reflected in the history of the Athena II "white tail." In 1996 the U.S. Air Force selected an Athena II to launch the demonstration satellite SBIRS-LADS. The spacecraft program was cancelled in early 1999, leaving a nearly complete rocket without a customer—what aircraft builders call a white tail. Later, the white tail was selected to launch a commercial imaging satellite, but the potential customer was unable to raise sufficient funding to sign a firm contract. In 2000, the vehicle was ordered by Blastoff!, a start-up company planning a commercial lunar landing mission. The company went out of business when internet and technology stocks crashed later that year, wiping out its sponsors' funding. Marketing efforts continued with several other potential customers, without success. The last launch in Athena's backlog, the Kodiak Star mission, was completed in late 2001. With no additional orders on the books, the program went into hibernation leaving only a small staff to continue program management and marketing efforts.

ATLAS



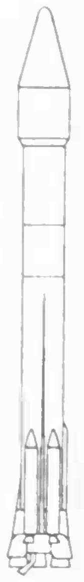
Courtesy International Launch Services.

The Atlas III (left) and V (right) are the newest developments in the Atlas family. Atlas III is a commercially developed upgrade that serves as a stepping stone to the Atlas V, developed as part of the Air Force's Evolved Expendable Launch Vehicle Program. The first Atlas IIIA and the inaugural flight of the Atlas V-500 configuration carrying the Rainbow-1 direct broadcast satellite are shown.

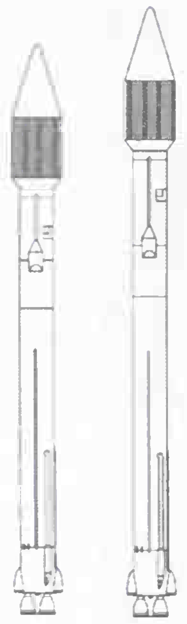
Contact Information

Marketing and Sales:
International Launch Services
1660 International Drive
McLean, VA 22102
USA
Phone: +1 (571) 633-7400
Fax: +1 (571) 633-7500
Web site: www.ilslaunch.com

General Description



Atlas IIAS



Atlas IIIA/IIIB

	Atlas IIAS	Atlas IIIA and IIIB

Summary

	Atlas IIAS	Atlas IIIA and IIIB
	The Atlas has a long history as an early U.S. ICBM and space launch vehicle. Since the 1960s Atlas has incorporated the Centaur upper stage. The primary versions in use in the 1990s, the Atlas II series, have been reliable commercial launch vehicles. The last remaining variant in this series, the Atlas IIAS, is being phased out in favor of Atlas V. The Atlas IIAS is similar to the Atlas IIA, with the addition of solid strap-on boosters. It is used to carry medium-sized commercial and government satellites to GTO.	Lockheed Martin developed the new Atlas III series commercically to provide higher performance and reliability at a lower cost than the Atlas IIAS. This is accomplished primarily by simplifying and improving propulsion systems. The characteristic stage-and-a-half main engine and the strap-on boosters are eliminated through the use of the new, high-performance RD-180 engine from Russia. The Atlas IIIA uses a new single-engine Centaur. Eliminating one engine improves cost and reliability. For higher performance, the Atlas IIIB uses a stretched Centaur that can use either one of two engines.

Status

	Atlas IIAS	Atlas IIIA and IIIB
	Operational. First launch in 1993. To be retired in 2004.	Operational. Atlas IIIA first launch in 2000. Atlas IIIB first launch in 2002. To be retired in 2005.

Origin

	Atlas IIAS	Atlas IIIA and IIIB
	United States	United States

Key Organizations

	Atlas IIAS	Atlas IIIA and IIIB
Marketing	International Launch Services	International Launch Services
Launch Service Provider	International Launch Services	International Launch Services
Prime Contractor	Lockheed Martin Space Systems	Lockheed Martin Space Systems

Primary Missions

	Atlas IIAS	Atlas IIIA and IIIB
	GTO	GTO

Estimated Launch Price

	Atlas IIAS	Atlas IIIA and IIIB
	Atlas IIAS: price negotiable	Atlas IIIA: price negotiable Atlas IIIB: price negotiable

Spaceports

	Atlas IIAS	Atlas IIIA and IIIB
Launch Site	Cape Canaveral AFS, SLC-36A and B	Cape Canaveral AFS, SLC-36B
Location	28.5° N, 81.0° W	28.5° N, 81.0° W
Available Inclinations	28.5–55 deg directly, up to 63.4 deg with plane change	28.5–55 deg directly, up to 63.4 deg with plane change
Launch Site	Vandenberg AFB, SLC-3E	
Location	34.7° N, 120.6° W	
Available Inclinations	63–120 deg	

Performance Summary

	Atlas IIAS	Atlas IIIA and IIIB
	Quoted mass is payload systems weight capability, which includes spacecraft mass plus payload adapter, harnesses, etc. Large Payload Fairing (LPF) is assumed.	Quoted mass is payload systems weight capability. Payloads above 9072 kg (20,000 lbm) may require mission-unique accommodations.
185 km (100 nmi), 28.5 deg	8618 kg (19,000 lbm)	Atlas IIIA: 8640 kg (19,048 lbm) Atlas IIIB: 10,759 kg (23,720 lbm)
185 km (100 nmi), 90 deg	7212 kg (15,900 lbm)	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	7225 kg (15,929 lbm)	Atlas IIIA: 7756 kg (17,099 lbm) Atlas IIIB: 9239 kg (20,369 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	?	No capability
GTO: 167×35,786 km (90×19,323 nmi), 27.0 deg	3719 kg (8200 lbm)	Atlas IIIA: 4037g (8900 lbm) Atlas IIIB: 4119 kg (9081 lbm)
Geostationary Orbit	No Capability	No Capability

Flight Record (through 31 December 2003)

	Atlas IIAS	Atlas IIIA and IIIB
Total Orbital Flights	26	4
Launch Vehicle Successes	26	4
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	0	0

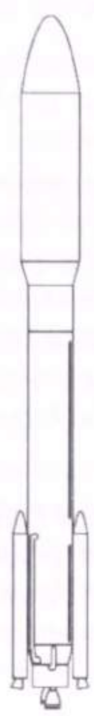
Flight Rate

	Atlas IIAS	Atlas IIIA and IIIB
	0–4 per year	0–1 per year

General Description

Atlas V

Atlas V 400



Atlas V 500

Summary

The Atlas V series of launch vehicles was developed with U.S. government support to meet new demands for government and commercial launch services. The Atlas V uses the RD-180 first stage engine and the Centaur design from the Atlas III, in conjunction with a new first-stage design called the common core booster (CCB). Two basic varieties of the Atlas V are available. The Atlas V 400 version use the same 4-m diameter class fairing as the Atlas II and III series, and can have up to three solid strap-on boosters for increased performance. The Atlas V 500 versions have a new 5-m diameter payload fairing, and can have up to 5 strap-on boosters.

Status

400 Series: Operational. First launch in 2002.
500 Series: Operational. First launch in 2003.

Origin

United States

Key Organizations

Marketing Organization	International Launch Services (Commercial), Lockheed Martin Commercial Launch Services (U.S. Government)
Launch Service Provider	ILS
Prime Contractor	Lockheed Martin Space Systems

Primary Missions

Medium and heavy spacecraft to GTO and GEO

Estimated Launch Price

Prices negotiable

Spaceports

Launch Site	Cape Canaveral AFS, SLC-41
Location	28.5° N, 81.0° W
Available Inclinations	28.5–55 deg

Performance Summary

The largest and smallest configurations are shown below. See the Performance section for a complete table. Performance shown is Payload Systems Weight Capability, and includes the mass of the spacecraft and adapter. Payloads above 9072 kg (20,000 lbm) may require mission-unique accommodations.

185 km (100 nmi), 28.5 deg	Atlas V 402: 12,500 kg (27,558 lbm) Atlas V 552: 20,520 kg (45,239 lbm)
185 km (100 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	Atlas V 402: 10,330 kg (22,774 lbm) Atlas V 552: 17,590 kg (38,779 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	No capability
GTO: 167×35,786 km (90×19,323 nmi), 27.0 deg	Atlas V 401: 4950 kg (10,913 lbm) Atlas V 551: 8670 kg (19,114 lbm)
Geostationary Orbit	Atlas V 401: No capability Atlas V 551: 3810 kg (8400 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	3
Launch Vehicle Successes	3
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

Dependent on market demand

Nomenclature

When Atlas began offering commercial launch services in the late 1980s, the old letter-based naming system (e.g. Atlas E, F, G, and H) was replaced with Roman numerals, leading to the Atlas I, II, and III families. Different variations in each family are denoted by letters, such as the Atlas IIA, and IIAS, where S indicates solid strap-on boosters, for example. Initially the follow-on to Atlas IIAS was to be designated the Atlas IIAR (R standing for re-engined). In 1998 Lockheed Martin announced that the Atlas IIAR would instead be called the Atlas IIIA, and that a larger Atlas IIIB also would be developed along the same lines. There is no vehicle designated Atlas IV. Instead, Lockheed Martin chose to go directly from Atlas III to Atlas V for its future vehicle development, explaining that the new vehicle family is such a significant improvement that it deserves a designation that emphasizes how much more advanced it is than previous versions.

Standard Atlas V versions are not distinguished by letter codes, but rather by a three-digit number. The Atlas V HLV (heavy-lift vehicle) is a proposed larger configuration that does not currently use the numbering system.

Example **Atlas V xyz**

First digit (x)

Approximate payload fairing diameter in meters (4 or 5)

Second digit (y)

Number of strap-on boosters (0–5)

Third digit (z)

Number of engines on Centaur upper stage

0 = Unspecified

1 = Single-engine Centaur (SEC)

2 = Dual-engine Centaur (DEC)

Cost

ILS has not released official Atlas prices, but prices for one signed contract and one offer have been disclosed by customers in documentation provided to the U.S. government. In a 1998 filing with the U.S. Federal Communications Commission, Comsat Corporation, the U.S. signatory to Intelsat, disclosed that it had paid $92 million for the Atlas IIAS launch of Intelsat 806 performed that year. In 1998 the company CD Radio stated in documentation provided to the U.S. Securities Exchange Commision that it had been offered launch prices of $90 million for Atlas IIIA and $95 million for Atlas IIIB for a multiple launch contract with launches beginning around 2000. Announced contract values for U.S. government launch services on Atlas IIAS have ranged from $87.6 to $91.1 million, although this may not reflect the total price because of the structure of the contracts. In 1999, the Air Force awarded a $70.7 million contract for an Atlas IIIB launch scheduled for 2002. The Air Force says it expects to pay roughly $75 million for medium-class spacecraft launches on Atlas V, and $110 million for launches of intermediate-class spacecraft.

Atlas is a predominantly commercial launch system, and therefore development costs for the Atlas IIA, IIAS, IIIA, and IIIB have been paid for primarily by Lockheed Martin. Commercialization of Atlas, beginning with Atlas I, was started with a $400 million investment in 1987 when the Atlas program was still part of General Dynamics. In 1995, Lockheed Martin announced that it was investing $300 million in its launch vehicle buisiness, most of which likely went toward the development of Atlas IIIA and B. Development costs of the Atlas V are being split between the U.S. Air Force and Lockheed Martin. In October 1998 the U.S. Air Force awarded $500 million in "Other Transaction Agreements" to Lockheed Martin as its contribution for Atlas V development. In addition, Lockheed Martin was awarded a $694 million contract for nine Atlas V launches.

Availability

The last Atlas IIA flew in late 2002, and the IIA variant is now retired. Production of the Atlas IIAS has ceased, and the IIAS variant with be retired. The last Atlas IIAS launch is planned for 30 June 2004. A total of eight Atlas III series rockets were built, all of which have been sold. The last Atlas III flight is planned for January 2005 and the III variant considered a bridge to the Atlas V will be retired. Atlas V became operational with a first launch in August 2002.

Performance

Atlas performance is quoted in terms of payload systems weight capability (PSWC), which includes not only the separated spacecraft mass, but also other payload-related components such as the payload adapter, mission-unique wiring harnesses, and, if necessary, a payload flight termination system. To determine the available spacecraft mass, the mass of the mission specific adapter must be subtracted from the PSWC. The adapter mass ranges from 45 to 129 kg (99 to 284 lbm) for typical payloads, depending on the spacecraft mass and type of interface.

The GTO mission is the standard mission design for communications satellite launches. Trajectories for GTO missions can be customized in several ways. Atlas frequently flies missions with supersynchronous and subsynchronous transfer orbits or lower transfer orbit inclinations. This can help satellites that are lighter or heavier than the nominal capability to a standard GTO to reach GEO with more propellant remaining, thereby providing an increased lifetime for the satellite. For precise orbit injection, Atlas offers a guidance commanded shutdown (GCS) option in which Centaur propellants are reserved to cover variations in vehicle performance. This ensures that in off-nominal cases, the Centaur will still achieve the targeted orbit. This reserve typically protects for 99% or greater probability (2.33 sigma) of achieving a GCS at the target orbit. Atlas also offers performance enhancement options such as in-flight retargeting (IFR), minimum residual shutdown (MRS), and an IFR/MRS combination. With IFR, the target orbit is updated in-flight to account for performance variations of the booster stage. Centaur can be retargeted to a variable transfer orbit inclination, apogee, perigee, argument of perigee, or any combination of the above. With MRS, the Centaur propellants are burned to minimum residual levels for a significant increase in nominal performance capability. When burning to minimum residuals, the flight propellant reserves (FPR) held for GCS options are eliminated to gain nominally additional delta-velocity from the Centaur. It is practical to implement the MRS option when a satellite has a liquid propulsion system that is capable of correcting for variations in launch vehicle performance. MRS is not an option for satellites using solid propellant (fixed-impulse) propulsion, because FPR propellants are required to ensure that the Centaur injection conditions will match the capability of the fixed-impulse stage. When Centaur burns all propellants to minimum residuals, the liquid propellant satellite corrects for the effects of launch vehicle dispersions. These dispersions primarily affect apogee altitude. Variations in other transfer orbit parameters are minor.

The new Atlas V 500 series and the Atlas V HLV will be able to deliver payloads directly to GEO. This is achieved with modifications to the Centaur that allow it to coast for 6 h in GTO before providing an additional injection burn into GEO at apogee. The Atlas V will generally use a single-engine Centaur (SEC) for GTO and other high-energy orbits.

LEO trajectories can be implemented in two ways. In the single-burn, or direct ascent, trajectory only one Centaur burn is used to deliver the satellite directly to its final orbit. This is effective for low orbits. For higher altitudes, it is more effective for the first burn to place the Centaur and payload on an elliptical transfer orbit, and then perform a short circularization burn at apogee. This requires an extended mission/short burn kit on the Centaur stage. The shortest burn that can be performed by the second stage RL10 engines is approximately 10 s. Therefore, the two-burn trajectory cannot be used for very low orbits, as the required circularization burn would be too short. From Cape Canaveral AFS, inclinations between 28.5 deg and 55 deg can be achieved with a direct ascent trajectory. A two-burn trajectory can achieve inclinations as high as 63.4 deg without overflying the east coast of the United States by yaw steering during the second burn to increase inclination. The performance penalties associated with this make Vandenberg AFB a more suitable launch site for high-inclination missions, and some Atlas IIAS launches have been conducted from Vandenberg to reach sun-synchronous or polar orbits. There is no pad at VAFB for the Atlas III. Lockheed Martin did not initially build a pad at VAFB for the Atlas V, but has commenced construction of SLC-3E, expected to be completed by the end of 2005.

For some heavy LEO payloads the Centaur stage requires strengthened structural interfaces, because it is designed primarily for lighter GTO spacecraft. A strengthened equipment module is available for payloads between 4500-6350 kg (9950-14000 lbm). Heavier payloads would require a new truss adapter and additional modifications to the Centaur. Heavy LEO payloads will make use of a dual engine Centaur (DEC) to provide additional thrust for orbital insertion.

Performance

Performance shown below includes the following assumptions:

Performance shown is Payload Systems Weight Capability (PSWC), which includes both the spacecraft and adapter mass. PSWC above 9,072 kg (20,000 lbm) may require mission-unique accommodations.

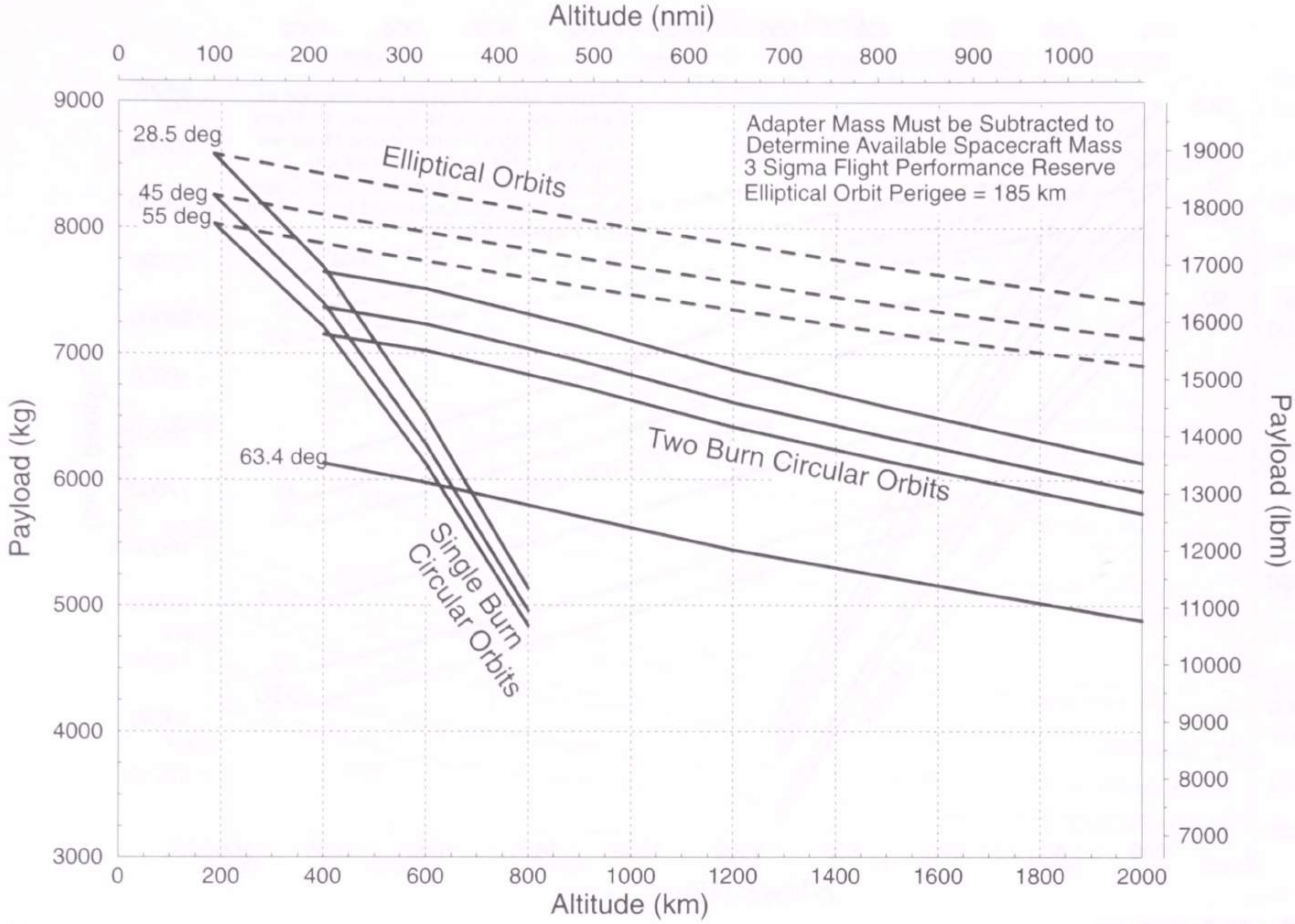
Probability of GCS is 99.87% (3 sigma) for LEO missions, 99% for GTO missions.

Atlas V uses SEC for GTO and GEO and a DEC for LEO.

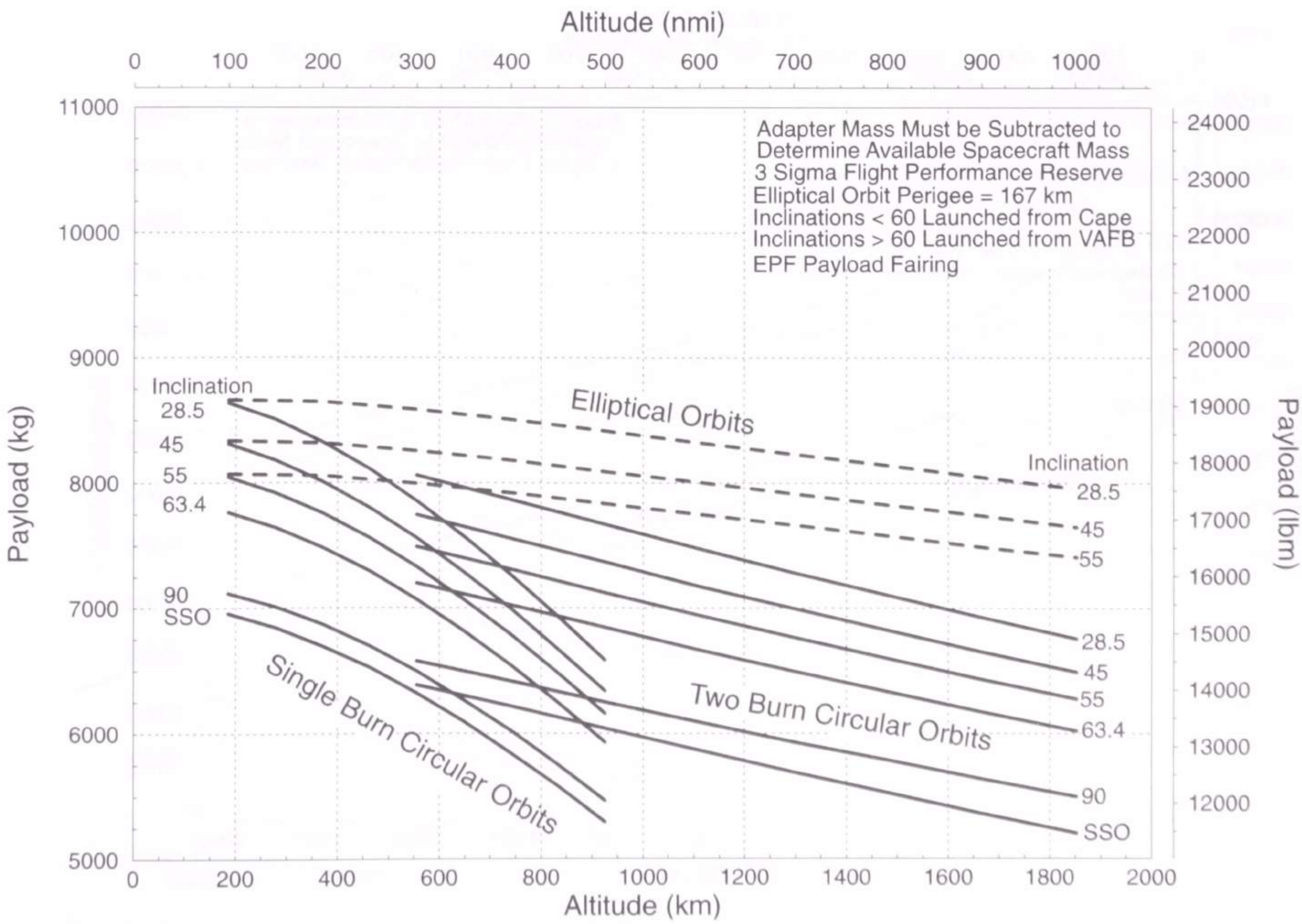
Launches to inclinations at or below 63.4 deg are conducted from CCAFS. Missions to polar or sun-synchronous orbits will be conducted at VAFB in the future (No capability presently for Atlas III and V).

Vehicle	Payload Fairing	185 km (100 nmi), 28.5 deg	185 km (100 nmi), 90 deg	Space Station Orbit: 407 km (220 nmi), 51.6 deg	SSO: 800 km (432 nmi), 98.6 deg	GTO: 167×35,788 km (90×19,324 nmi), 27.0 deg	GEO
IIAS	LPF	8618 kg (19,000 lbm)	7212 kg (15,900 lbm)	7225 kg (15,928 lbm)	?	3719 kg (8200 lbm)	No capability
IIIA	EPF	8640 kg (19,048 lbm)	No capability	7756 kg (17,099 lbm)	?	4037 kg (8900 lbm)	No capability
IIIB	LPF	10,759 kg (23,720 lbm)	No capability	9239 kg (20,369 lbm)	?	4119 kg (9081 lbm)	No capability
V 40Z	EPF	12,500 kg (27,558 lbm)	No capability	10,330 kg (22,774 lbm)	?	4950 kg (10,913 lbm)	No capability
V 41Z	EPF	?	No capability	?	?	5950 kg (13,118 lbm)	No capability
V 42Z	EPF	?	No capability	?	?	6830 kg (15,058 lbm)	No capability
V 43Z	EPF	?	No capability	?	?	7640 kg (16,843 lbm)	No capability
V 50Z	5 m Short	10,300 kg (22,708 lbm)	No capability	8730 kg (19,246 lbm)	?	3970 kg (8752 lbm)	No capability
V 51Z	5 m Short	12,590 kg (27,756 lbm)	No capability	10,710 kg (23,612 lbm)	?	5270 kg (11,618 lbm)	No capability
V 52Z	5 m Short	15,080 kg (33,246 lbm)	No capability	12,880 kg (28,396 lbm)	?	6285 kg (13,856 lbm)	2680 kg (5908 lbm)
V 53Z	5 m Short	17,250 kg (38,030 lbm)	No capability	14,750 kg (32,518 lbm)	?	7200 kg (15,873 lbm)	3190 kg (7033 lbm)
V 54Z	5 m Short	18,955 kg (41,789 lbm)	No capability	16,230 kg (35,781 lbm)	?	7980 kg (17,593 lbm)	3540 kg (7804 lbm)
V 55Z	5 m Short	20,520 kg (45,239 lbm)	No capability	17,590 kg (38,779 lbm)	?	8670 kg (19,114 lbm)	3810 kg (8400 lbm)
V Heavy	5 m Short	28,800 kg (63,492 lbm)	No capability	26,900 kg (59,304 lbm)	?	13,154 kg 29,000 (lbm)	6350 kg (14,000 lbm)

Performance

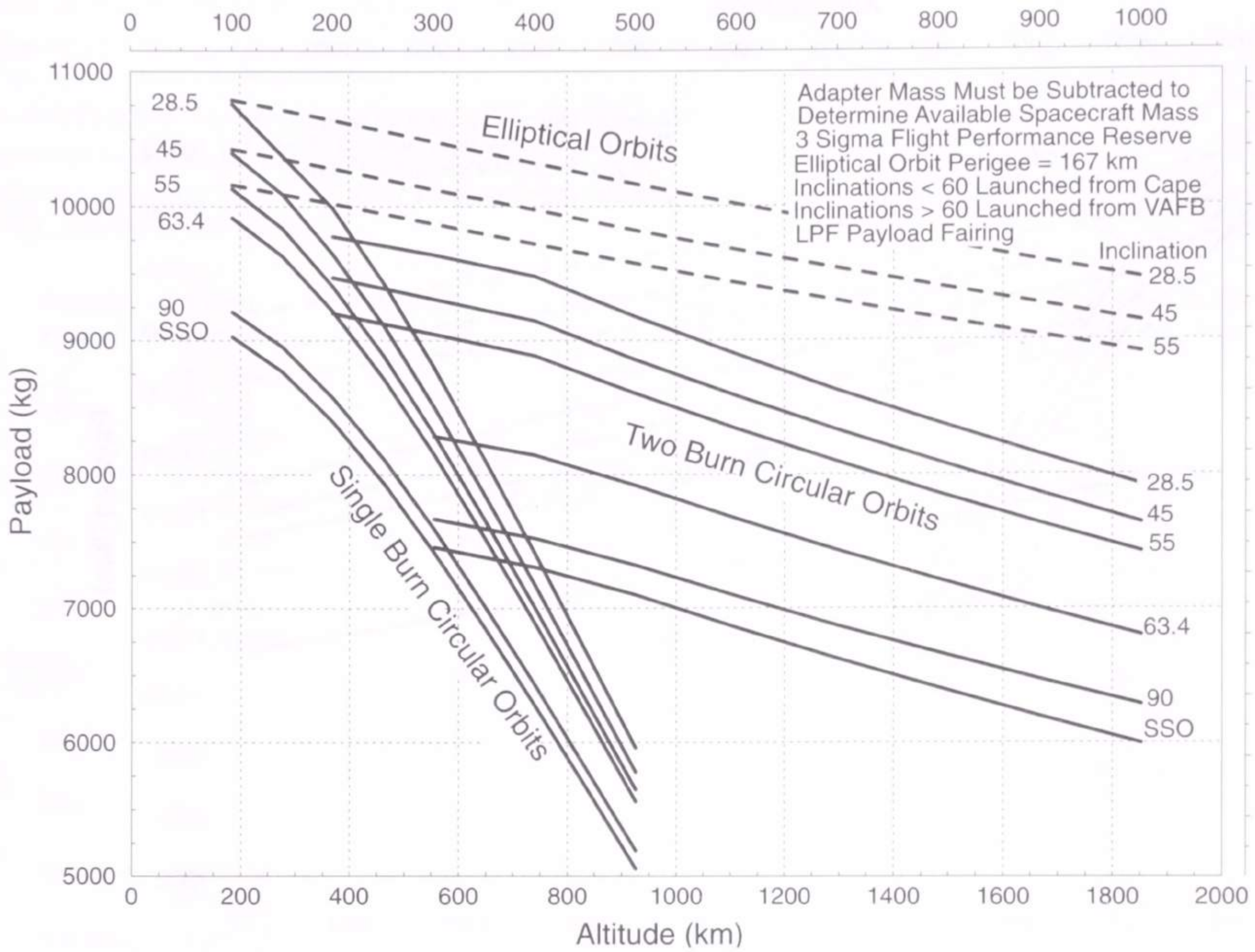


Atlas IIAS: LEO Performance from Cape Canaveral

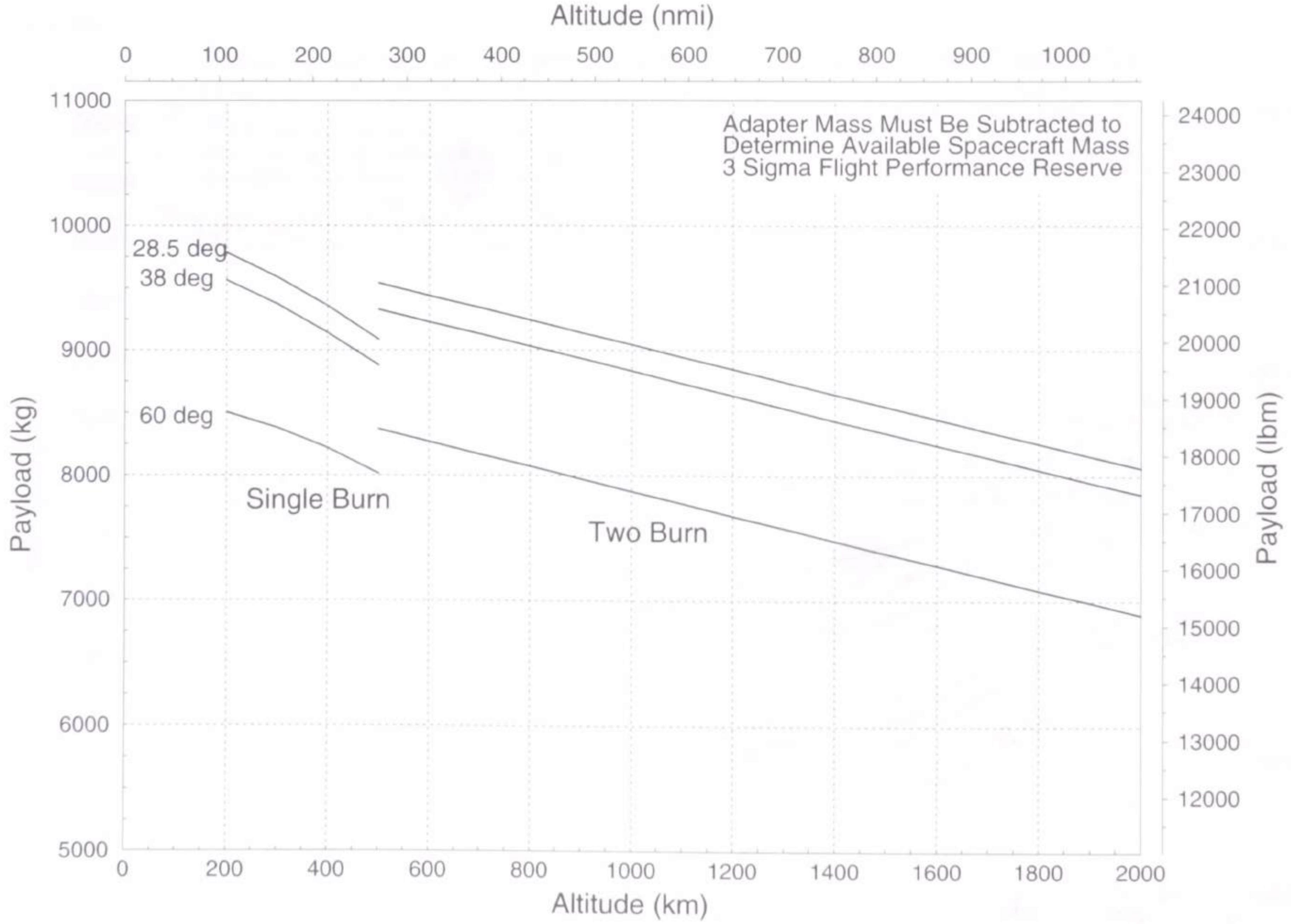


Atlas IIIA: LEO Performance

Performance

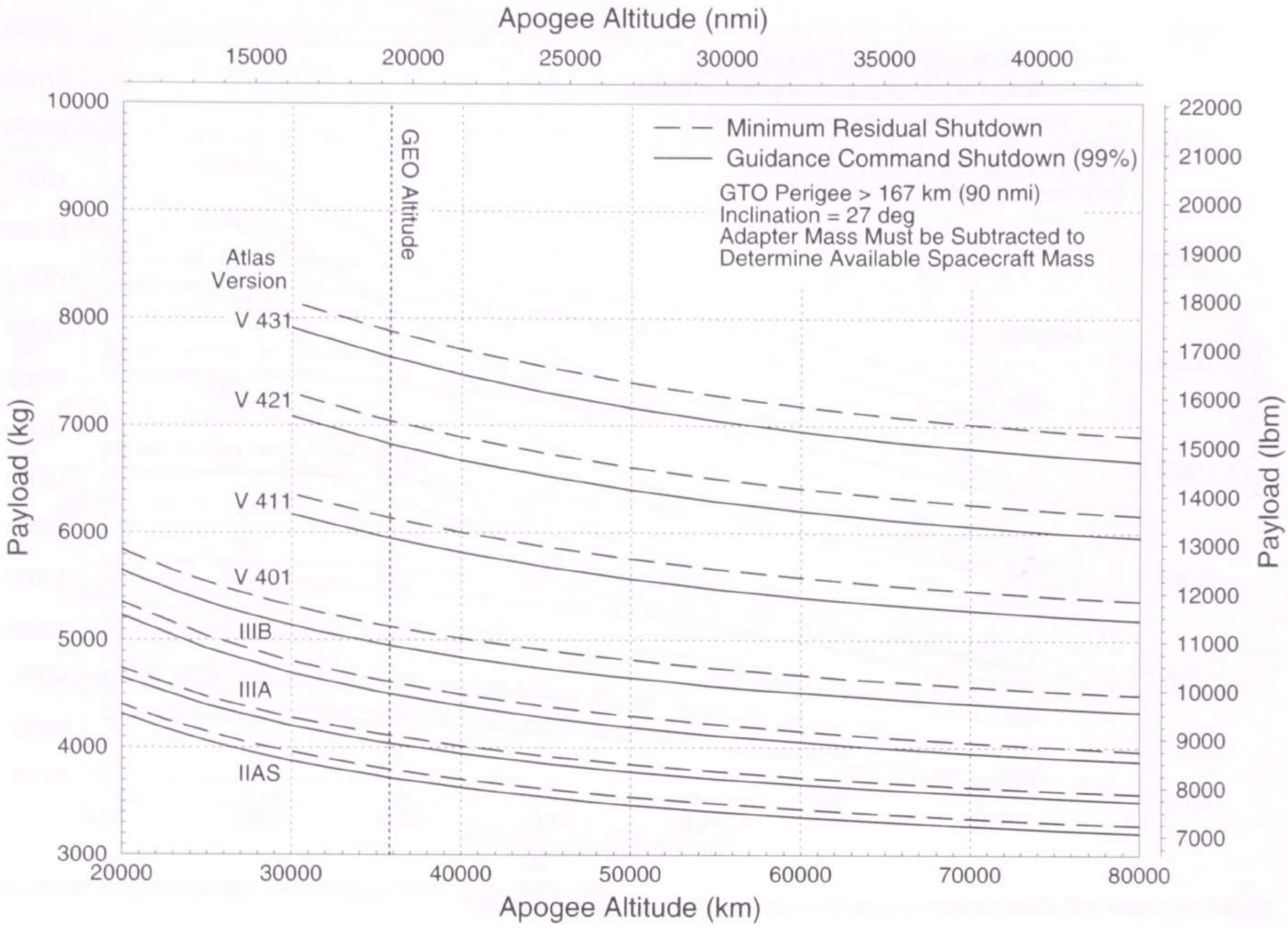


Atlas IIIB: LEO Performance

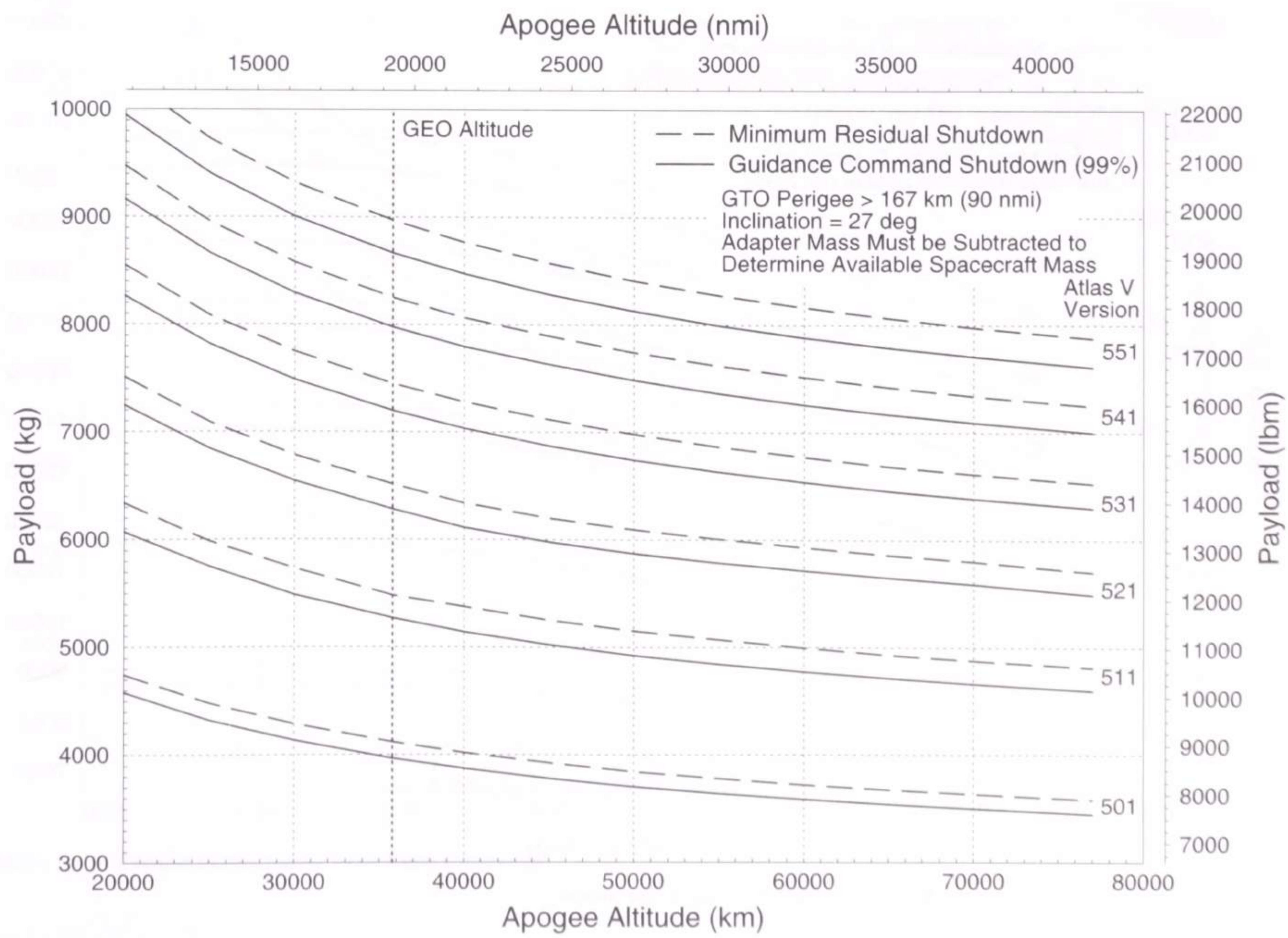


Atlas V 401: LEO performance

Performance

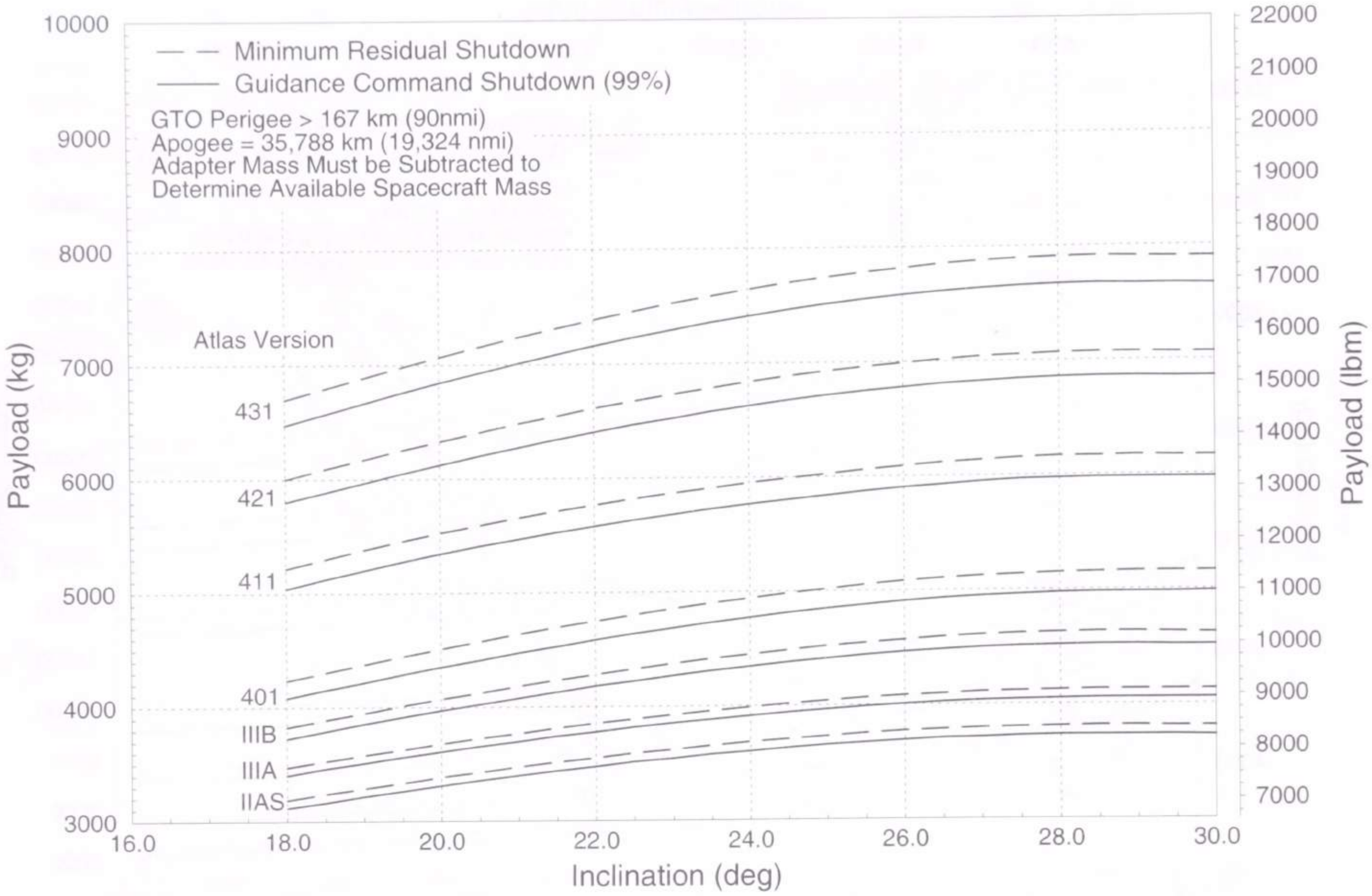


Atlas II, III, and V 400: Performance to GTO from Cape Canaveral

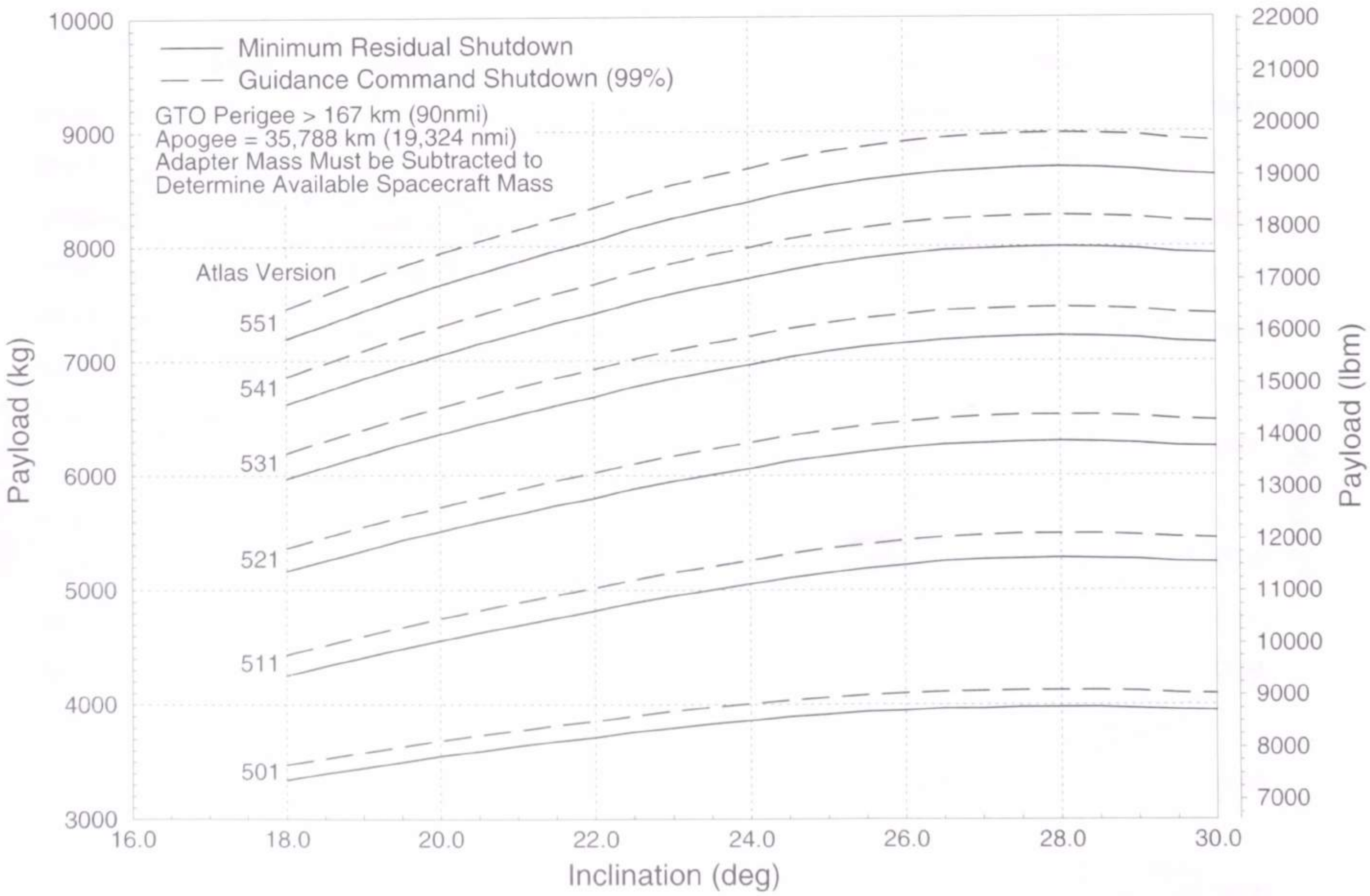


Atlas V 500: Performance to GTO from Cape Canaveral

Performance

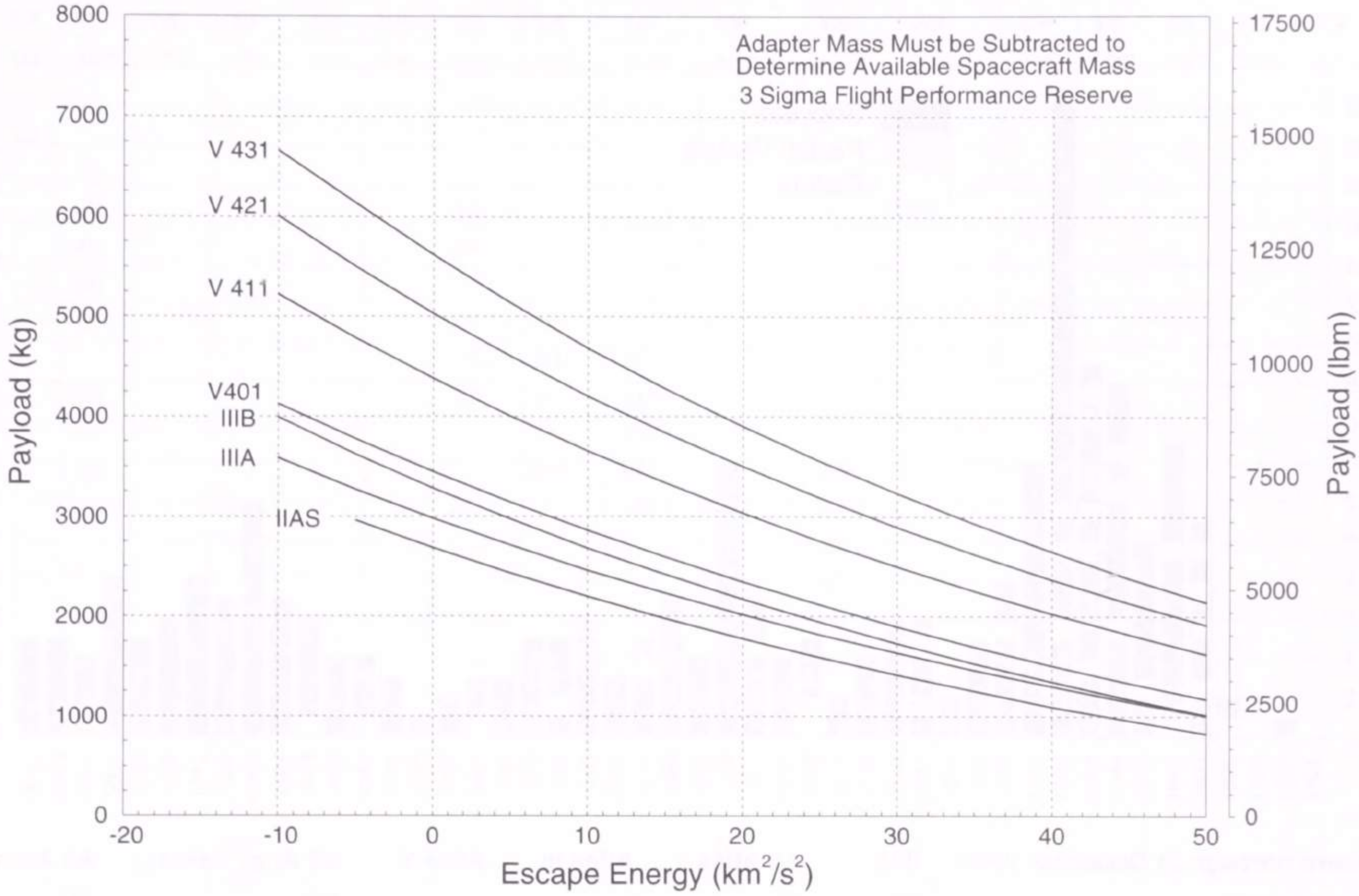


Atlas II, III, and V 400: Performance to Low Inclination GTO from Cape Canaveral

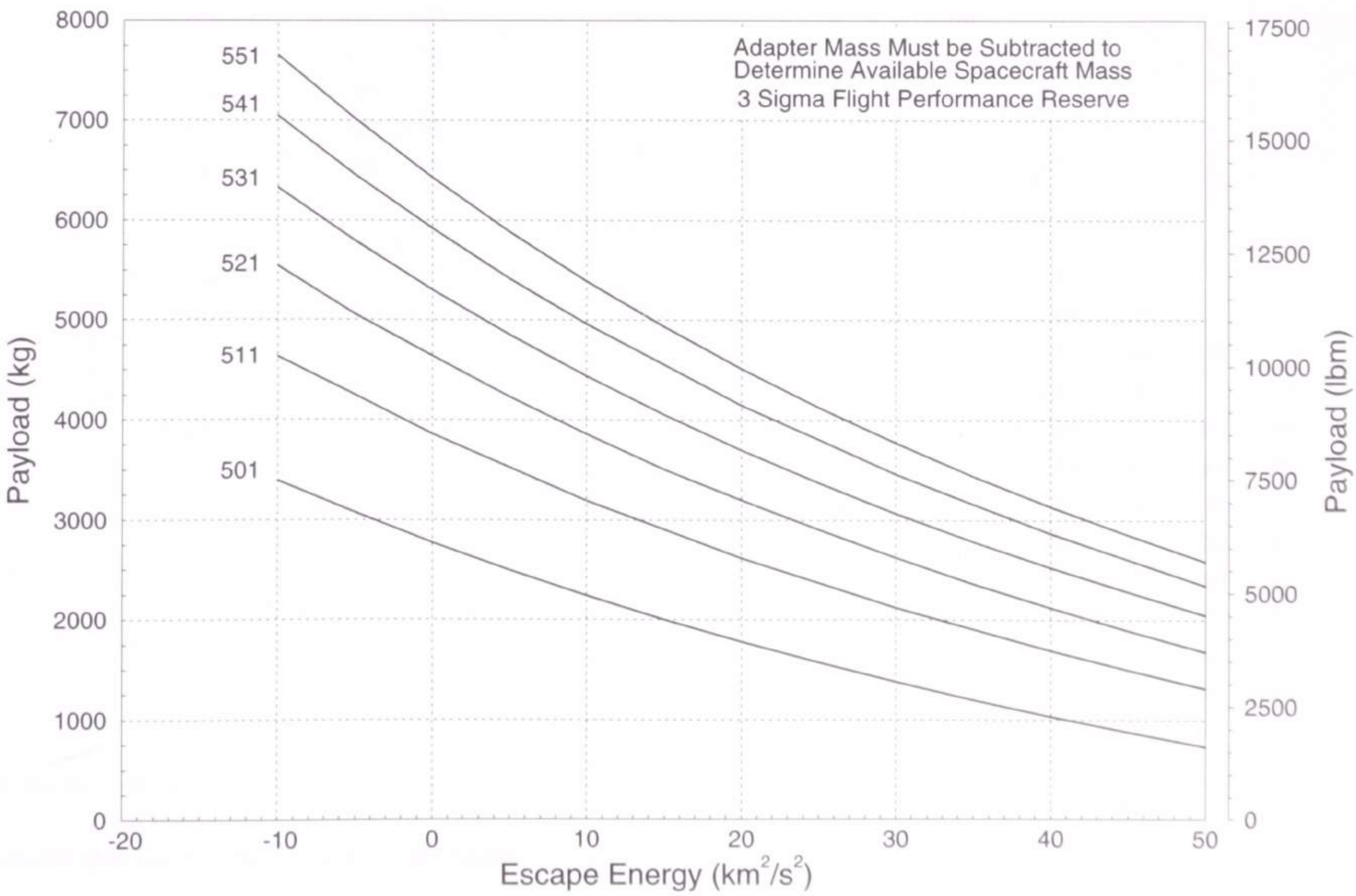


Atlas V 500: Performance to Low Inclination GTO from Cape Canaveral

Performance



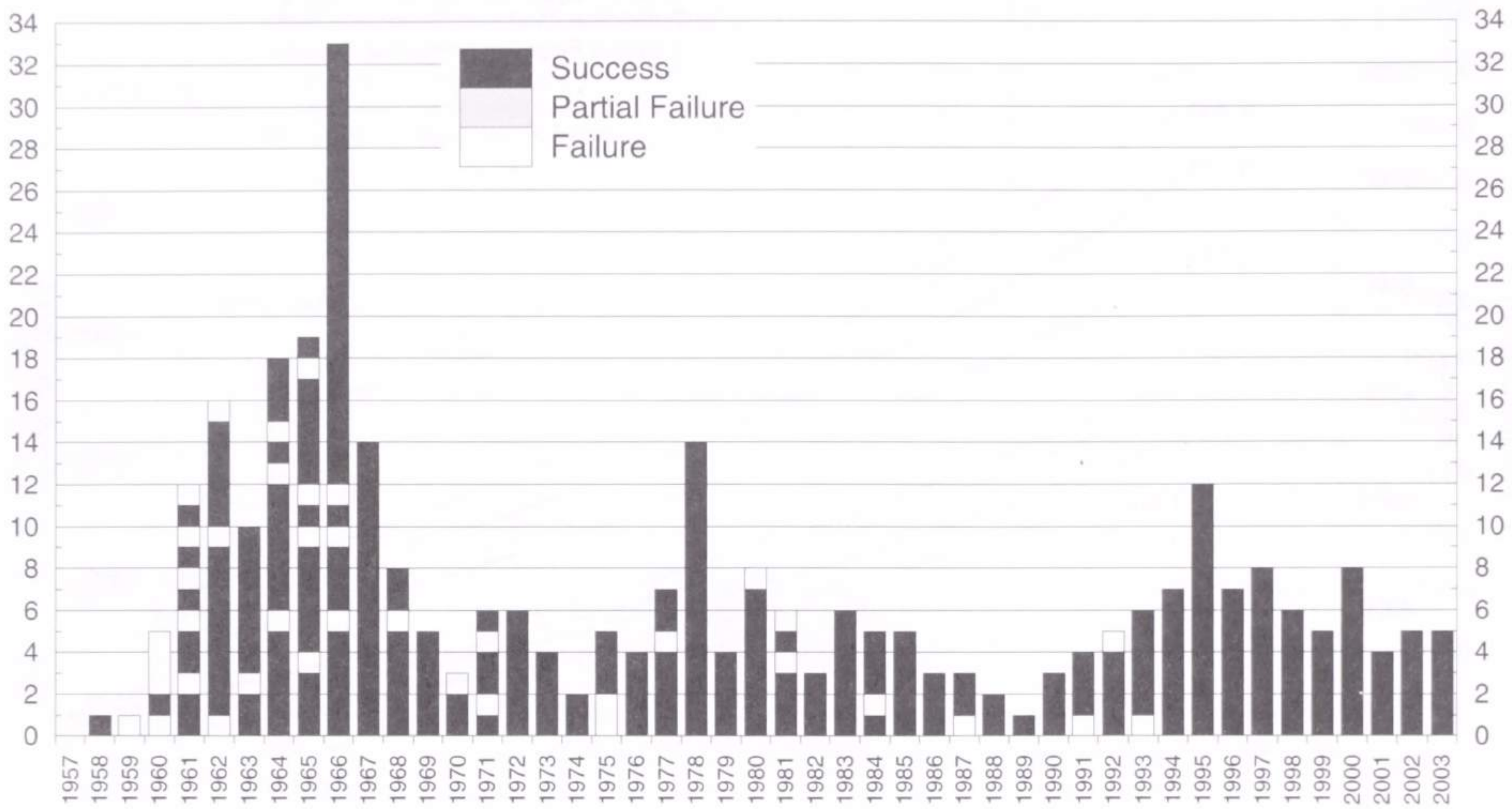
Atlas II and III: Earth Escape Performance from Cape Canaveral



Atlas V: Earth Escape Performance from Cape Canaveral

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	IIAS	All Atlas II	Atlas III	Atlas V	All Atlas Centaur	All Atlas Family
Total Orbital Launches	26	59	4	3	128	324
Launch Vehicle Successes	26	59	4	3	115	285
Launch Vehicle Partial Failures	0	0	0	0	0	0
Launch Vehicle Failures	0	0	0	0	13	39

Flight History

			B	D	E	F	G	H	LV 3A	LV 3C	SLV 3	SLV 3A	SLV 3C	SLV 3D	I	II	IIA	IIAS	IIIA	IIIB	V
	T	F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F
Total	324	39	1/0	18/5	23/2	23/1	7/2	5/0	49/11	11/3	51/5	10/1	17/3	32/3	11/3	10/0	23/0	26/0	1/0	3/0	3/0
1957	0	0																			
1958	1	0	1/0																		
1959	1	1		1/1																	
1960	5	4		2/2					3/2												
1961	12	5		4/1					8/4												
1962	16	3		3/0					13/3												
1963	10	1		1/0					8/1	1/0											
1964	18	3							12/1	2/1	4/1										
1965	19	4		3/1					3/0	2/1	11/2										
1966	33	3		3/0						4/1	26/2										
1967	14	0		1/0						2/0	9/0		2/0								
1968	8	1				2/0			2/0		1/0		3/1								
1969	5	0				1/0						1/0	3/0								
1970	3	1										2/0	1/1								
1971	6	2				1/0						1/1	4/1								
1972	6	0				1/0						1/0	4/0								
1973	4	0										1/0		3/0							
1974	2	0				1/0								1/0							
1975	5	2				1/1						1/0		3/1							
1976	4	0				1/0								3/0							
1977	7	1				2/0						2/0		3/1							
1978	14	0				6/0						1/0		7/0							
1979	4	0				2/0								2/0							
1980	8	1			1/1	4/0								3/0							
1981	6	2			1/1	1/0								4/1							
1982	3	0			1/0									2/0							
1983	6	0			3/0			2/0						1/0							
1984	5	1			3/0		1/1	1/0													
1985	5	0			2/0		3/0														
1986	3	0			1/0		1/0	1/0													
1987	3	1			1/0		1/1	1/0													
1988	2	0			2/0																
1989	1	0					1/0														
1990	3	0			2/0										1/0						
1991	4	1			2/0										1/1	1/0					
1992	5	1													2/1	2/0	1/0				
1993	6	1			1/0										2/1	2/0		1/0			
1994	7	0			2/0										2/0		2/0	1/0			
1995	12	0			1/0										1/0	3/0	3/0	4/0			
1996	7	0													1/0	1/0	4/0	1/0			
1997	8	0													1/0		2/0	5/0			
1998	6	0														1/0	3/0	2/0			
1999	5	0															1/0	4/0			
2000	8	0															4/0	3/0	1/0		
2001	4	0															1/0	3/0			
2002	5	0															2/0	1/0		1/0	1/0
2003	5	0																1/0		2/0	2/0

T = Total Launch Attempts; F = Failures and Partial Failures

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	194	1979 Feb 24	75	F	27F	SLC 3W	1979 017A		Solwind	1331	LEO (97.8)	MIL	USA
	195	May 04	69	SLV 3D	AC 47	LC 36A	1979 038A		Fltsatcom 2	1884	GTO	MIL	USA
	196	Jun 27	54	F	25F	SLC 3W	1979 057A		NOAA 6 (NOAA A)	723	LEO (98.5)	CIV	USA
	197	Sep 20	85	SLV 3D	AC 53	LC 36B	1979 082A		HEAO 3 (HEAO C)	3150	LEO (43.6)	CIV	USA
	198	1980 Jan 18	120	SLV 3D	AC 49	LC 36A	1980 004A		Fltsatcom 3	1884	GTO	MIL	USA
	199	Feb 09	22	F	35F	SLC 3E	1980 011A		GPS 1-5 (5)	759	MEO (63.8)	MIL	USA
	200	Mar 03	23	F	67F	SLC 3W	1980 019A		Unannounced		LEO (63.5)	MIL	USA
	201	Apr 26	54	F	34F	SLC 3E	1980 032A		GPS 1-6 (6)	759	MEO (63.2)	MIL	USA
S	202	May 29	33	F	19F	SLC 3W	1980 043A		NOAA B	1405	LEO (92.2)	CIV	USA
	203	Oct 31	155	SLV 3D	AC 57	LC 36A	1980 087A		Fltsatcom 4	1884	GTO	MIL	USA
	204	Dec 06	36	SLV 3D	AC 54	LC 36B	1980 098A		Intelsat 502	1928	GTO	CML	Int'l
F	**205**	**Dec 09**	**3**	**E**	**68E**	**SLC 3W**	1980 F03A		Unannounced			MIL	USA
	206	1981 Feb 21	74	SLV 3D	AC 42	LC 36A	1981 018A		Comstar 4	1520	GTO	CML	USA
	207	May 23	91	SLV 3D	AC 56	LC 36B	1981 050A		Intelsat 501	1928	GTO	CML	Int'l
	208	Jun 23	31	F	87F	SLC 3W	1981 059A		NOAA 7 (NOAA C)	1405	LEO (99.1)	CIV	USA
F	**209**	**Aug 06**	**44**	**SLV 3D**	**AC 59**	**LC 36A**	1981 073A		FltSatcom 5	1884	GTO	MIL	USA
	210	Dec 15	131	SLV 3D	AC 55	LC 36B	1981 119A		Intelsat 503	1928	GTO	CML	Int'l
F	**211**	**Dec 18**	**3**	**E**	**76E**	**SLC 3E**	1981 F03A		GPS 1-7 (7)	759	MEO (63)	MIL	USA
	212	1982 Mar 05	77	SLV 3D	AC 58	LC 36A	1982 017A		Intelsat 504	1928	GTO	CML	Int'l
	213	Sep 28	207	SLV 3D	AC 60	LC 36B	1982 097A		Intelsat 505	1928	GTO	CML	Int'l
	214	Dec 21	84	E	60E	SLC 3W	1982 118A		DMSP 5D-2 6	751	SSO	MIL	USA
	215	1983 Feb 09	50	H	6001H	SLC 3E	1983 008A		Unannounced		LEO (63.4)	MIL	USA
	216	Mar 28	47	E	73E	SLC 3W	1983 022A		NOAA 8 (NOAA E)	3775	SSO	CIV	USA
	217	May 19	52	SLV 3D	AC 61	LC 36A	1983 047A		Intelsat 506	1928	GTO	CML	Int'l
	218	Jun 09	21	H	6002H	SLC 3E	1983 056A		Unannounced			MIL	USA
	219	Jul 14	35	E	75E	SLC 3W	1983 072A		GPS 1-8 (8)	759	MEO (62.8)	MIL	USA
	220	Nov 18	127	E	58E	SLC 3W	1983 113A		DMSP 7	751	SSO	MIL	USA
	221	1984 Feb 05	79	H	6003H	SLC 3E	1984 012A		Unannounced			MIL	USA
F	**222**	**Jun 09**	**125**	**G**	**AC 62**	**LC 36B**	1984 057A		Intelsat 509	1928	GTO	CML	Int'l
	223	Jun 13	4	E	42E	SLC 3W	1984 059A		GPS 1-9 (9)	759	MEO (62.8)	MIL	USA
	224	Sep 08	87	E	14E	SLC 3W	1984 097A		GPS 1-10 (10)	759	MEO (63.5)	MIL	USA
	225	Dec 12	95	E	39E	SLC 3W	1984 123A		NOAA 9 (NOAA F)	1712	SSO	MIL	USA
	226	1985 Mar 13	91	E	41E	SLC 3W	1985 021A		Geosat 1	635	LEO (108)	MIL	USA
	227	Mar 22	9	G	AC 63	LC 36B	1985 025A		Intelsat 510A	2013	GTO	CML	Int'l
	228	Jun 30	100	G	AC 64	LC 36B	1985 055A		Intelsat 511A	2013	GTO	CML	Int'l
	229	Sep 28	90	G	AC 65	LC 36B	1985 087A		Intelsat 512A	2013	GTO	CML	Int'l
	230	Oct 09	11	E	55E	SLC 3W	1985 093A		GPS 1-11 (11)	760	MEO (63.4)	MIL	USA
	231	1986 Feb 09	123	H	6004H	SLC 3E	1986 014	M	USA 15 - 18			MIL	USA
	232	Sep 17	220	E	52E	SLC 3W	1986 073A		NOAA 10 (NOAA G)	1712	SSO	CIV	USA
	233	Dec 05	79	G	AC 66	LC 36B	1986 096A		FltSatcom 7 (USA 20)	2310	GTO	MIL	USA
F	**234**	**1987 Mar 26**	**111**	**G**	**AC 67**	**LC 36B**	1987 F02A		FltSatcom 6	2300	GTO	MIL	USA
	235	May 15	50	H	6005H	SLC 3E	1987 043	M	USA 22 - 25			MIL	USA
	236	Jun 20	36	E	59E	SLC 3W	1987 053A		DMSP 8 (USA 26)	823	SSO	MIL	USA
	237	1988 Feb 03	228	E	54E	SLC 3W	1988 006A		DMSP 9 (USA 29)	823	SSO	MIL	USA
	238	Sep 24	234	E	63E	SLC 3W	1988 089A		NOAA 11 (NOAA H)	1712	SSO	CIV	USA
	239	1989 Sep 25	366	G	AC 68	LC 36B	1989 077A		FLTSatcom 8 (USA 46)	2310	GTO	MIL	USA
	240	1990 Apr 11	198	E	28E	SLC 3W	1990 031	M	USA 56 - 58			MIL	USA
	241	Jul 25	105	I	AC 69	LC 36B	1990 065A		CRRES	1629	EEO (18.2)	CIV	USA
S	242	Dec 01	129	E	61E	SLC 3W	1990 105A		DMSP 10 (USA 68)	823	SSO	MIL	USA
F	**243**	**1991 Apr 18**	**138**	**I**	**AC 70**	**LC 36B**	1991 F01A		Yuri 3H (BS 3H)		GTO	CML	Japan
	244	May 14	26	E	50E	SLC 3W	1991 032A		NOAA 12 (NOAA D)	1416	SSO	CIV	USA
	245	Nov 28	198	E	53E	SLC 3W	1991 082A		DMSP 11 (USA 73)	823	SSO	MIL	USA
	246	Dec 07	9	II	AC 102	LC 36B	1991 083A		Eutelsat 203	1874	GTO +	CML	Europe
	247	1992 Feb 11	66	II	AC 101	LC 36A	1992 006A		DSCS 3 B-14 (USA 78)	2615	GTO	MIL	USA
	248	Mar 14	32	I	AC 72	LC 36B	1992 013A		Galaxy 5	1412	GTO	CML	USA
	249	Jun 10	88	IIA	AC 105	LC 36B	1992 032A		Intelsat K	2928	GTO	CML	Int'l
	250	Jul 03	23	II	AC 103	LC 36A	1992 038A		DSCS 3 B-12 (USA 82)	2615	GTO	MIL	USA
F	**251**	**Aug 22**	**50**	**I**	**AC 71**	**LC 36B**	1992 F02A		Galaxy 1R		GTO	CML	USA
F	**252**	**1993 Mar 25**	**215**	**I**	**AC 74**	**LC 36B**	1993 015A		UHF F/O 1	2866	GTO -	MIL	USA
	253	Jul 19	116	II	AC 104	LC 36A	1993 046A		DSCS 3 B-9 (USA 93)	2615	GTO	MIL	USA
S	254	Aug 09	21	E	34E	SLC 3W	1993 050A		NOAA 13 (NOAA I)	1712	SSO	CIV	USA
	255	Sep 03	25	I	AC 75	LC 36B	1993 056A		UHF F/O 2	2844	GTO -	MIL	USA
	256	Nov 28	86	II	AC 106	LC 36A	1993 074A		DSCS 3 B-10 (USA 97)	2615	GTO	MIL	USA
	257	Dec 16	18	IIAS	AC 108	LC 36B	1993 077A		Telstar 401	3375	GTO	CML	USA
	258	1994 Apr 13	118	I	AC 73	LC 36B	1994 022A		GOES 8 (GOES I)	2105	GTO +	CIV	USA
	259	Jun 24	72	I	AC 76	LC 36B	1994 035A		UHF F/O 3	2847	GTO -	MIL	USA
	260	Aug 03	40	IIA	AC 107	LC 36A	1994 047A		DBS 2	2860	GTO +	CML	USA

SLC-3E and 3W are at Vandenberg Air Force Base; LC-36A and 36B are at Cape Canaveral Air Force Station

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
261	Aug 29	26	E	20E	SLC 3W	1994 057A	DMSP 12	830	SSO	MIL	USA
262	Oct 06	38	IIAS	AC 111	LC 36B	1994 064A	Intelsat 703	3656	GTO +	CML	Int'l
263	Nov 29	54	IIA	AC 110	LC 36A	1994 079A	Orion 1	2358	GTO +	CML	USA
264	Dec 30	31	E	11E	SLC 3W	1994 089A	NOAA 14 (NOAA J)	1712	SSO	CIV	USA
265	1995 Jan 10	11	IIAS	AC 113	LC 36B	1995 001A	Intelsat 704	3656	GTO +	CML	Int'l
266	Jan 29	19	II	AC 112	LC 36A	1995 003A	UHF F/O 4	3015	GTO -	MIL	USA
267	Mar 22	52	IIAS	AC 115	LC 36B	1995 013A	Intelsat 705	3669	GTO +	CML	Int'l
268	Mar 24	2	E	45E	SLC 3W	1995 015A	DMSP 13	830	SSO	MIL	USA
269	Apr 07	14	IIA	AC 114	LC 36A	1995 019A	AMSC 1	2700	GTO +	CML	USA
270	May 23	46	I	AC 77	LC 36B	1995 025A	GOES 9 (GOES J)	2105	GTO +	CIV	USA
271	May 31	8	II	AC 116	LC 36A	1995 027A	UHF F/O 5	3015	GTO -	MIL	USA
272	Jul 31	61	IIA	AC 118	LC 36A	1995 038A	DSCS 3B-5	2610	GTO	MIL	USA
273	Aug 29	29	IIAS	AC 117	LC 36B	1995 043A	JCSat 3	1841	GTO +	CML	Japan
274	Oct 22	54	II	AC 119	LC 36A	1995 057A	UHF F/O 6	3015	GTO -	MIL	USA
275	Dec 02	41	IIAS	AC 121	LC 36B	1995 065A	SOHO	1875	L1 Halo	CIV	Europe
276	Dec 15	13	IIA	AC 120	LC 36A	1995 069A	Galaxy 3R	2980	GTO -	CML	USA
277	1996 Feb 01	48	IIAS	AC 126	LC 36B	1996 006A	Palapa C1	2980	GTO +	CML	Indonesia
278	Apr 03	62	IIA	AC 122	LC 36A	1996 020A	Inmarsat 301	2066	GTO	CML	Int'l
279	Apr 30	27	I	AC 78	LC 36B	1996 027A	BeppoSAX	1400	LEO (4)	CIV	Italy
280	Jul 25	86	II	AC 125	LC 36A	1996 042A	UHF F/O 7	3020	GTO -	MIL	USA
281	Sep 08	45	IIA	AC 123	LC 36B	1996 054A	GE 1	2770	GTO +	CML	USA
282	Nov 21	74	IIA	AC 124	LC 36A	1996 067A	Hot Bird 2	2910	GTO	CML	Europe
283	Dec 18	27	IIA	AC 129	LC 36B	1996 070A	Inmarsat 303	2074	GTO	CML	Int'l
284	1997 Feb 17	61	IIAS	AC 127	LC 36B	1997 007A	JCSat 4	3094	GTO +	CML	Japan
285	Mar 08	19	IIA	AC 128	LC 36A	1997 011A	Tempo 2	3394	GTO -	CML	USA
286	Apr 25	48	I	AC 79	LC 36B	1997 019A	GOES 10 (GOES K)	2106	GTO +	CIV	USA
287	Jul 28	94	IIAS	AC 133	LC 36B	1997 036A	Superbird C	3130	GTO +	CML	Japan
288	Sep 04	38	IIAS	AC 146	LC 36A	1997 050A	GE 3	2580	GTO +	CML	USA
289	Oct 05	31	IIAS	AC 135	LC 36B	1997 059A	Echostar 3	3282	GTO	CML	USA
290	Oct 25	20	IIA	AC 131	LC 36A	1997 065A	DSCS 3 B-13	2700	GTO -	MIL	USA
						1997 065B A	Falcon Gold		EEO (26.2)	NGO	USA
291	Dec 08	44	IIAS	AC 149	LC 36B	1997 078A	Galaxy 8i	3560	GTO +	CML	USA
292	1998 Jan 29	52	IIA	AC 109	LC 36A	1998 005A	Capricorn		GTO	MIL	USA
293	Feb 28	30	IIAS	AC 151	LC 36B	1998 014A	Intelsat 806	3570	GTO	CML	Int'l
294	Mar 16	16	II	AC 132	LC 36A	1998 016A	UHF F/O 8	3200	GTO -	MIL	USA
295	Jun 18	94	IIAS	AC 135	LC 36B	1998 037A	Intelsat 805	3692	GTO	CML	Int'l
296	Oct 09	113	IIA	AC 134	LC 36B	1998 057A	Hot Bird 5	2900	GTO	CML	Europe
297	Oct 20	11	IIA	AC 130	LC 36A	1998 058A	UHF F/O 9	3200	GTO -	MIL	USA
298	1999 Feb 16	119	IIAS	AC 152	SLC 36A	1999 006A	JCSat 6	2904	GTO	CML	Japan
299	Apr 12	55	IIAS	AC 154	SLC 36A	1999 018A	Eutelsat W3	3178	GTO	CML	Europe
300	Sep 23	164	IIAS	AC 155	LC 36A	1999 050A	Echostar 5	3177	GTO+	CML	USA
301	Nov 23	61	IIA	AC 136	LC 36B	1999 063A	UHF F/O 10	3200	GTO	MIL	USA
302	Dec 18	25	IIAS	AC 141	SLC 3E	1999 068A	Terra	5190	SSO	CIV	USA
303	2000 Jan 21	34	IIA	AC 138	LC 36A	2000 001A	DSCS 3 B8	2610	GTO	MIL	USA
304	Feb 03	13	IIAS	AC 158	LC 36B	2000 007A	Hispasat 1C	3100	GTO	CML	Spain
305	May 03	90	IIA	AC 137	LC 36A	2000 022A	GOES 11 (GOES L)	2105	GTO	CIV	USA
306	May 24	21	IIIA	AC 201	LC 36B	2000 028A	Eutelsat W4	3190	GTO+	CML	Europe
307	Jun 30	37	IIA	AC 139	LC 36A	2000 034A	TDRS 8 (TDRS H)	3180	GTO-	CIV	USA
308	Jul 14	14	IIAS	AC 161	LC 36B	2000 038A	EchoStar 6	3700	GTO+	CML	USA
309	Oct 20	98	IIA	AC 140	LC 36A	2000 065A	DSCS 3 B11	1225	GTO	MIL	USA
310	Dec 06	47	IIAS	AC 157	LC 36A	2000 080A	USA 155		GTO	MIL	USA
311	2001 Jun 19	195	IIAS	AC 156	LC 36B	2001 026A	ICO 2	2750	MEO (45)	CML	USA
312	Jul 23	34	IIA	AC 142	LC 36A	2001 031A	GOES 12 (GOES M)	2279	GTO	CIV	USA
313	Sep 08	47	IIAS	AC 160	SLC 3E	2001 040A	USA 160	5000	LEO (63)	MIL	USA
314	Oct 11	33	IIAS	AC 162	LC 36B	2001 046A	USA 162		GTO	MIL	USA
315	2002 Feb 21	133	IIIB DEC	AC 162	LC 36B	2002 006A	Echostar 7	4027	GTO+	CML	USA
316	Mar 08	15	IIA	AC 146	LC 36A	2002 011A	TDRS 9 (TDRS I)	3192	GTO-	CIV	USA
317	Aug 21	166	V 401	AV 001	SLC 41	2002 038A	Hot Bird 6	3905	GTO+	CML	France
318	Sep 18	28	IIAS	AC 159	LC 36A	2002 044A	Hispasat 1D	3250	GTO+	CML	Spain
319	Dec 05	78	IIA	AC 144	LC 36A	2002 055A	TDRS 10 (TDRS J)	3192	GTO-	CIV	USA
320	2003 Apr 12	128	IIIB SEC	AC 205	SLC 36B	2003 014A	AsiaSat 4	4042	GTO	CML	China
321	May 13	31	V 401	AV 002	SLC 41	2003 020A	Hellas Sat 2	3292	GTO	CML	Greece
322	Jul 17	65	V 521	AV 003	SLC 41	2003 033A	Rainbow 1	4328	GTO	CML	USA
323	Dec 02	138	IIAS	AC 164	SLC 3E	2003 054A	USA 173		SSO	MIL	USA
324	Dec 18	16	IIIB	AC 203	SLC 36B	2003 057A	UHF F/O F11	3200	GTO	MIL	USA

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

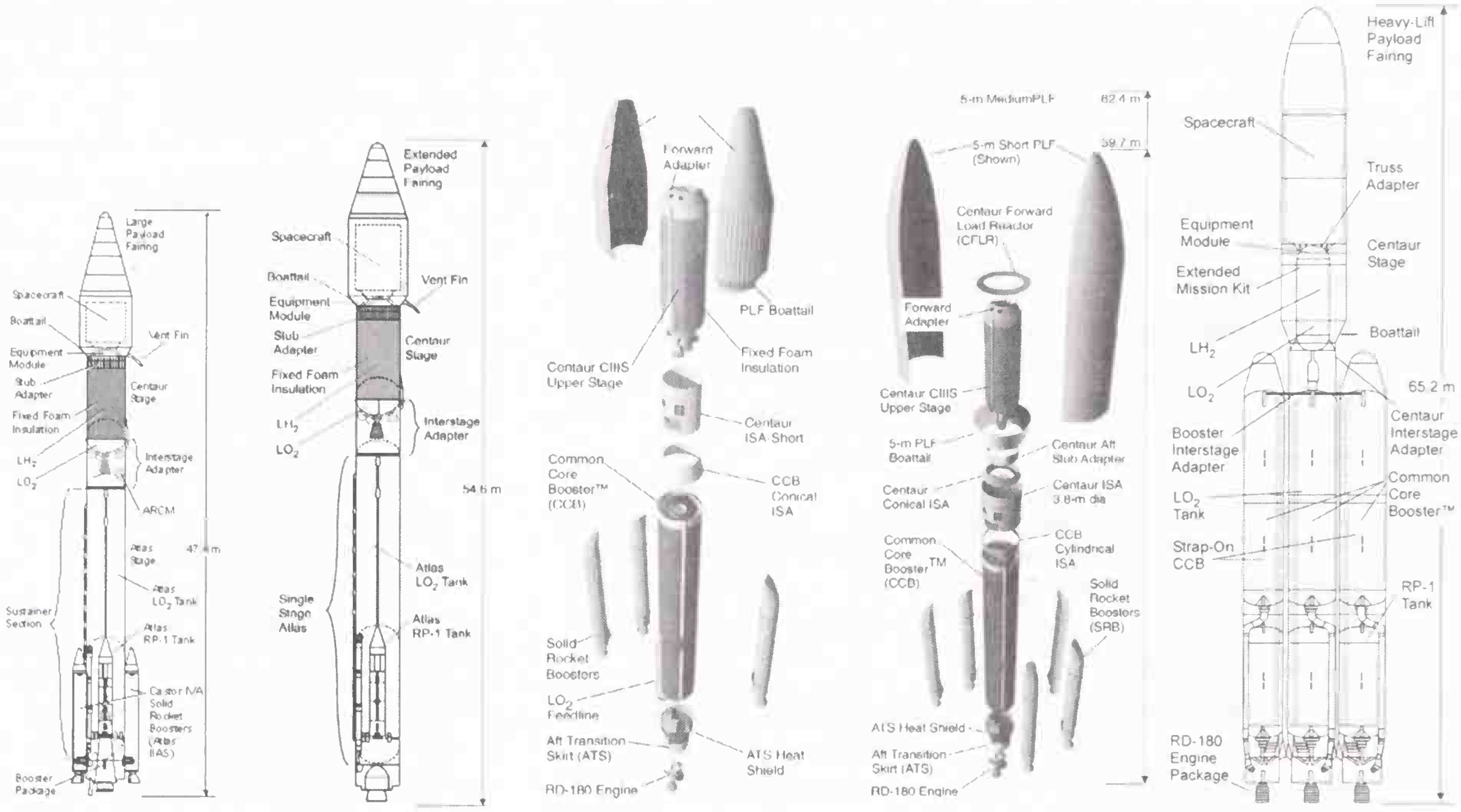
FLIGHT HISTORY

Failure Descriptions:

F	1959 Nov 26	20D	1959 F09	Agena payload shroud broke away prematurely at T+45 s.
F	1960 Feb 26	29D	1960 F03	Agena second stage failed to separate.
F	1960 Sep 25	80D	1960 F11	Second-stage oxidizer system malfunction.
F	1960 Oct 11	57D	1960 F13	Second-stage Agena guidance system malfunction, stage ignited but did not reach orbit.
F	1960 Dec 15	91D	1960 F18	Airframe failure; vehicle exploded T+70 s.
F	1961 Apr 25	100D	1961 F03	Flight control malfunction; destroyed by range safety officer.
S	1961 Aug 23	Ranger 1	1961 021A	Failed attempt to reach deep space orbit.
F	1961 Sep 09	106D	1961 F09	Electrical malfunction; exploded on launch pad.
F	1961 Oct 21	105D	1961 028A	Flight control system malfunction.
S	1961 Nov 18	Ranger 2	1961 032A	Failed attempt to reach deep space orbit.
F	1961 Nov 22	108D	1961 F13	Flight control system malfunction.
F	1961 Dec 22	114D	1961 035A	Flight control system malfunction.
F	1962 Jan 26	121D (AA3)	1962 001A	Guidance system malfunction.
F	1962 Jul 22	145D (AA5)	1962 F07	Vehicle veered off course at launch and was destroyed because of incorrectly coded guidance software equation.
F	1962 Dec 17	131D	1962 F09	Hydraulics system malfunction.
F	1963 Jun 13	139D	1963 F09	Hydraulics system malfunction.
S	1963 Jul 19	ERS 9	1963 030A	Spacecraft failed to eject.
F	1964 Jun 30	AC 3	1964 F09	Hydraulic pump failure caused short Centaur engine burn.
F	1964 Oct 08	7103	1964 F11	Failure to orbit.
F	1964 Nov 05	289D (AA11)	1964 073A	Agena shroud failed to separate.
S	1965 Jan 21	OV1 1	1965 F01	Propulsion failure.
F	1965 Mar 02	AC 5	1965 F04	Fuel prevalve inadvertently closed.
F	1965 May 28	68D	1965 F06	Engine failure, launch vehicle exploded.
F	1965 Jul 12	7112	1965 F07	Propulsion failure, vehicle destroyed by range safety officer.
F	1965 Oct 25	5301	1965 F11	Agena exploded at T+360 s.
F	1966 Apr 07	AC 8	1966 030A	Centaur failure.
S	1966 April 8	OAO 1	1966 031A	Satellite battery failed second day in orbit.
F	1966 May 17	5303	1966 F07	Flight control system malfunction.
F	1966 Jun 01	5304	1966 046A	Fairing separation failed.
S	1966 Jul 14	OV1 7	1966 063A	Injection motor failed.
S	1967 Apr 06	ATS 2	1967 031A	Satellite failed to reach GEO.
S	1967 Jul 27	OV1 11	1967 072	Satellite propulsion system malfunctioned.
S	1968 Apr 06	OV1 14	1968 026A	Satellite power system failed after one week.
F	1968 Aug 10	AC 17	1968 068A	Centaur oxidizer leak prevented engine restart.
S	1968 Aug 16	10 payloads	1968 F07	Did not reach orbit because of failure of Burner II upper stage.
S	1969 Aug 12	ATS 5 (ATS E)	1969 069A	Failure of spacecraft nutation damper caused flat spin after reaching GSO.
F	1970 Nov 30	AC 21	1970 F09	Shroud failed to separate.
F	1971 May 09	AC 24	1971 F04	Centaur failed to reach orbit because of avionics malfunction.
F	1971 Dec 04	5503A	1971 F13	First-stage sustainer enginer turbine malfunction.
F	1975 Feb 20	AC 33	1975 F01	Failure because of staging electrical disconnect.
F	1975 Apr 13	71F	1975 F03	External explosion in flame bucket.
F	1977 Sep 30	AC 43	1977 F05	Failure because of hot gas leak from first stage engine gas generator.
S	1980 May 29	NOAA B	1980 043A	Spacecraft failed to reach orbit because of satellite propulsion malfunction.
F	1980 Dec 09	68E	1980 F03	Booster lube oil flow loss.
F	1981 Aug 06	AC 59	1981 073A	Satellite reached GEO but did not become operational because of damage caused by shroud during launch. Used for engineering studies of onboard systems.
F	1981 Dec 19	76E	1981 F03	Booster engine gas generator cooling plugged.
F	1984 Jun 09	AC 62	1984 057A	A leak in the Centaur LOX tank, which started at first/second-stage separation resulted in a pressure differential across the common tank bulkhead that caused it to collapse before the second burn, stranding satellite in wrong orbit.
F	1987 Mar 26	AC 67	1987 F02	Vehicle was struck by lightning at T+48 s. Resulting electrical transients in guidance system caused vehicle to yaw and loose control, resulting in destruction by range safety officer.
S	1990 Dec 01	DMSP 10 (USA 68)	1990 105A	Spacecraft propulsion broken nozzle resulted in lower than planned orbit perigee, but spacecraft did become operational.
F	1991 Apr 18	AC 70	1991 F01	Following ground prechilling, air entered Centaur C-1 engine through stuck check valve and froze in LH_2 turbopump and gearbox. At Centaur ignition, engine did not achieve full thrust, causing stage to tumble. Vehicle was destroyed by range safety officer. Fault was not properly diagnosed until flight AC 71.
F	1992 Aug 22	AC 71	1992 F02	Following ground prechilling, air entered Centaur C-1 engine through stuck check valve and froze in turbopump and gearbox. At Centaur ignition, engine did not achieve full thrust, causing stage to tumble. Vehicle was destroyed by range safety officer.
F	1993 Mar 25	AC 74	1993 015A	Inadequately torqued set screw in first-stage engine precision regulator resulted in reduced oxygen flow to gas generator. Engine suffered reduced power and early shut-down, stranding satellite in low orbit.
S	1993 Aug 09	NOAA 13 (NOAA I)	1993 050A	Spacecraft failed after 12 days in orbit because of power malfunction.
S	2002 Mar 8	TDRSS 9	2002 011A	A stuck valve prevented one of two spacecraft propellant tanks from being properly pressurized. By performing a series of short burns and repressurizing after each one, the spacecraft was able to reach GEO 6 months later.

Vehicle Design

Overall Vehicle



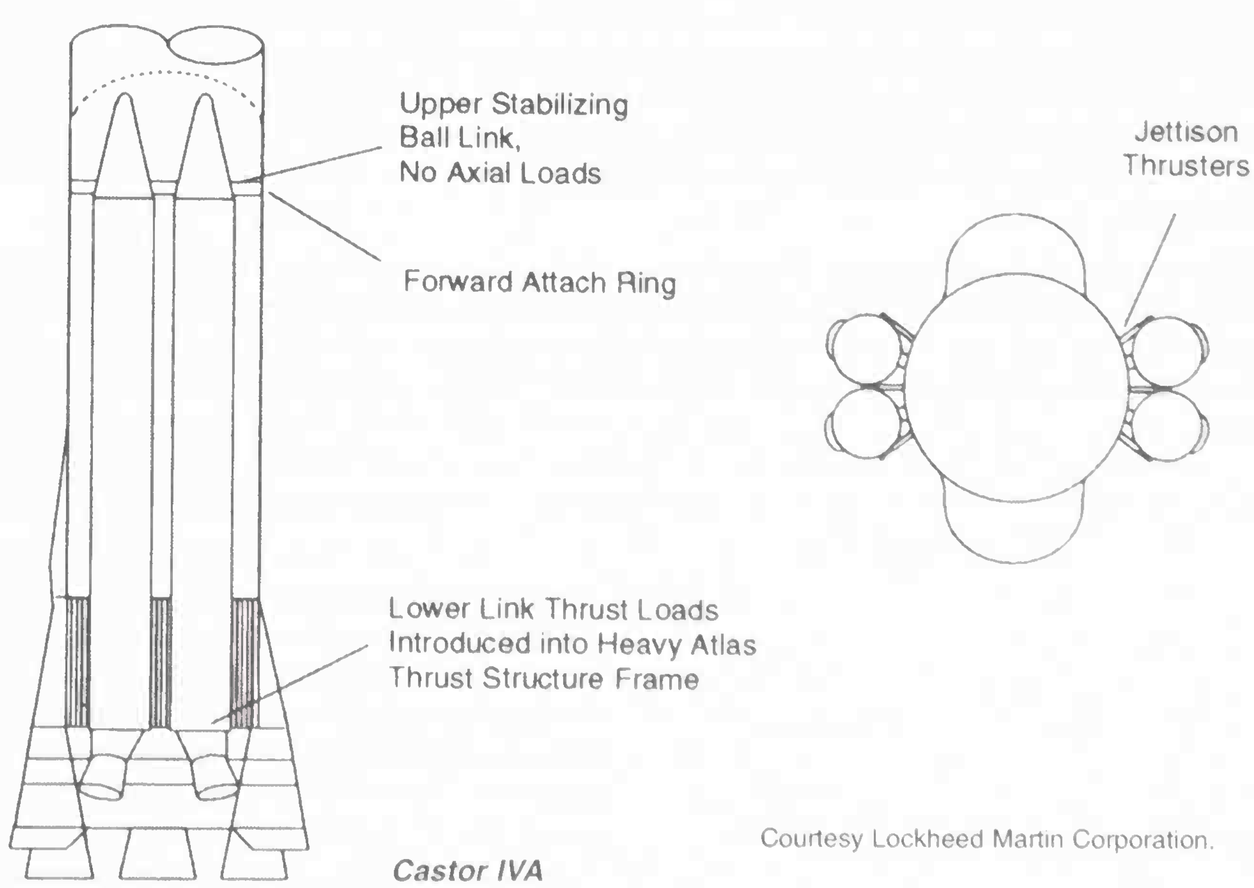
Courtesy Lockheed Martin Corporation.

	Atlas IIAS	Atlas III	Atlas V 400	Atlas V 500	Atlas V HLV
Height	47.4 m (156 ft) with LPF fairing	IIIA: 52.8 m (173.2 ft) IIIB: 53.1 m (174.2 ft) with EPF fairing	58.3 m (191.2 ft) with EPF fairing	62.2 m (204 ft) with 5-m Medium fairing	65.2 m (214 ft)
Gross Liftoff Mass	237.2 t (522.9 klbm)	IIIA: 220.7 t (486.5 klbm) IIIB: 225.4 t (496.9 klbm)	333.3 t (734.8 klbm)	V 551: 540.3 t (1191.2 klbm)	?
Thrust at Liftoff	3.0 MN (676.2 klbf)	2.6 MN (585,000 klbf) (~70% throttle)	3827 kN (860.2 klbf)	?	?

Solid Rocket Boosters

The Atlas IIAS uses four standard Castor IVA solid strap-on motors provided by ATK Thiokol. Two motors ignite at liftoff, and the remaining pair ignite shortly after the first two burn out and are jettisoned. To accommodate the SRBs, several changes were made to the Atlas IIAS first stage. These include a stronger booster package structure and a new tank ring to support the attachment points. Cork heat shields and thermal radiation shields have been enhanced and first-stage retro-rockets were relocated. The motors contain an inadvertent separation destruct system (ISDS). In the event of premature separation, the ISDS fires a conical-shaped charge, mounted on the forward bulkhead of the motor, that terminates motor thrust.

The Atlas V strap-on solid rocket motor is a new design produced by Aerojet. It is approximately four times larger than the Castor IVA. The Atlas V 400 series can be configured with up to three motors, while the Atlas V 500 series can use up to five. All motors are ignited at liftoff and jettisoned in pairs after burnout.



Courtesy Lockheed Martin Corporation.

Vehicle Design

	Atlas IIAS Castor IVA	Atlas V SRB
Dimensions		
Length	13.6 m (44 ft)	19.5 m (64.0 ft)
Diameter	1 m (3.3 ft)	1.55 m (5.1 ft)
Mass		
Propellant Mass	10.1 t (22.3 klbm) each	42.5 t (93.8 klbm)
Inert Mass	1450 kg (3200 lbm) each	4052 kg (8933 lbm)
Gross Mass	11.6 t (25.5 klbm) each	46.5 t (102.5 klbm)
Propellant Mass Fraction	0.87	0.92
Structure		
Type	Monocoque	Filament-wound monocoque
Material	Steel	Graphite–epoxy
Propulsion		
Motor Designation	Castor IVA (Thiokol)	SRB (Aerojet)
Number of Motors	4	0–5
Propellant	HTPB	HTPB
Number of Segments	1	1
Average Thrust (per motor)	433.7 kN (97,500 lbf)	1361 kN (305.9 klbf)
Isp	Sea level: 237.8 s	Vacuum: 275 s
Chamber Pressure	47.6 bar (691 psi)	100.9 bar (1463 psi)
Nozzle Expansion Ratio	8.3:1	16:1
Attitude Control		
Pitch, Yaw, Roll	First stage provides control. SRBs have fixed 11-deg nozzle cant for ground-lit motors, 7-deg nozzle cant for air-lit motors.	First stage provides control. SRBs have fixed 3-deg nozzle cant.
Staging		
Nominal Burn Time	56.2 s	94 s
Shutdown Process	Burn to depletion	Burn to depletion
Stage Separation	Spring ejection	Small thruster firings

Stage 1

The Atlas II booster stage includes two unique features that derive from its history as an early ICBM. The first-stage propellant is stored in lightweight "balloon" tanks, made of thin-walled fully monocoque stainless steel. When pressurized or mechanically stretched by ground support equipment, the tanks are very strong, but without tension the structure would collapse under its own weight. The upper LOX tank and lower RP-1 tank are separated by a common ellipsoidal bulkhead. The second unique feature is the "stage-and-a-half" propulsion system, which was developed to address early concerns about reliably igniting second-stage engines in flight. All three thrust chambers of the Rocketdyne MA-5A propulsion system are ignited before liftoff. Midway through the first-stage flight, the booster section, containing two thrust chambers and associated structures, is jettisoned. The remaining engine and the complete tank assembly are referred to as the sustainer section and continue operating until all the first-stage propellants are exhausted. Because the single-chamber sustainer engine cannot control roll, an Atlas roll control module (ARCM) with hydrazine thrusters is attached to the interstage.

Both of these historical features of the Atlas are eliminated in new Atlas versions. Beginning with the Atlas III first stage, the propulsion system was replaced by a single Russian RD-180 engine. Derived from the RD-170 engine used in the Zenit first stage, the RD-180 is a twin-chambered engine with throttling capability. The RD-180 is powered by a turbopump with a LOX-rich staged combustion cycle. This enables a very high chamber pressure, making the RD-180 one of the highest performance LOX–kerosene engines in the world. The engine is also very self contained, and includes its own thrust structure, hydraulic gimbal systems, and pneumatics for valve control. Because the RD-180 is a single unit, the half-stage booster–sustainer separation is eliminated. However, the stainless steel tank design was retained in the Atlas III, with increased skin thickness and a 3-m (10-ft) longer LOX tank to accommodate the higher mixture ratio of the RD-180 engine. Because the twin nozzles of the RD-180 can be gimbaled throughout flight to control roll, the ARCM is eliminated.

The Atlas V continues to use the RD-180 engine, but the first-stage structure has been completely redesigned. The first stage is referred to as the common core booster (CCB). One CCB is used for the Atlas V 400 and 500 series. In the CCB the stainless steel pressure-stabilized tanks are replaced by structurally stable aluminum isogrid tanks with a larger diameter of 3.8 m (12.5 ft). The total stage length is increased, and instead of sharing a common bulkhead the tanks are independent. Two interstage assemblies are added to the top of the CCB, with different configurations depending on the payload fairing size. For the Atlas V 400, a conical 420-kg (925-lbm) graphite–epoxy booster interstage adapter provides the interface between the larger diameter first stage and the smaller diameter Centaur. A 375-kg (825-lbm) aluminum–lithium Centaur interstage adapter on top of the booster adapter supports the upper stage. To interface with the larger fairing of the Atlas V 500 series, a short cylindrical 270-kg (600-lbm) booster interstage adapter is used. This is topped by a 1300-kg (4250-lbm) composite-sandwich Centaur interstage adapter. These structures are not included in the inert mass listed next.

Vehicle Design

	Atlas IIAS	Atlas IIIA/B	Atlas V
Dimensions			
Length	24.9 m (81.7 ft) + 4-m (13-ft) interstage	28.91 m (94.9 ft) + 4.4-m (14.5-ft) interstage	Core: 32.46 m (106.5 ft) HLV strap-ons: 36.34 m (119.2 ft)
Diameter	3.05 m (10 ft)	3.05 m (10 ft)	3.8 m (12.5 ft)
Mass			
Propellant Mass	156.4 t (344.8 klbm)	183.2 t (403.9 klbm)	284.1 t (626.3 klbm)
Inert Mass	10.7 t (23.6 klbm) + 545-kg (1200-lbm) interstage	13725 kg (30,260 lbm) + 465-kg (1025-lbm) interstage	20,743 kg (45,730 lbm)
Gross Mass	167.6 t (369.6 klbm) with interstage	196.9 t (434.1 klbm)	304.8 t (672.0 klbm)
Propellant Mass Fraction	0.93	0.93	0.93
Structure			
Type	Tanks: pressure-stabilized monocoque Thrust structure: integrally machined panels Interstage: skin-stringer	Tanks: pressure-stabilized monocoque Thrust structure: integrally machined panels Interstage: skin-stringer	Tanks: Structurally stable isogrid Aft transition skirt: Integrally machined structure
Material	Tanks: stainless steel Thrust structure: Aluminum Interstage: Aluminum	Tanks: stainless steel Thrust structure: Aluminum Interstage: Aluminum–lithium	Aluminum
Propulsion			
Engine Designation	MA-5A (Rocketdyne)	RD-180 (Pratt & Whitney/ NPO Energomash)	RD-180 (Pratt & Whitney/ NPO Energomash)
Number of Engines	1 propulsion system with 2 sets of turbopumps and 3 thrust chambers (2 boosters, 1 sustainer)	1 (2 thrust chambers)	1 (2 thrust chambers)
Propellant	LOX/RP-1 (kerosene)	LOX/RP-1 (kerosene)	LOX/RP-1 (kerosene)
Average Thrust	Booster: Sea level: 1854 kN (416 klbf) Vacuum: 2065 kN (464 klbf) Sustainer: Sea level: 266kN (59.8 klbf) Vacuum: 380.6 kN (85.6 klbf)	Sea level: 3827 kN (860.2 klbf) Vacuum: 4152 kN (933.4 klbf)	Sea level: 3827 kN (860.2 klbf) Vacuum: 4152 kN (933.4 klbf)
Isp	Booster: Sea level: 262.1 s Vacuum: 293.4 s Sustainer: Sea level: 216.1 s Vacuum: 311.0 s	Sea level: 311.3 s Vacuum: 337. 8 s	Sea level: 311.3 s Vacuum: 337. 8 s
Chamber Pressure	Booster: 44.1 bar (639 psi) Sustainer: 50.7 bar (735 psi)	257 bar (3734 psi)	257 bar (3734 psi)
Nozzle Expansion Ratio	Booster: 8:1 Sustainer: 25:1	36.87:1	36.87:1
Propellant Feed System	Gas-generator turbopump	Staged-combustion turbopump	Staged-combustion turbopump
Mixture Ratio (O/F)	Booster: 2.25:1 Sustainer: 2.27:1	2.72:1 Sustainer: 2.27:1	2.72:1
Throttling Capability	100% only	47–100%	47–100%
Restart Capability	None	None	None
Tank Pressurization	High-pressure helium	High-pressure helium	High-pressure helium

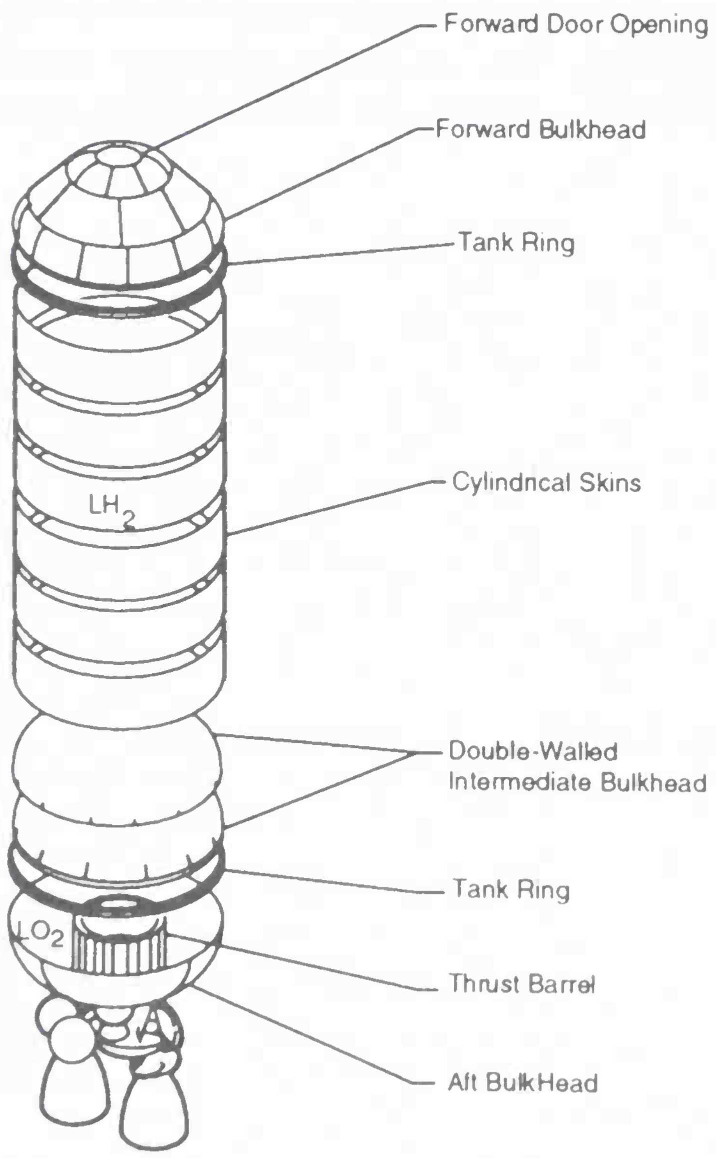
Vehicle Design

	Atlas IIAS	Atlas IIIA/B	Atlas V
Attitude Control			
Pitch, Yaw	Hydraulic nozzle gimbaling	Pumped hydraulic nozzle gimbaling ±8 deg	Pumped hydraulic nozzle gimbaling ±8 deg
Roll	Nozzle gimbaling, hydrazine thrusters on ARCM	Pumped hydraulic nozzle gimbaling ±8 deg	Pumped hydraulic nozzle gimbaling ±8 deg
Staging			
Nominal Burn Time	Booster: 163 s Sustainer: 289 s	190 s	235–250 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion
Stage Separation	Booster: 16 pneumatic tension-release mechanisms Sustainer: 8 retro-rockets	12 retro-rockets	8 retro-rockets

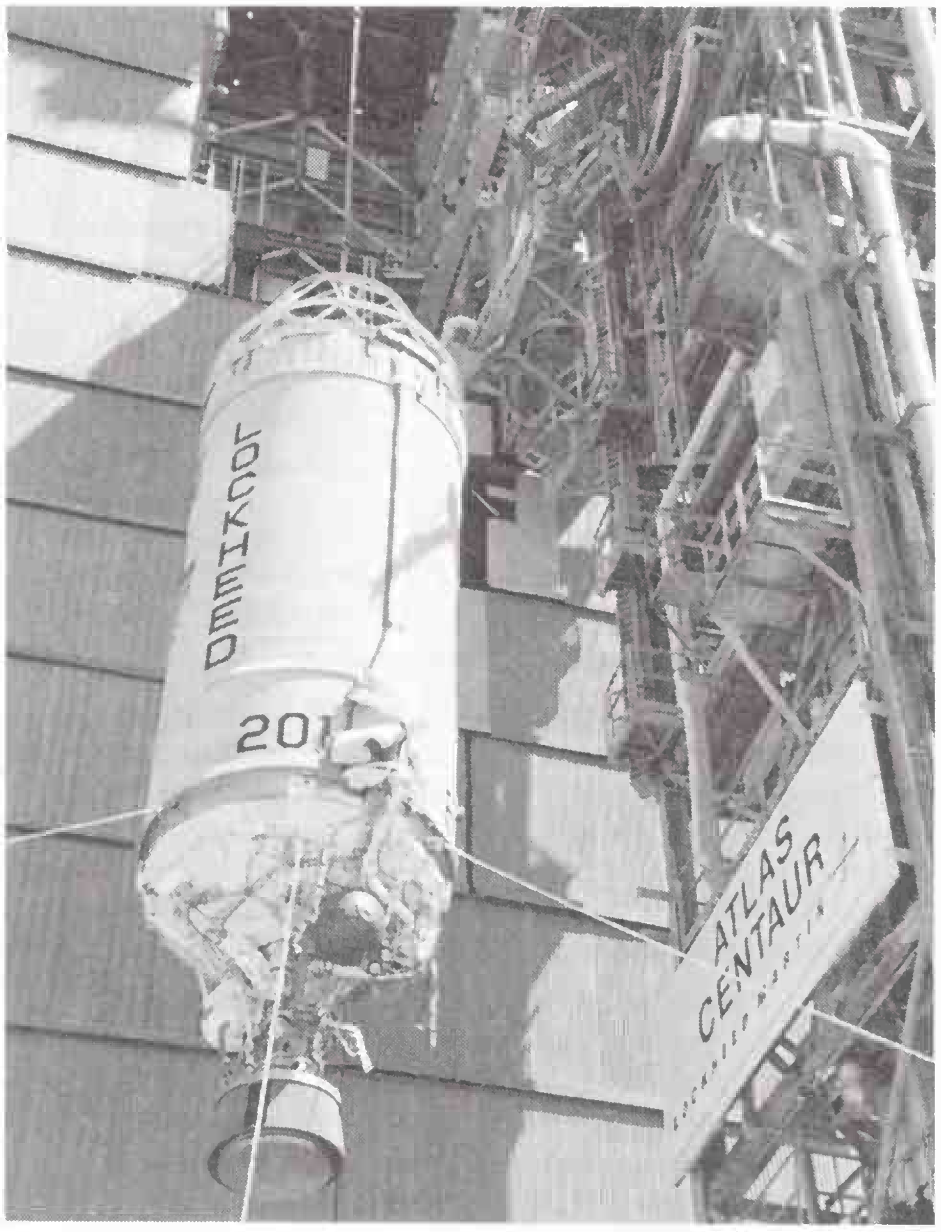
Stage 2

The Centaur was the world's first oxygen–hydrogen upper stage when it was developed in the early 1960s. Since then, the propulsion and avionics systems have been significantly upgraded. The tankage is similar to the original Atlas stage, with pressure-stabilized steel tanks sharing a common bulkhead. The RL10 engines and associated systems are mounted on the bottom of the tanks. On the top are the stub adapter, which interfaces with the fairing, and the equipment section, which houses the avionics and supports the payload adapter.

A single-engine Centaur (SEC) configuration was developed for Atlas IIIA, which reduces cost and improves reliability compared to the traditional twin-engine configuration. While the SEC provides improved performance to GTO through reduced weight, the lower thrust penalizes performance to LEO. Therefore, the Atlas IIIB and Atlas V use a common Centaur design called the Centaur III that can be produced as either a SEC or dual-engine Centaur (DEC). The Centaur III has 1.7-m (5.5-ft) longer propellant tanks than the Centaur used on Atlas IIAS. The SEC version uses a single RL10A-4-2 engine with electromechanical gimbal actuators. The DEC uses two engines with hydraulic gimbal actuators. The RL10A-4-2 has similar performance to the A-4-1 version, with operational and reliability upgrades. When used inside the 5-m payload fairing of the Atlas V 500 or HLV, the Centaur can be modified to support injection directly into GEO. This is implemented by adding the extended mission kit, which includes improved insulation and thermal radiation shields to reduce propellant boiloff, added hydrazine and helium bottles, and improved avionics batteries.



Centaur



Courtesy Lockheed Martin Corporation.

The first single-engine Centaur (SEC) is lifted into place atop vehicle AC-201.

Vehicle Design

	Centaur IIA/IIAS (Atlas IIA/IIAS)	Centaur IIIA (Atlas IIIA)	Centaur III (Atlas IIIB and V)
Dimensions			
Length	10.06 m (33 ft)	10.5 m (34.5 ft)	12.68 m (41.6 ft) with nozzles extended
Diameter	3.05 m (10 ft)	3.05 m (10 ft)	3.05 m (10 ft)
Mass			
Propellant Mass	16,780 kg (37,000 lbm)	16,930 kg (37,320 lbm)	20,830 kg (45,920 lbm)
Inert Mass	2200 kg (4850 lbm)	1720 kg (3790 lbm)	SEC: 1914 kg (4220 lbm) DEC: 2106 kg (4623 lbm)
Gross Mass	18,980 kg (41850 lbm)	18,650 kg (41,110 lbm)	SEC: 22,744 kg (50,141 lbm) DEC: 22,936 kg (50,565 lbm)
Propellant Mass Fraction	0.88	0.91	SEC: 0.92 DEC: 0.91
Structure			
Type	Tanks: pressure-stabilized monocoque Equipment section: skin-stringer	Tanks: pressure-stabilized monocoque Equipment section: skin-stringer	Tanks: pressure-stabilized monocoque Equipment section: skin-stringer
Material	Tanks: stainless steel Equipment section: Aluminum	Tanks: stainless steel Equipment section: Aluminum	Tanks: stainless steel Equipment section: Aluminum
Propulsion			
Engine Designation	RL10A-4 or RL10A-4-1 (Pratt & Whitney)	RL10A-4-1 (Pratt & Whitney)	RL10A-4-2 (Pratt & Whitney)
Number of Engines	2	1	1 or 2
Propellant	LOX/LH_2	LOX/LH_2	LOX/LH_2
Average Thrust (Total)	RL10A-4: 185.2 kN (41.6 klbf) RL10A-4-1: 198.4 kN (44.6 klbf)	99.2 kN (22.3 klbf)	SEC: 99.2 kN (22.3 klbf) DEC: 198.4 kN (44.6 klbf)
Isp	RL10A-4: 449.0 s RL10A-4-1: 450.5 s	450.5 s	450.5 s
Chamber Pressure	32.1 bar (465 psi)	32.1 bar (465 psi)	32.1 bar (465 psi)
Nozzle Expansion Ratio	85:1	85:1	85:1
Propellant Feed System	Split-expander turbopump	Split-expander turbopump	Split-expander turbopump
Mixture Ratio (O/F)	5.5:1	5.5:1	5.5:1
Throttling Capability	None	None	None
Restart Capability	Multiple	Multiple	Multiple
Tank Pressurization	Helium gas, autogenous hydrogen	Helium gas, autogenous hydrogen	Helium gas, autogenous hydrogen
Attitude Control			
Pitch, Yaw	Hydraulic gimbaling	Electromechanical gimbaling	SEC: Electromechanical gimbaling DEC: Hydraulic gimabling
Roll	Hydraulic gimbaling	27-N (6-lbf) hydrazine thrusters	SEC: 27-N (6-lbf) hydrazine thrusters DEC: Hydraulic gimabling
Staging			
Nominal Burn Time	370 s	740 s	SEC: 900 s DEC: 440 s
Shutdown Process	Burn to depletion or command shutdown	Burn to depletion or command shutdown	Burn to depletion or command shutdown
Stage Separation	Payload separation system	Payload separation system	Payload separation system

Avionics

The Atlas first stage is integrated with the Centaur avionics for guidance, flight control, and event sequencing. Systems specific to the first stage, such as range safety flight termination, propellant management, telemetry, and power, are housed in an equipment pod running vertically along the outside of the first stage. The first stage also includes a flight control system.

The Centaur avionics components are mounted on the outside of the conical equipment module for easy access. The primary component of the guidance system is the inertial navigation unit (INU). The INU combines ring laser gyros with a 1750A processor to provide inertial navigation and all guidance and control computation, as well as management of the Centaur tank pressures and propellant usage. Commands are distributed through 128 solid-state switches in the Centaur RCU and 64 switches in the Atlas RCU, through a MIL-STD-1553B data bus. The telemetry system includes a master data unit, which receives and formats data directly from the INU, and from vehicle sensors through two remote data units (one for each stage). The avionics are powered by a 28-Vdc power distribution unit, supplied by a battery. A redundant secure flight termination system has independent receivers and power, and can cause engine shutdown and vehicle destruct upon command from the range safety officer.

Vehicle Design

Attitude Control System

First-stage attitude control is provided by hydraulically gimbaling the main engine nozzles for pitch and yaw. The Atlas V controls roll with opposing nozzles. The Atlas II series sustainer section, with only one engine, used a hydrazine-fueled ARCM mounted on the interstage to control roll. This consists of a small hydrazine tank and two opposing pairs of thrusters.

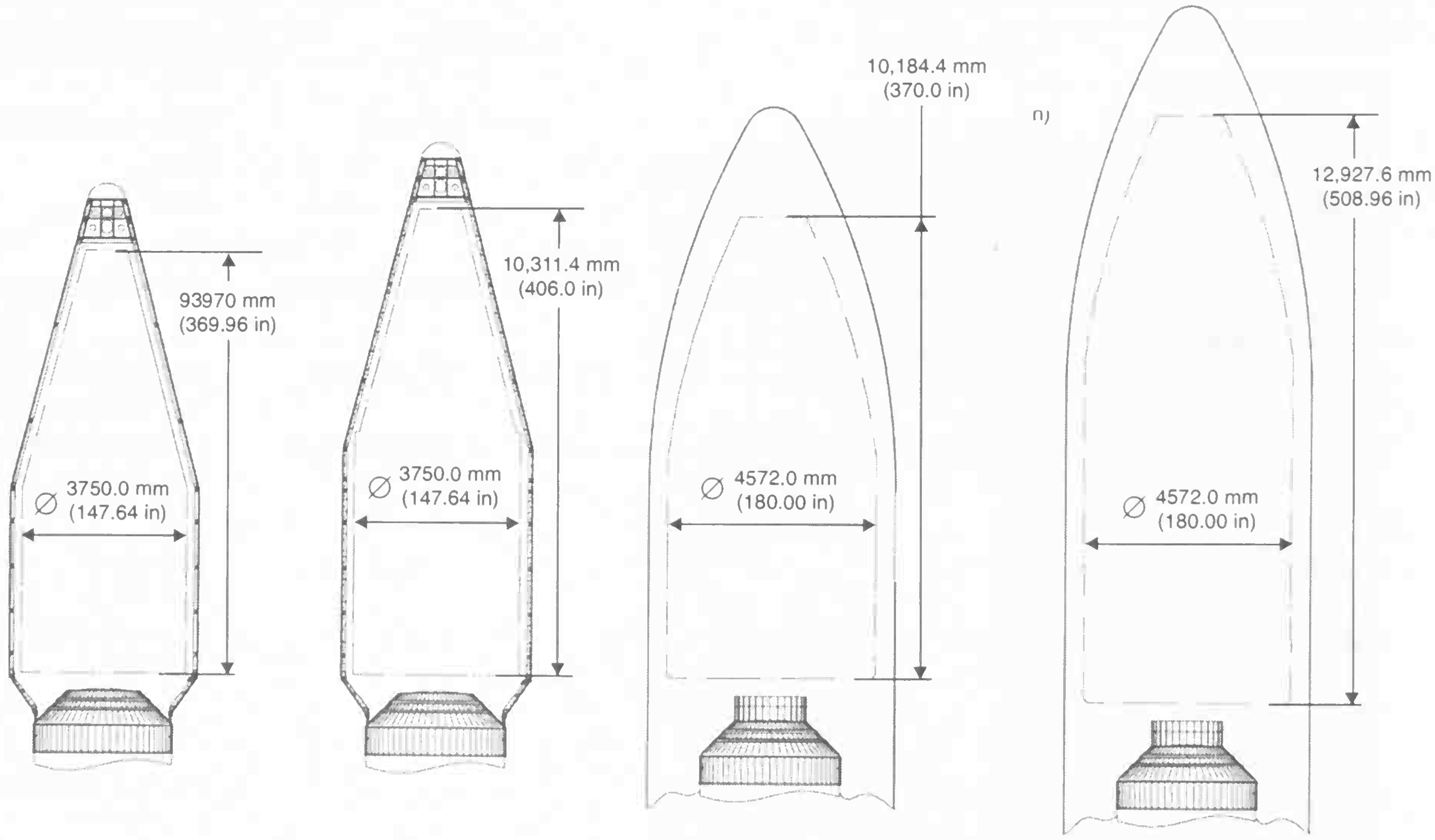
During main engine burns, the Centaur stage provides three-axis control by gimbaling the Centaur main engines. DEC uses hydraulic gimbal actuators, while the SEC uses electromechanical actuators. Three-axis control during coast phases, and roll control of the SEC during burns are provided by the Centaur RCS consisting of four settling thrusters and four pitch and four yaw thrusters. All of the reaction control thrusters use hydrazine propellant, which is pressure fed from one or more common storage bottles. The RCS bottles and thrusters are mounted on the lower bulkhead to prevent payload contamination. During the parking orbit coast, propellant settling is accomplished using a four settling thrusters on mode and propellant retention using a two settling thrusters on mode. Before and during pre-Centaur second main engine start, the four settling thrusters on mode is again used. During chilldown, the propellants issuing from the main engines provide the main percentage of the total longitudinal thrust; and gimbaling the Centaur main engines provides the primary pitch and yaw attitude control, with the pitch and yaw thrusters providing roll control and backup pitch and yaw control. During the first main engine burn, the pitch and yaw thrusters are turned on for 3 s to warm them before the parking orbit coast.

Current missions may operate the four settling thrusters during the second Centaur burn to burn excess N_2H_4, (if any), and, thus, increase performance. Various mission-specific combinations of reaction control thruster operation are used for spacecraft separation, attitude control, and for the Centaur collision/contamination avoidance maneuver (CCAM).

Payload Fairings

The fairings encapsulate the payload to provide a clean, controlled environment, and protect the spacecraft and Centaur avionics from aerodynamic loads and heating during ascent. To prevent spacecraft contamination, the fairings are cleaned before integration, and use noncontaminating separation systems. Options such as acoustic or thermal blankets and payload access doors are available on each fairing. The fairing halves are separated with pyrotechnic bolts and spring thrusters.

The Atlas IIAS, Atlas III series, and Atlas V 400 series uses two fairings: The Large Payload Fairing (LPF) and Extended Payload Fairing (EPF) are 4-meter-class fairings made of aluminum. The EPF uses the same design as the LPF, but is 1 m (3 ft) longer. An extended EPF fairing designated XEPF is under development, with an additional 1 m (3 ft) stretch. The Atlas V 500 uses a new, larger diameter payload fairing produced by Contraves of Switzerland. Contraves builds a similar payload fairing for the Ariane 5. The fairing is a graphite composite structure that covers not only the payload but also the Centaur upper stage, connecting instead to a boattail at the top of the first stage. The fairing is separated into two halves using pyrotechnic cord. The short and medium length fairings are available for the Atlas V 500, while the long fairing is designed for the Atlas V HLV. Acoustic absorbers and fairing access doors are standard.



Payload Fairing	**LPF**	**EPF**	**5-m Short payload fairing**	**5-m Medium payload fairing**
Length	12.0 m (39.4 ft)	12.9 m (42.4 ft)	20.7 m (67.9 ft)	23.4 m (76.8 ft)
Primary Diameter	4.2 m (13.7 ft)	4.2 m (13.7 ft)	5.4 m (17.7 ft)	5.4 m (17.7 ft)
Mass	2085 kg (4600 lbm)	2255 kg (4970 lbm)	4085 kg (9000 lbm)	4649 kg (10,250 lbm)
Sections	2	2	2	2
Structure	Skin-stringer	Skin-stringer	Composite sandwich	Composite sandwich
Material	Aluminum	Aluminum	Graphite–epoxy face sheets over aluminum honeycomb core	Graphite–epoxy face sheets over aluminum honeycomb core

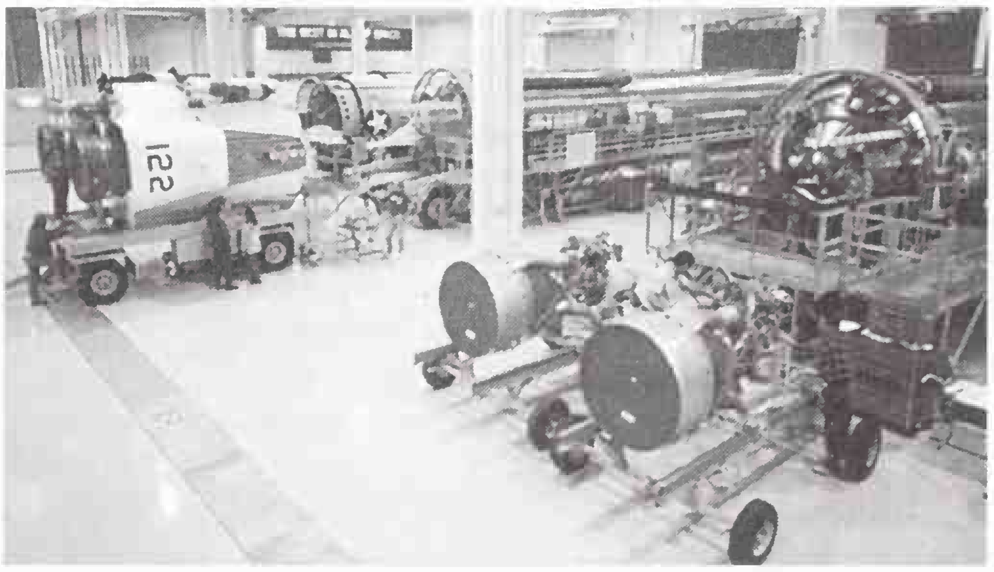
Payload Accommodations

	Atlas IIAS/Atlas III	Atlas V
Payload Compartment		
Maximum Payload Diameter	3750 mm (147.6 in.)	LPF and EPF: 3750 mm (147.6 in.) 5-m Fairing: 4572 mm (180.0 in.)
Maximum Cylinder Length	LPF: 4100 mm (161.5 in.) EPF: 5015 mm (197.5 in.)	LPF: 4100 mm (161.5 in.) EPF: 5015 mm (197.5 in.) 5-m Short: 4888 mm (192.5 in.) 5-m Medium: 7631 mm (300.5 in.)
Maximum Cone Length	5296 mm (208.5 in.)	LPF and EPF: 5296 mm (208.5 in.) 5-m Fairing: 5296 mm (208.4 in.)
Payload Adapter Interface Diameter	Type A: 937 mm(36.9 in.) Type B: 1147 mm (45.2 in.) Type D: 1666 mm (65.6 in.)	Type B1: 1215 mm (47.0 in.) Type D: 1666 mm (66.0 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–18 to 21 months	T–18 to 21 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–4 s	?
On-Pad Storage Capability	Months without propellant	Months without propellant
Last Access to Payload	T–3 h	?
Environment		
Maximum Axial Load	+6.0 *g*	+6.0 *g*
Maximum Lateral Load	±2.0 *g*	±2.0 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz / 15 Hz	8 Hz/15 Hz
Maximum Acoustic Level (Without Blankets)	Atlas IIAS: 131 dB at 200 Hz Atlas IIIA/IIIB: 132 dB at 200 Hz	Atlas V 400: 131 dB at 200 Hz Atlas V 500: 126 dB at 125 Hz Atlas V 550: 131 dB at 125 Hz
Overall Sound Pressure Level (Without Blankets)	Atlas IIAS: 140.5 dB Atlas IIIA/IIIB: 140.8 dB	Atlas V 400: 140.5 dB Atlas V 500: 136.3 dB Atlas V 550: 140.3 dB
Maximum Flight Shock	3000–4000 *g* at 1500 Hz depending on adapter	3000–5000 *g* at 800–1500 Hz depending on adapter
Maximum Dynamic Pressure on Fairing	Atlas IIAS: 36 kPa (750 lbf/ft^2) Atlas IIIA/IIIB: 29 kPa (600 lbf/ft^2)	43.1 kPa (900 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	< 7 kPa/s (1 psi/s)	< 7 kPa/s (1 psi/s)
Cleanliness Level in Fairing	Class 100,000	Class 100,000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	LEO: 1111 km (600 nmi) ± 19.4 km (10.5 nmi) semimajor axis 63.4 ± 0.15 deg inclination GTO: 167 km (90 nmi) ± 2.4 km (1.3 nmi) perigee 35,941 km (19,407 nmi) ± 117 km (63.2 nmi) apogee 27.0 ± 0.02 deg inclination	LEO: 1111 km (600 nmi) ± 19.4 km (10.5 nmi) semimajor axis 63.4 ± 0.15 deg inclination GTO: 167 km (90 nmi) ± 2.4 km (1.3 nmi) perigee 35,941 km (19,407 nmi) ± 117 km (63.2 nmi) apogee 27.0 ± 0.02 deg inclination
Attitude Accuracy (3 sigma)	±0.7 deg; ±0.2 deg/s in each axis	±0.7 deg; ±0.2 deg/s in each axis
Nominal Payload Separation Rate	0.3–0.9 m/s (1.0–3.0 ft/s)	0.3–0.9 m/s (1.0–3.0 ft/s)
Deployment Rotation Rate Available	0–5 rpm	0–5 rpm
Loiter Duration in Orbit	Limited	Hours
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes
Multiple/Auxiliary Payloads		
Multiple Manifest	Atlas usually carries single payloads, and does not carry dual payloads for separate customers. However, multiple payloads for a single customer (e.g., LEO constellations) can be accommodated.	Atlas V will usually carry single payloads or multiple payloads for a single customer (e.g. LEO constellations).
Auxiliary Payloads	Atlas does not generally carry secondary payloads. However, small secondary experiments, such as the Falcon Gold payload, are occasionally launched.	Atlas V will not generally carry auxiliary payloads.

Production and Launch Operations

Production

Atlas is assembled and tested in the final assembly building at Lockheed Martin Space Systems' Waterton facility south of Denver, Colorado. Production was moved to Denver from San Diego, California, in 1995, following Lockheed Martin's purchase of General Dynamic's Space Systems Division. Propellant tankage for Centaur is still manufactured in the Naval In-Service Engineering (NISE) West Facility in San Diego. There, thin steel sheets are welded into 3-m (10-ft) diameter rings, which are then stacked and welded together to form the required tank lengths. Tank domes and propellant lines are also assembled there. A manufacturing facility in Harlingen, Texas, produces structural components such as fairings, interstages, and the first-stage thrust sections. In Denver, the tanks and structural elements are integrated with the propulsion systems, avionics, wiring harnesses, and pneumatic components. Tanks for the Atlas V first stage are also assembled in Denver, where milled isogrid panels are welded together with tank domes and intertank assemblies. Acceptance testing of the flight software, electrical systems, and pressurized components is conducted on the assembled stages before they are shipped in a USAF C-5 or a commercial An-124 cargo aircraft to the launch site. Some components, such as fairings and interstages, are shipped directly from Harlingen to the launch site.



Courtesy Lockheed Martin Corporation.

Atlas Assembly

Production of the newer Atlas III and Atlas V involves several international suppliers. The RD-180 engine for both vehicles is provided by RD AMROSS, a joint venture of Pratt & Whitney of the United States and Russian engine maker NPO Energomash. Contraves of Switzerland provides the 5-meter payload fairings for the Atlas V 500 series. CASA of Spain manufactures the composite interstages.

Organization	**Responsibility** Atlas IIAS	Atlas III	Atlas V
Lockheed Martin Space Systems	Prime contractor Stages 1 and 2, payload fairing Mission integration Launch operations	Prime contractor Stages 1 and 2, payload fairing Mission integration Launch operations	Prime contractor Stages 1 and 2, payload fairing Mission integration Launch operations
BF Goodrich	Data acquisition system	Data acquisition system	Data acquisition system
Boeing Rocketdyne Propulsion and Power	MA-5A Stage 1 engine		
CASA			Composite interstage structures
Contraves			5-m payload faring
ATK Thiokol	Castor IVA strap-on booster		
GenCorp Aerojet			Strap-on booster
Honeywell	Inertial navigation unit	Inertial navigation unit	Inertial navigation unit
Marconi Integrated Systems, Inc.	Avionics components	Avionics components	Avionics components
Pratt & Whitney	RL10 Stage 2 engine	RL10 Stage 2 engine	RL10 Stage 2 engine
RD AMROSS		RD-180 Stage 1 engine	RD-180 Stage 1 engine
SAAB	Payload separation systems	Payload separation systems	Payload separation systems

Production and Launch Operations

Launch Operations—Cape Canaveral Air Force Station

SLC-36

The Atlas II and III launch facility at Cape Canaveral Air Force Station (CCAFS), Florida, is Space Launch Complex 36, which has two launch pads. The pad at SLC-36A can launch Atlas II series vehicles, while SLC-36B has been modified to accommodate both Atlas II and III vehicles. Each pad has a mobile service tower (MST) and a fixed umbilical tower (UT). Both pads share a blockhouse.

The MST is an open steel structure with an interior enclosure that contains retractable vehicle servicing and checkout platforms. The tower contains an electric, trolley-mounted overhead bridge crane that is used to hoist the upper stage, spacecraft, and fairing into position. The crane at SLC-36A is rated at 18 t (40 klbm), while the crane at SLC-36B is rated at 27 t (60 klbm). Two elevators serve all MST levels. The entire MST assembly is on a rail system, which allows it to be moved for launch. Radio frequency cabling and reradiating antennas are mounted on the service tower for spacecraft use.

The UT is a fixed steel tower extending above the launch pad. Retractable service booms are attached to the UT. The booms supply electrical power, instrumentation, propellants, pneumatics, and conditioned air or gaseous nitrogen (GN_2) to the launch vehicle and spacecraft. These systems also provide quick-disconnect mechanisms at the vehicle interfaces and permit boom retraction at vehicle launch. The UT also provides a wind damper system that reduces wind-induced lateral oscillations of the vehicle during periods of service tower removal. The ground wind damper mechanism is released from the vehicle during the launch sequence.

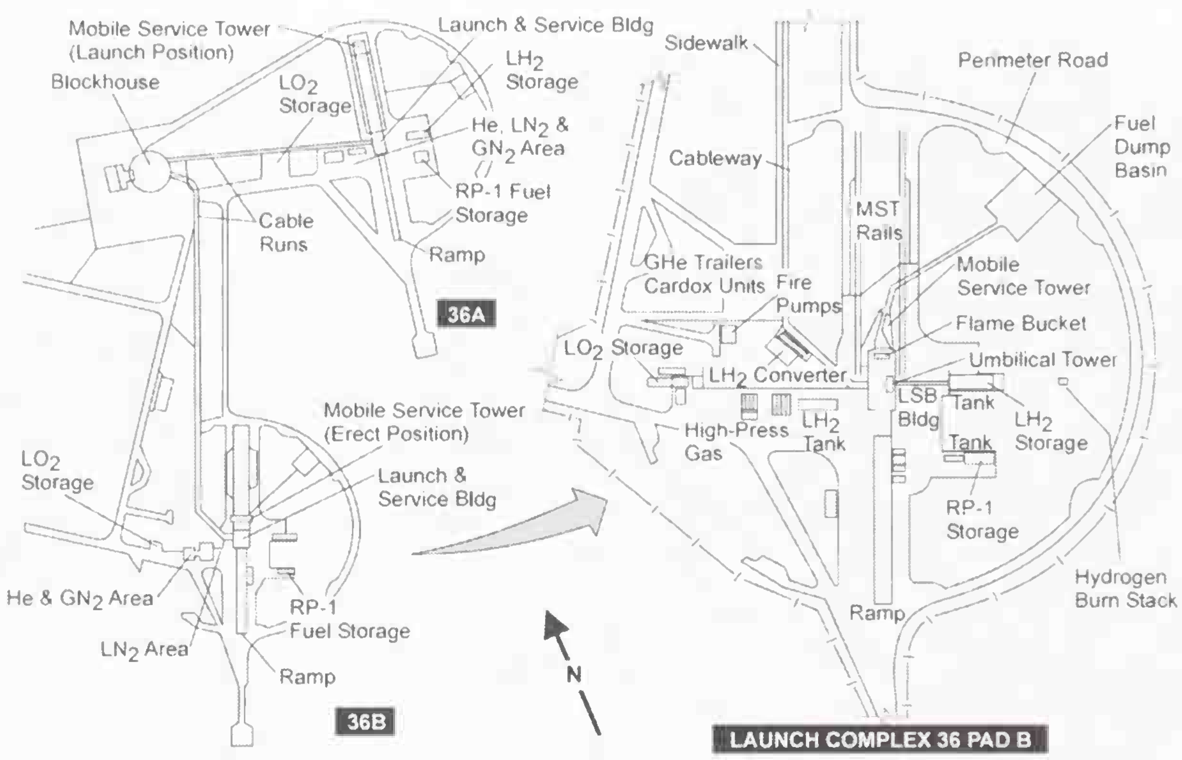
The launch complex is serviced by GN_2, gaseous helium (GHe), and propellant storage facilities within the complex area. Environmental control systems (ECS) exist for both the launch vehicle and the spacecraft.

The blockhouse serves as the operations and communications center for the launch complex. It contains all necessary control and monitoring equipment. The launch control, electrical, landline instrumentation, and ground computer systems are the major systems in this facility.

Spacecraft processing for commercial Atlas launches at CCAFS is typically performed at the Astrotech payload processing facilities near Titusville, Florida. Astrotech is a division of Spacehab, and provides spacecraft processing services and facilities for many payloads launched from CCAFS and the KSC. Astrotech facilities include areas for controlled storage, fueling, ordnance installation, spin balancing, and testing of spacecraft. If necessary, additional NASA and USAF facilities can also be used for spacecraft processing.

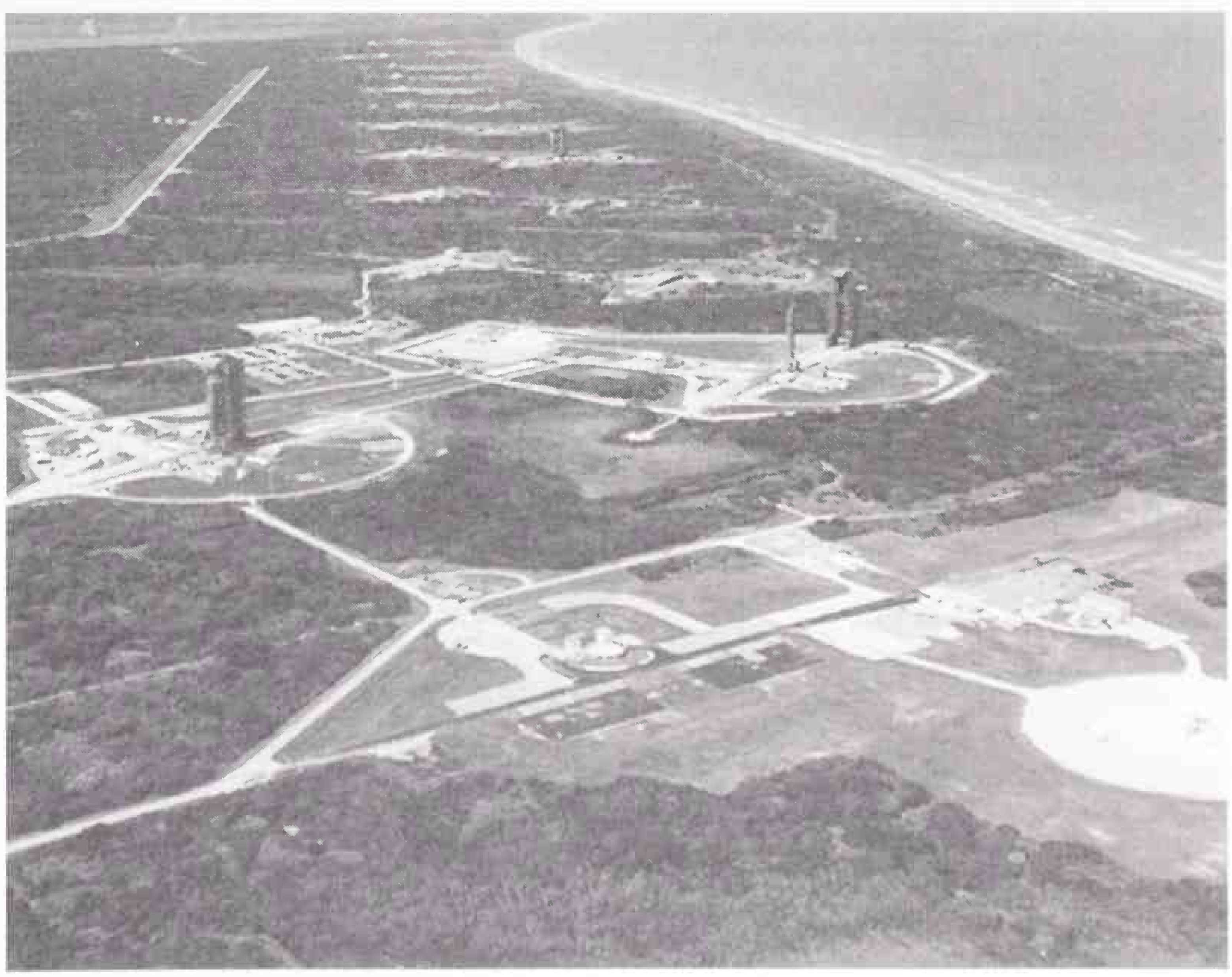
With the final launch of the Atlas III, scheduled for January 2005, launch operations at SLC-36 will cease.

See the Spaceports chapter for further information.



Courtesy Lockheed Martin Corporation.

Cape Canaveral Air Force Station, SLC-36 Layout



Courtesy Lockheed Martin Corporation.

Cape Canaveral Air Force Station, SLC-36

Production and Launch Operations

SLC-41

Atlas V launches are conducted from SLC-41. The launch complex was originally used for Titan III and IV launches. The umbilical tower and service tower were demolished in 1999 to make way for the new Atlas V "clean pad." Atlas vehicles are assembled and tested vertically in the 87 m (287 ft) tall Vertical Integration Facility, adjacent to the launch complex. The VIF has a 54,000-kg (60-ton) crane for stacking vehicle elements. The spacecraft is hoisted into position three days before launch. The day before launch the launch vehicle is rolled out to the launch pad on the 700-ton (1.5-million-lb) Mobile Launch Platform (MLP). The MLP includes an umbilical mast that provides all fluid, electric, and conditioned air connections to the launch vehicle. The pad itself is a simplified facility, and includes a flame trench, equipment vaults, four lightning protection towers, and tank farms for propellants and gases. There is no launch pad umbilical tower or mobile service tower as is typically used for launch vehicles of this size.

Launches are controlled from the new Atlas V Spaceflight Operations Center (ASOC), 6.6 km (4.1 mi) from the launch pad. The ASOC combines a high bay, where Atlas V elements are received for testing before delivery to the VIF, a Launch Operations Center, a customer viewing area overlooking the Launch Operations Center, secure customer work areas, and hospitality areas for viewing the launch.

Atlas V launch facilities at CCAFS SLC-41. In the lower left is the VIF. The mobile launch platform is shown in the launch position in the center, surrounded by four lightning towers.

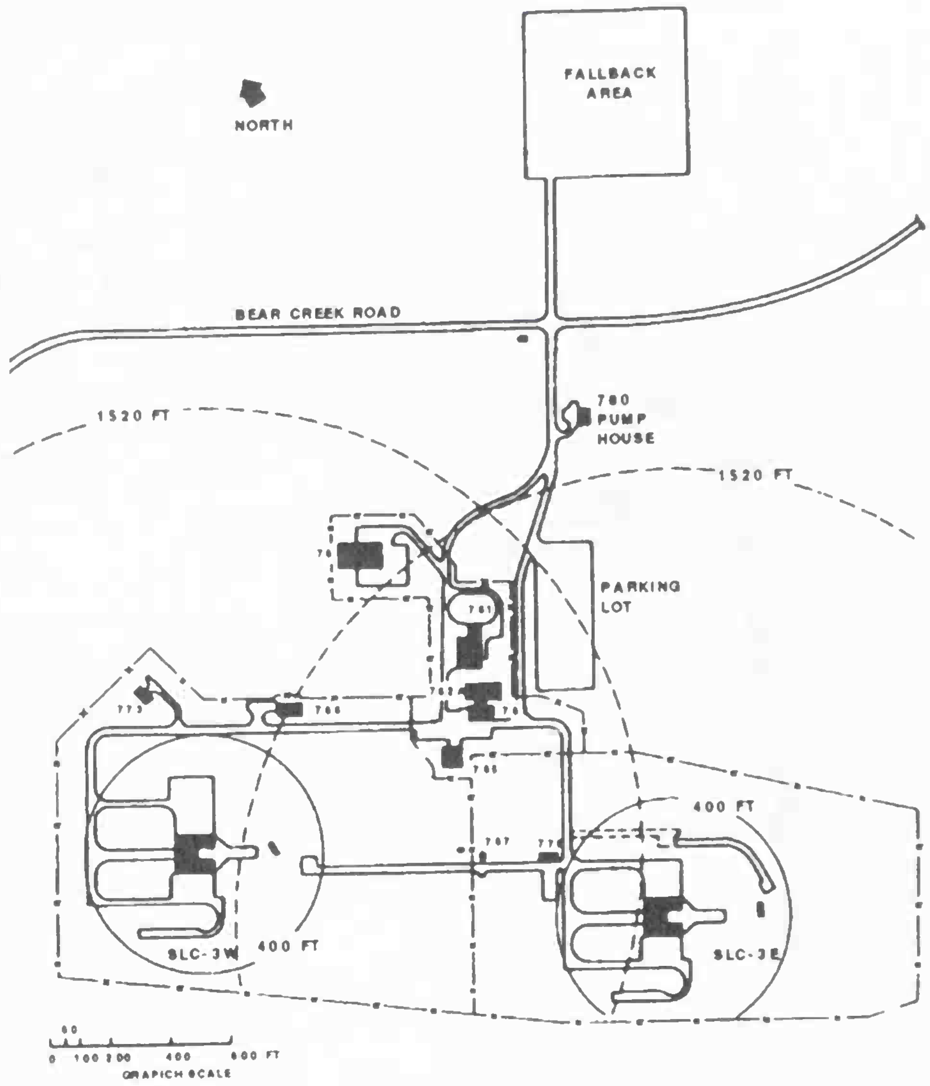
Production and Launch Operations

Launch Operations—Vandenberg Air Force Base

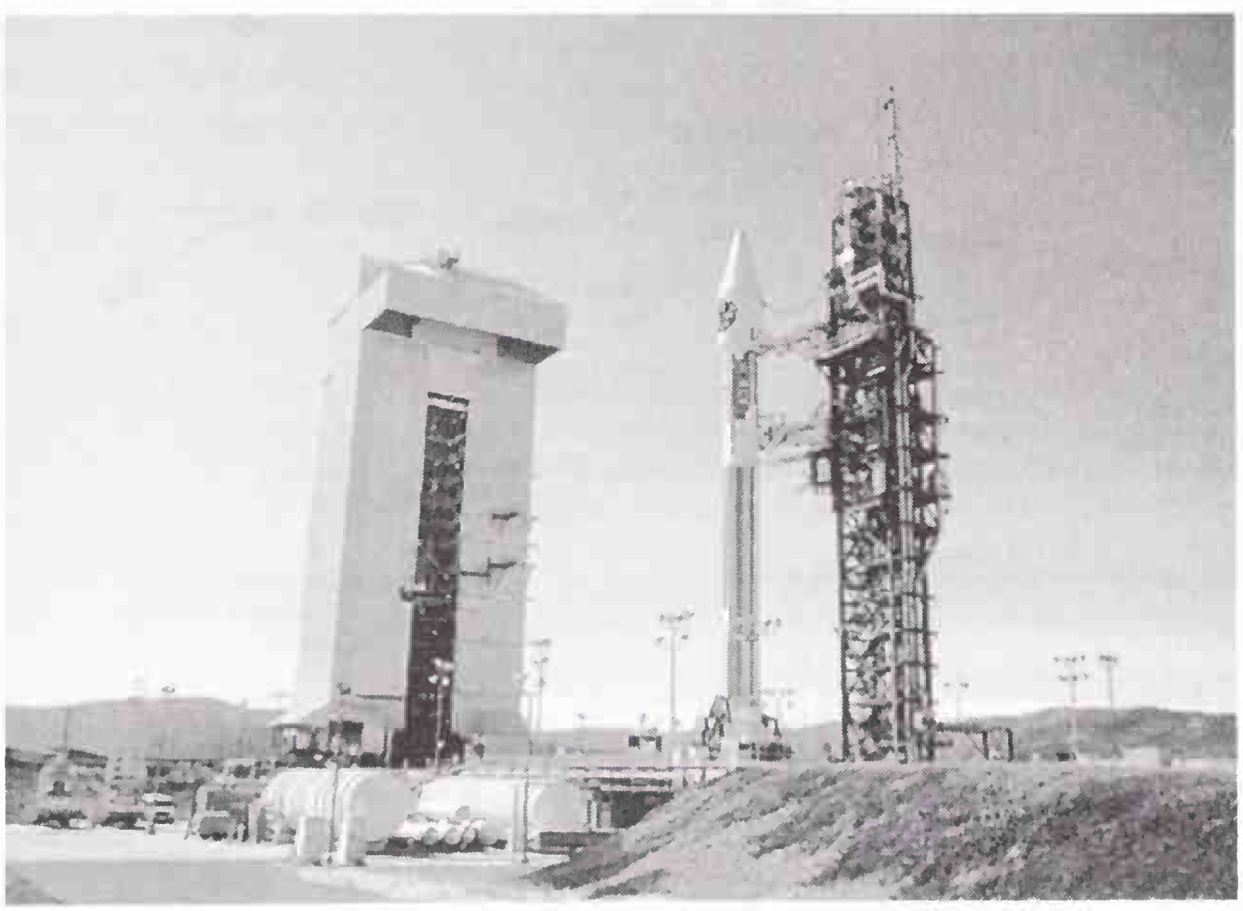
Atlas launches from Vandenberg Air Force Base (VAFB), California, take place at Space Launch Complex 3 on South VAFB. SLC-3W supported launches of refurbished Atlas ICBMs until the last remaining Atlas E vehicle lifted off in 1995. The SLC-3W pad has since been deactivated and demolished. Under contract to the U.S. Air Force, Lockheed Martin modified SLC-3E to support Atlas/Centaur launches from VAFB for the first time. The pad has supported three launches to date, beginning in December 1999 when an Atlas IIAS carried a NASA Earth Observing System satellite to orbit. SLC-3E is designed only for the Atlas II series vehicles. By the end of 2005 modifications to SLC-3E will make it able to accommodate Atlas V launches.

The SLC-3E facilities include an MST, UT, and the launch and services building (LSB). The launch control center has been relocated from the nearby blockhouse to a Remote Launch Control Center (RLCC) in Building 8510 in the northern section of VAFB. The MST is a fully enclosed steel structure mounted on rails that allow it to roll back before launch. The MST has an 18-t (40-klbm) overhead bridge crane to lift the Centaur and payload into place, as well as an environmentally controlled area to service the Centaur and payload. The UT provides retractable umbilicals for power, command and control cabling, propellant and gas lines, and conditioned air. It also incorporates a ground wind damper to stabilize the vehicle during high winds. The LSB is a large concrete and steel structure that serves as a foundation below the MST and UT. The upper surface is the pad deck where the launch vehicle is assembled. The LSB includes protected areas for shop areas, storage, launch support equipment, and utility equipment for power, air conditioning, and fluids and gases.

Payload processing at VAFB can be conducted at an Astrotech facility near the VAFB airfield.



Vandenberg Air Force Base, SLC-3 Layout

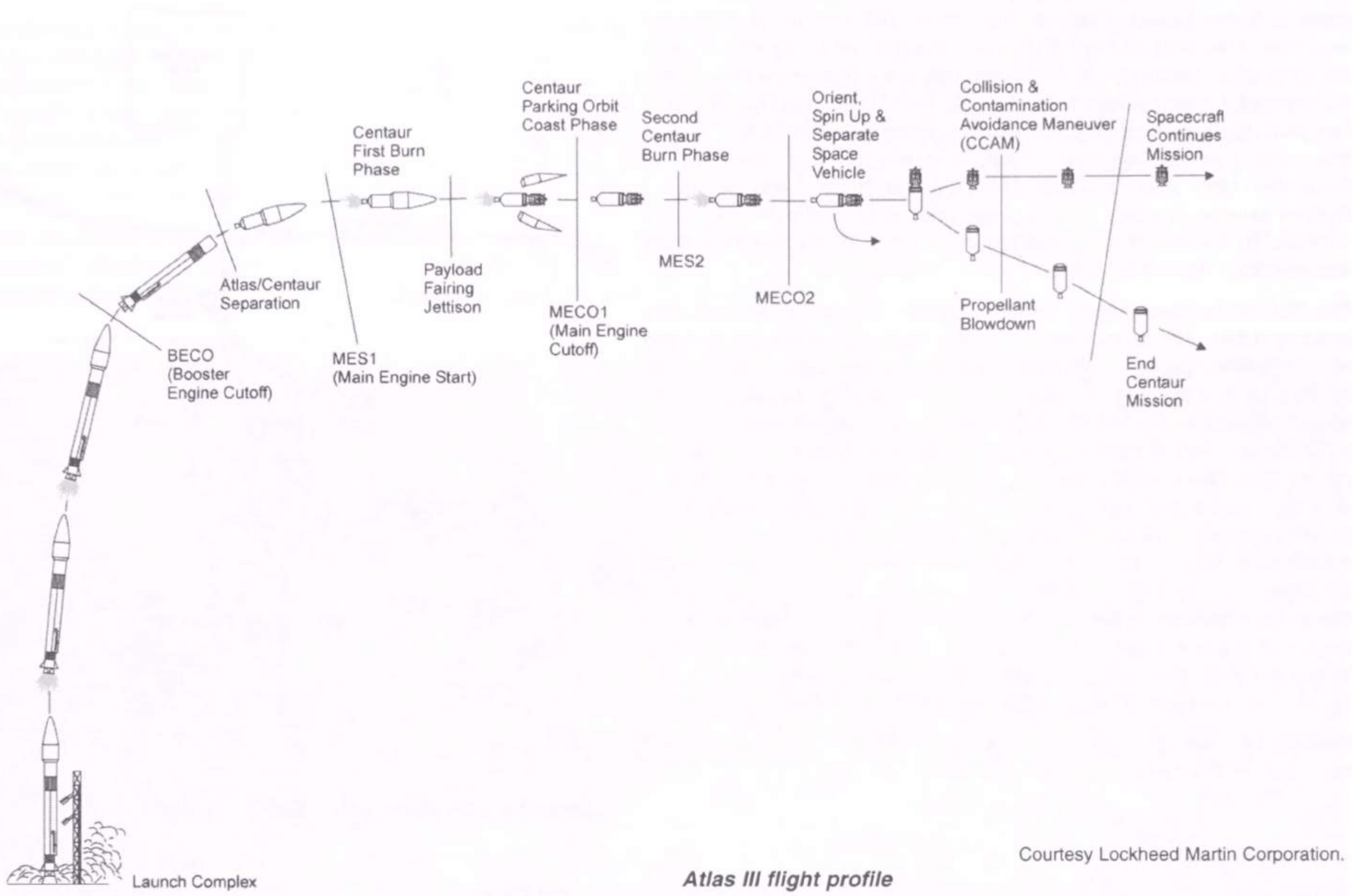


Courtesy Lockheed Martin Corporation.

Vandenberg Air Force Base, SLC-3

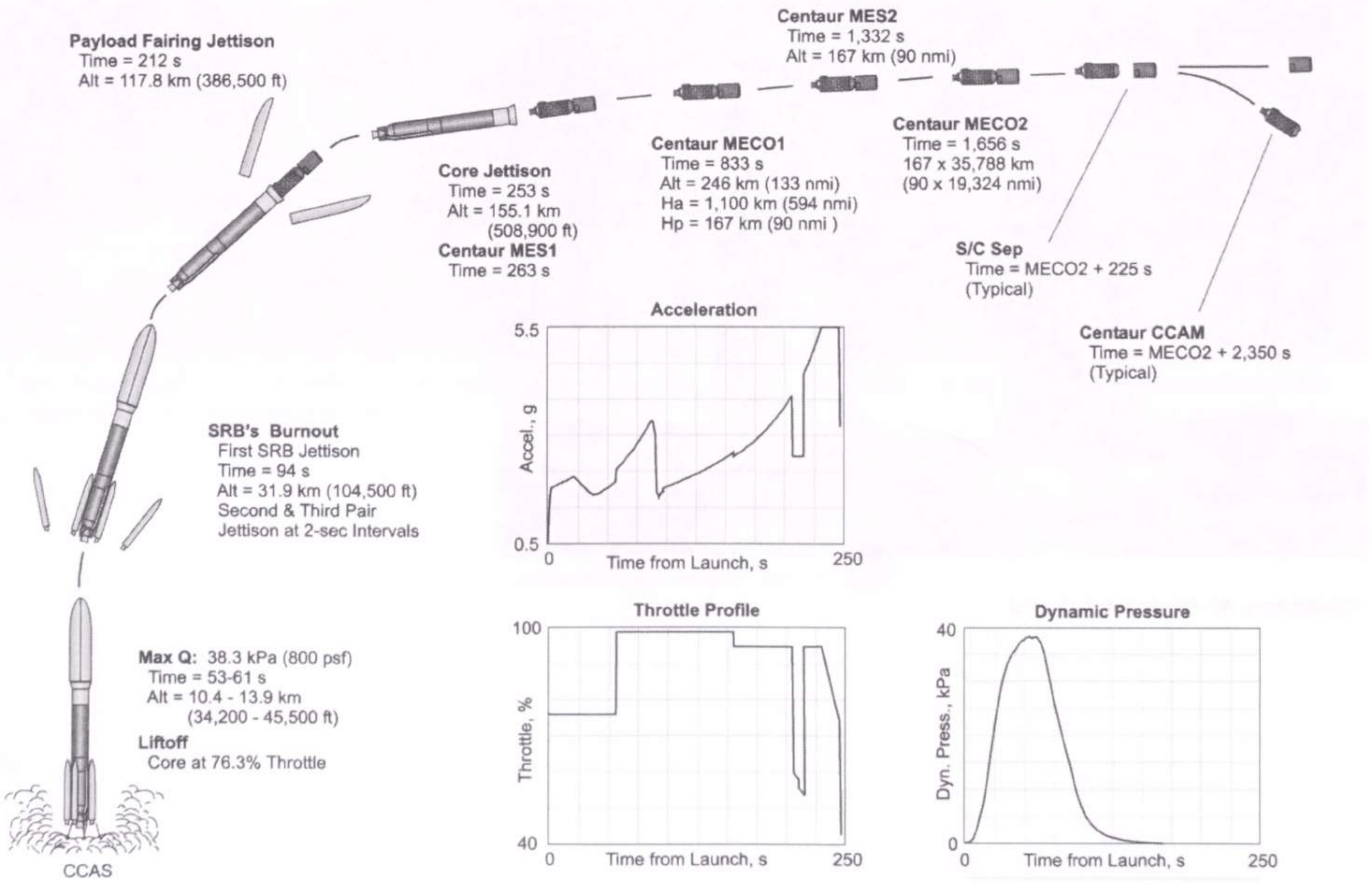
Production and Launch Operations

Flight Sequence



Courtesy Lockheed Martin Corporation.

Atlas III flight profile



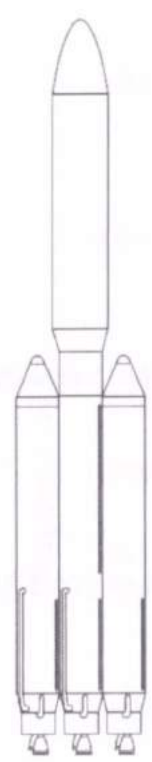
Courtesy Lockheed Martin Corporation.

Atlas V flight profile

Production and Launch Operations

Event (Time, s)	Atlas IIAS	Atlas IIIA	Atlas IIIB DEC	Atlas V 401	Atlas V 551
Guidance Go Inertial	−11.0	−11.0	−11.0	−11.0	−11.0
Ground-Lit SRB Ignition	−0.5	—	—	—	?
Liftoff	0.0	0.0	0.0	0.0	0.0
Core Throttle Up	—	—	—	—	58.0
Ground-Lit SRB Burnout	54.7	—	—	—	?
Air-Lit SRB Ignition	60.0	—	—	—	—
Ground-Lit SRB Jettison	77.1	—	—	—	94.0
Air-Lit SRB Burnout	115.3	—	—	—	—
Air-Lit SRB Jettison	117.2	—	—	—	—
Atlas Booster Engine Cutoff	163.3	184.0	182.0	236.0	248.0
Booster Package Jettison	166.4	—	—	—	—
Payload Fairing Jettison	214.5	—	—	—	212.0
Atlas Sustainer Engine Cutoff	289.2	—	—	—	—
Atlas/Centaur Separation	293.3	195.0	193.0	241.0	253.0
Begin Extendible Nozzle Deployment (When Used)	294.8	196.5	194.5	?	?
End Extendible Nozzle Deployment	301.8	203.5	201.5	?	?
Centaur Main Engine Start	309.8	212.9	202.9	251.0	263.0
Payload Fairing Jettison	—	220.9	225.5	259.0	—
Centaur Main Engine Cutoff	584.8	767.7	541.7	930.0	833.0
Start Turn to Main Engine Start 2 Attitude	1180.8	1101.2	982.0	1162.0	992.0
Centaur Main Engine Start	1475.8	1401.2	1380.0	1502.0	1332.0
Centaur Main Engine Cutoff	1571.9	1585.3	1494.8	1723.0	1656.0
Start Alignment to Separation Attitude	1573.9	1587.3	1496.8	1733.0	1666.0
Begin Spinup	1691.9	1717.3	1626.8	1843.0	?
Separate Spacecraft	1798.9	1810.3	1719.8	1948.0	1881.0
Start Turn to CCAM Attitude	1918.9	1930.3	1839.8	2048.0	1981.0
Centaur End of Mission	4239.9	4250.3	4159.8	4073.0	4006.0

Vehicle Upgrade Plans

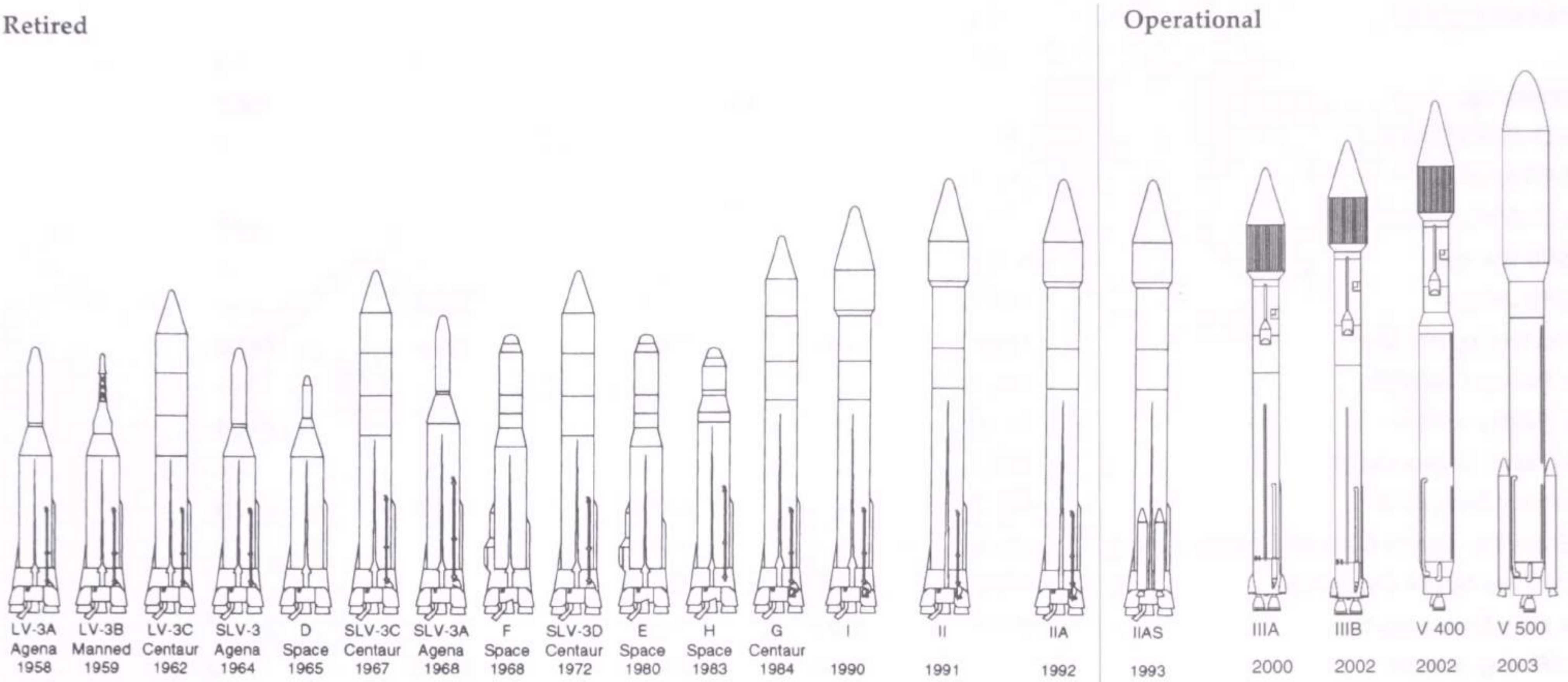


Atlas V Heavy

The EELV program initially included development of a Titan IV-class Heavy Lift Vehicle (HLV) configuration. Lockheed Martin postponed the development of the HLV after completing much of the initial design work but continues to study the vehicle. The HLV would add two liquid strap-on boosters, based on the first stage, to provide a substantial increase in performance. The Atlas V HLV could lift 12,650 kg (27,880 lbm) to GTO, or deliver 6350 kg (14,000 lbm) directly to GEO.

VEHICLE HISTORY

Vehicle Evolution

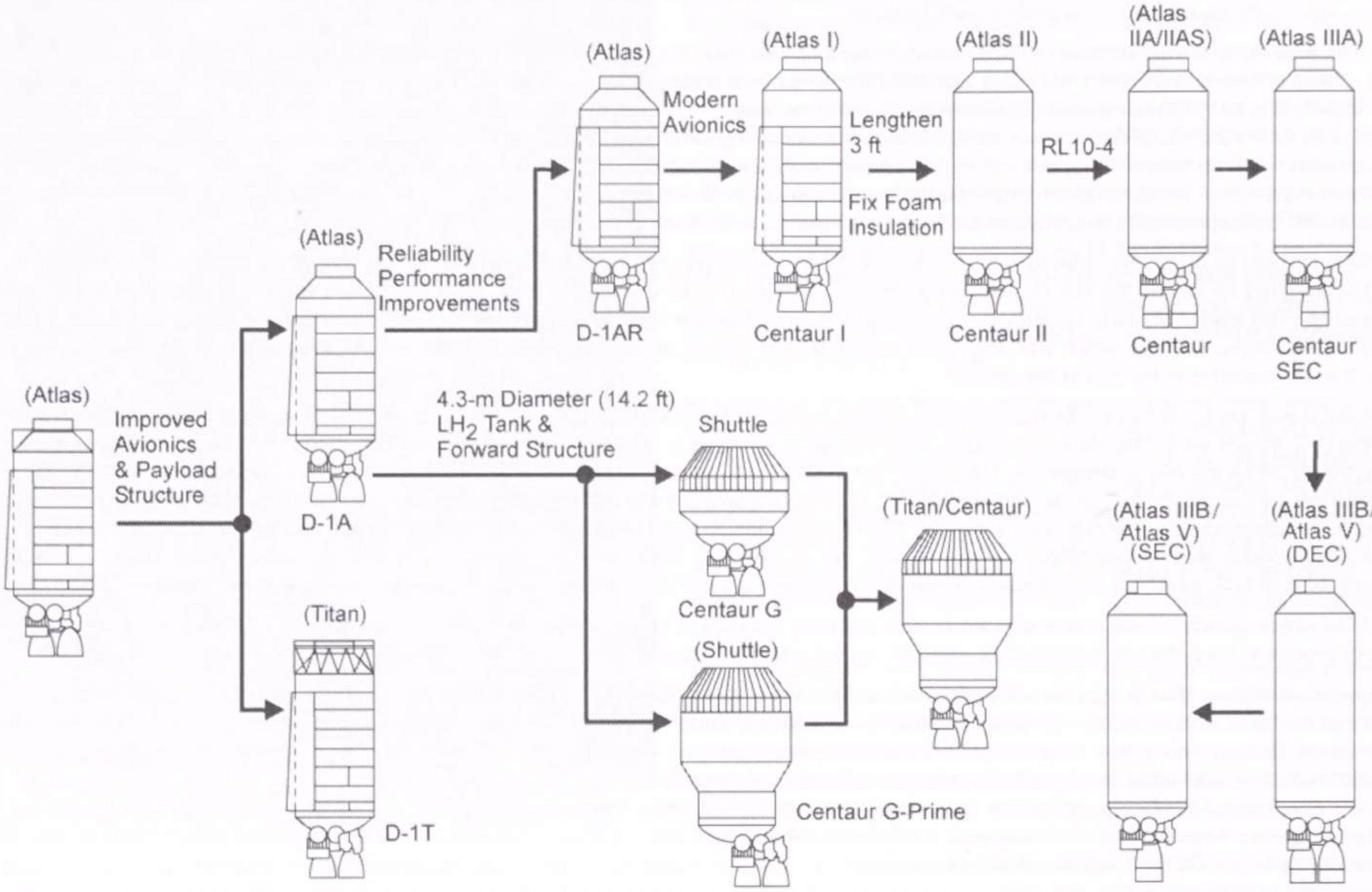


Vehicle Description

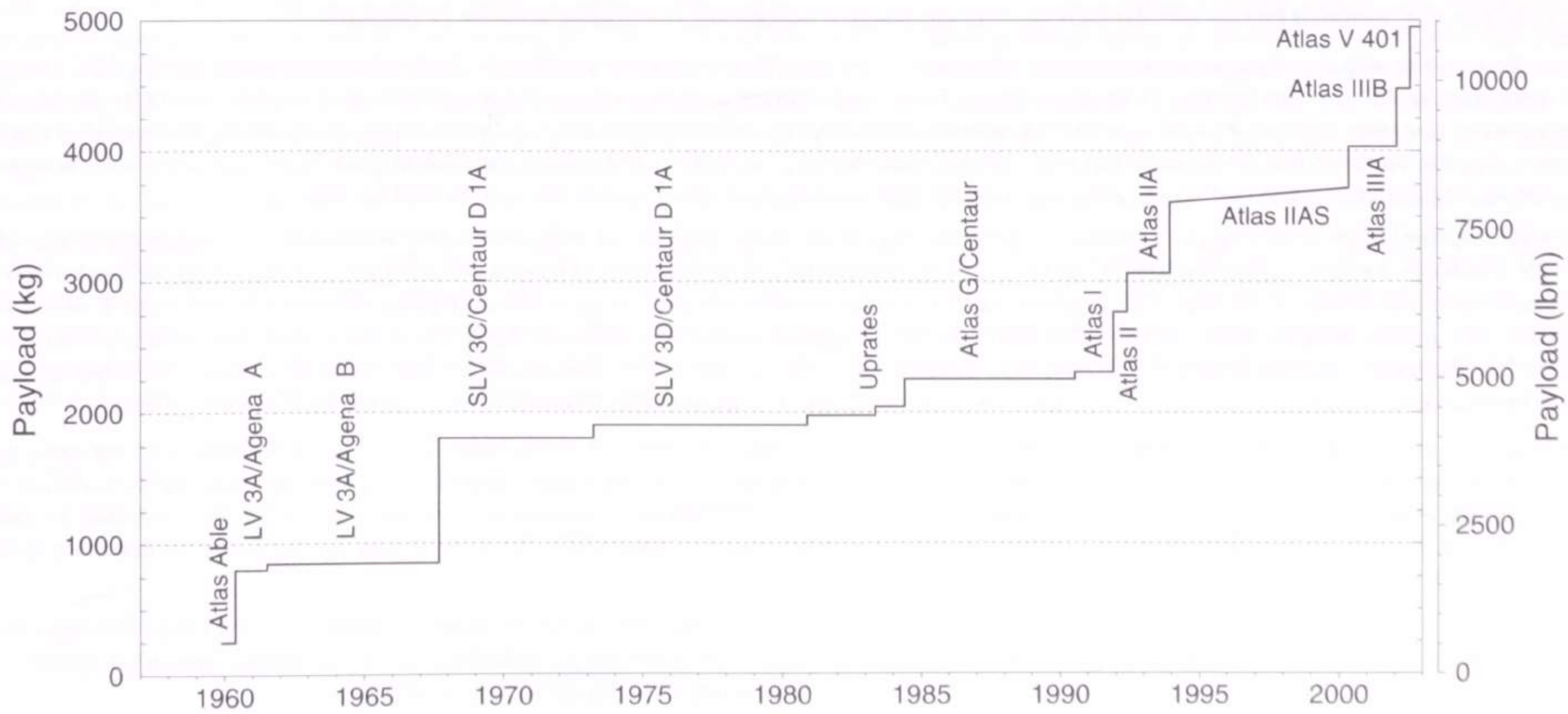
•A	ICBM single-stage test vehicle
•B and C	ICBM 1-1/2 stage test vehicles
•D	Operational ICBM and later space launch vehicle
•E and F	First ICBM (1960), then reentry test vehicles (1964), then space launch vehicles (1968)
•LV-3A	Same as D with added Agena upper stage
•LV-3B	Same as D except human-rated for project Mercury
•SLV-3	Same as LV-3A except reliability improvements
•SLV-3A	Same as SLV-3 except stretched 2.97 m (117 in)
•LV-3C	Launched with Centaur D upper stage
•SLV-3C	Same as LV-3C except stretched 1.3 m (51 in)
•SLV-3D	Same as SLV-3C except Centaur uprated to D-1A and Atlas electronics integrated with Centaur
•G	Same as SLV-3D but Atlas longer 2.06 m (81 in)
•H	Same as SLV-3D except with E / F avionics and no Centaur upper stage.
•I	Same as G except strengthened for 4.27-m (14-ft) payload fairing and ring laser gyro added
•II	Same as I except Atlas lengthened 2.74 m (108 in), engines uprated, added hydrazine roll control, fixed foam insulation, deleted verniers, and Centaur stretched 0.9 m (36 in.)
•IIA	Same as II except Centaur RL10 engines uprated to 88 kN (20 klbf) of thrust and 6.5 s Isp increase from extendible RL10 nozzles
•IIAS	Same as IIA except 4 Castor IVA strap-on boosters added
•IIIA	Similar to IIA, with new RD-180 first-stage engine, first-stage booster-sustainer separation eliminated, first stage stretched 4.4-m (14.5-ft), single-engine Centaur.
•IIIB	Same as IIIA, except single- or dual-engine Centaur, and Centaur extended 1.7 m (5.5 ft)
•V 400	Similar to IIIA with new common core booster first-stage structure
•V 500	Similar to V 400 with optional solid strap-on boosters, Centaur encapsulated inside 5.4-m (17.7-ft) payload fairing

VEHICLE HISTORY

Centaur Genealogy



Courtesy Lockheed Martin Corporation



Historical Growth in Atlas GTO Performance

Vehicle History

Historical Summary

The Atlas space launch vehicles evolved from the successful Atlas ICBM. While the Atlas has grown dramatically and evolved over the years, many of its features can be traced back to its early missile heritage.

The Atlas ICBM project was initiated by General Dynamics as project MX-774 for the U.S. Air Force in 1945. After being canceled in 1947 for lack of funds, it was reinstated four years later. The ICBM underwent a major scaling down in 1955 as a result of breakthroughs in thermonuclear weapons and made its first test flight two years later. From 1957 to 1959, research and development on Atlas produced the A, B, and C versions. A modified Atlas B was used for Project SCORE on one of its 10 successful tests during 1958 and 1959. Project SCORE, the first of many notable space launch missions performed by an Atlas vehicle, was the world's first communications satellite. Launched in 1958, this satellite transmitted President Eisenhower's Christmas message. Atlas development continued with an improved guidance system on the Atlas C. The Atlas A, B, and C versions had a total of 23 research and development flights and led to the first operational Atlas flight using the Atlas D in 1959.

The Atlas D could be called the "granddaddy" of the current operational system. It was launched more times (123) than any other version of Atlas and was human-rated for use on the historic Mercury program. The Atlas D used a cluster of three engines (two boosters, one sustainer) to comprise its one and one-half stages. Igniting the sustainer engine on the ground before liftoff reduced concern over the reliability of lighting upper-stage engines in flight. This staging scheme, which was first tested on the Atlas B, was to be used on all of the following Atlas vehicles until the development of the Atlas III series vehicles at the end of the century.

The Atlas D was the basis for two distinct branches of Atlas vehicles. One branch contained the Atlas E and F ICBMs, which were used with the Atlas D in the U.S. missile silos. The other branch grew from the use of the Atlas D as a space launch vehicle. The Atlas D was modified, human-rated, and called the LV-3B for the Mercury missions. The 28.8-m (94.5-ft) tall Atlas used the same basic Atlas D system with the addition of a 1400 kg (3000 lbm) crewed Mercury capsule on top. Unlike Project SCORE, in which the payload and Atlas sustainer remained attached, the Mercury payloads were separated to fly independently after orbit was achieved. The Mercury capsule had been tested on suborbital flights launched on Redstone rockets, but the larger Atlas booster was needed to put Mercury in orbit. On 20 February 1962, after a successful launch on the human-rated Atlas D, John Glenn became the first U.S. astronaut to orbit Earth. Seven of the ten Mercury flights were successful, including all four crewed missions.

The Atlas space launch vehicle also was used in all of the early unmanned lunar exploration missions: Ranger, Lunar Orbiter, and Surveyor. Finally, Mariner probes to Mars, Venus, and Mercury, and the Pioneer probes to Jupiter, Saturn, and Venus, were launched by Atlas Centaur vehicles.

As needed, additional Atlas D vehicles were converted to Atlas LV-3A or LV-3C configurations by modifying vehicle structure and subsystems for each mission to be flown. This led to a successful set of launch vehicles and upper stages. LV-3A used the Agena upper stage, and LV-3C used the Centaur upper stage. Centaur was a new upper stage that used cryogenic hydrogen and oxygen for the first time in a launch vehicle. The high-performance Centaur was especially useful for high-energy missions, and would eventually become the standard Atlas upper stage. In 1962, management for both the Atlas Agena and Atlas Centaur programs was transferred to the NASA Lewis Research Center. The LV-3A was involved heavily in the Ranger and Mariner programs; it made a total of 43 successful launches in 53 attempts. The LV-3C's 11 successes in 12 attempts consisted of research and development flights for the Centaur and Surveyor lunar landers.

Unfortunately, the mission tailoring that was required to convert Atlas missiles to space launch vehicles caused long lead times that detracted from their low cost. As a result, a contract was awarded to General Dynamics in 1962 to develop a standardized launch vehicle (SLV). The SLV line began with the SLV-3. This vehicle, like its predecessor the LV-3A, used primarily Agena upper stages. From its first launch in August 1964 to its final launch in August 1968, the SLV-3 was successful on 49 of 51 orbital launch attempts, including all five Lunar Orbiter flights.

In 1965, Convair Division of General Dynamics won a contract to improve the performance of its vehicles, and the SLV-3A and -3C were introduced. Both vehicles had been lengthened to add propellant, had increased engine thrust, and had reduced vehicle weight. They used the Rocketdyne MA-5 engine system with a total thrust of 1.92 MN (431,300 lbf). The radio-guided SLV-3A stood 24.0 m (78.7 ft) tall by itself—3.0 m (9.75 ft) taller than the SLV-3—and 36.0 m (118 ft) tall with an Agena upper stage and payload fairing; it could deliver 3900 kg (8500 lbm) of payload into a 185 km (100 nmi) circular orbit with the Agena. The SLV-3A was used primarily for classified missions and was successful on 11 of 12 flights. The SLV-3C was similar to the SLV-3A, but was designed for use with the Centaur D upper stage. It successfully completed all of its 17 missions.

Because the Centaur was emerging as an exceptional upper stage, the SLV-3C evolved into the SLV-3D. Unlike its predecessor, the SLV-3D was integrated electronically with the new Centaur D-1A upper stage. Thus, the Centaur's autopilot and guidance systems were used to control the launch vehicle, as opposed to earlier vehicles that were guided by an independent Atlas autopilot. Most other systems remained the same as those in preceding SLVs, including the same engine thrust as the SLV-3C. The SLV-3D stood 21.2 m (69.5 ft) by itself and 59.5 m (131 ft) with the Centaur and payload fairing added. All 32 SLV-3D launches used Centaur D-1As; its 30th success took place on the last launch in May 1983.

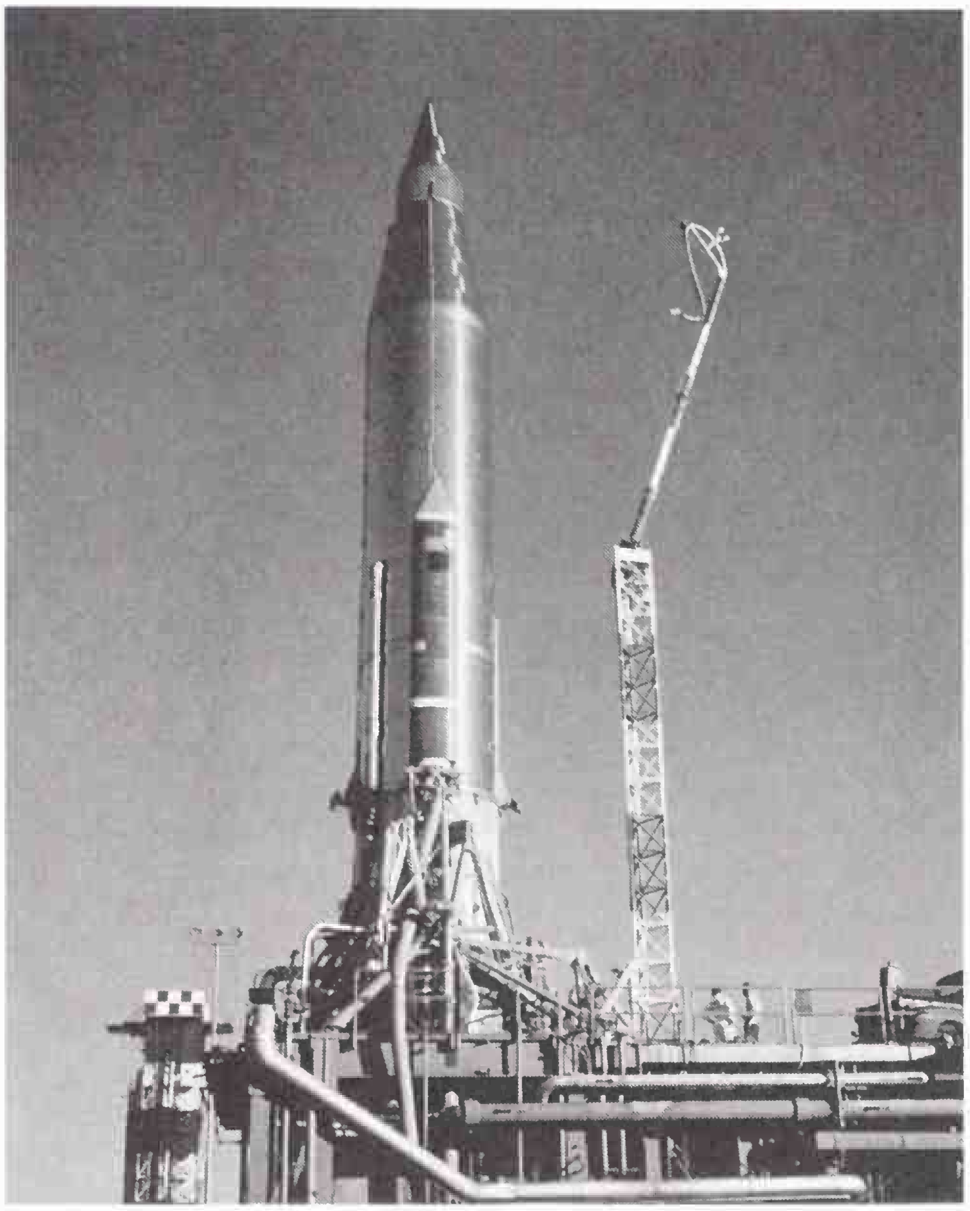
As the Atlas was progressing as a space launch vehicle, it also was developing as an ICBM. Atlas Es and Fs were being developed along with Atlas Ds as U.S. ICBMs in the late 1950s. The Atlas E and F were virtually identical one and one-half stage vehicles that used inertial guidance. A major difference between the E/F vehicle and the SLVs was that the E/F used a Rocketdyne MA-3 engine system instead of the MA-5. The Atlas E and F were deployed in U.S. missile silos for much of the 1960s until they were replaced by the Minuteman in 1965. The Atlas F then was used primarily for the advanced ballistic reentry system (ABRES) program from 1965 to 1974. By the late 1970s, the remaining Atlas Es and Fs were converted for space launch. The Atlas E or F could place 795 kg (1750 lbm) into polar LEO with no upper stage. The last Atlas E launch took place in 1995.

Two vehicles emerged from the SLV-3D: the Atlas H and the Atlas G. The Atlas H used most of the basic SLV systems but employed the radio guidance and avionics of the Atlas E. The MA-5 engines used LOX/RP-1 propellants and provided over 1954 kN (43.3 klbf) of thrust; 1679 kN (377.5 klbf) from two booster engines, 269 kN (60.5 klbf) from a sustainer, and 5950 N (1338 lbf) from two verniers. In place of the Centaur, the Atlas H used a solid propellant kick motor as a second stage to propel up to 2000 kg (4400 lbm) into polar LEO. The Atlas H was successful on all five of its launches; the last launch occurred in 1987.

The Atlas G was a stretched version, (22.2 m [72.7 ft] tall), of its predecessor, the SLV-3D. Its MA-5 engine provided 33.3 kN (7500 lbf) more thrust than the SLV-3D's MA-5. The Atlas G was designed for use with the Centaur D-1A upper stage, including use of the Centaur guidance system. The Atlas G/Centaur combination stood 41.8 m (137 ft) tall and was capable of delivering 2400 kg (5200 lbm) to GTO.

As a result of an unprecedented string of launch vehicle failures and a related decision to remove commercial payloads from the Space Shuttle manifest, General Dynamics decided in 1987 to develop and build 18 Atlas/Centaurs (designated Atlas I) for commercial sale without having firm contracts for their purchase. Two new metal payload fairings were introduced with 3.3-m (11-ft) and 4.2-m (14-ft) diameters. The Atlas I began launching in 1990, but suffered 3 failures in 11 launches before it was phased out in 1997.

Vehicle History



The first Atlas space launch, and the only space launch of an Atlas B, carried the Project SCORE payload in December 1958.

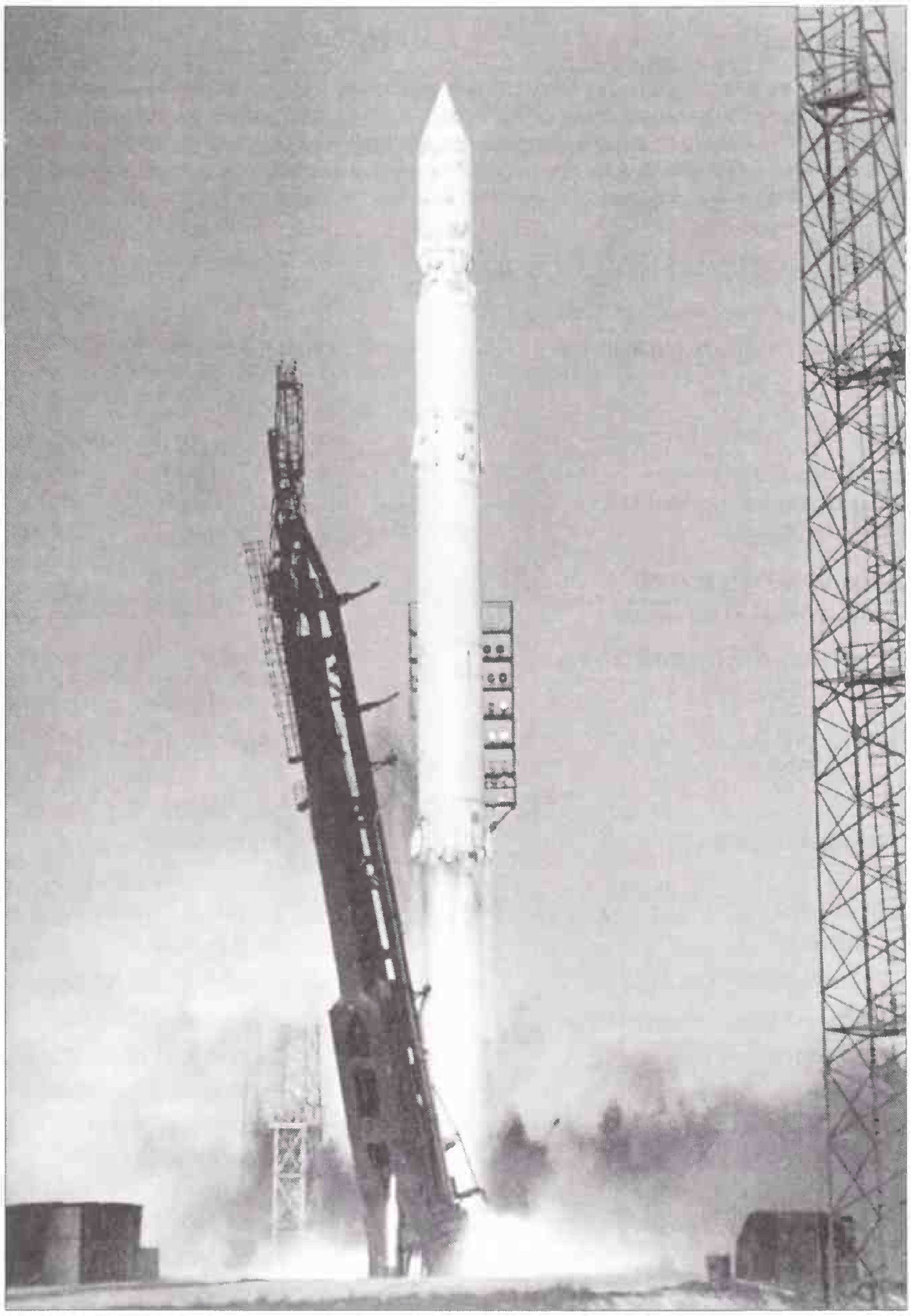
In May 1988, the U.S. Air Force chose General Dynamics to develop the Atlas II vehicle, primarily to launch Defense Satellite Communications System (DSCS) payloads. Subsequently, General Dynamics decided to scale back the Atlas I program to 11 vehicles and use the excess assets for the Atlas II series vehicles. The basic Atlas II configuration was developed in response to a DoD requirement for a medium-lift launch vehicle (MLV-2) to boost 10 DSCS satellites into orbit. The Atlas booster was stretched 2.7 m (9 ft) and employed an improved Rocketdyne engine set, the MA-5A. The booster engines had an increased thrust level of 1842 kN (414 klbf), but the sustainer engine was not modified and provided the nominal 269 kN (60,500 lbf) of thrust. The vernier engines were replaced by a hydrazine-fueled ARCM mounted on the interstage adapter. In addition, the Centaur upper stage was stretched by 0.9 m (3 ft), and its old fiberglass honeycomb insulation panels were replaced with fixed polyvinyl chloride foam panels. With the 3.3-m (11-ft) payload fairing, this configuration was capable of placing 2770 kg (6100 lbm) into GTO. All 10 Atlas II launches were successful, with the last flight occurring in 1998.

In addition to the basic Atlas II, two advanced versions were developed, primarily for the commercial market. The Atlas IIA and IIAS include an upgraded RL10 engine, providing 90 kN (20,250 lbf) of thrust, and extendible nozzles to add 6.5 s of specific impulse. In addition, the Atlas IIAS adds four Thiokol Castor IVA solid rocket motors (SRMs) to the booster stage. Each SRM provides an average thrust of approximately 434 kN (97,500 lbf). The Atlas II, IIA, and IIAS have been very successful in the commercial space launch industry, in part because of their 100% reliability demonstrated to date.

The Atlas program went through several dramatic changes during the mid 1990s. After decades at General Dynamics, the Atlas and Centaur programs were sold to Martin Marietta in 1994. Most production moved from San Diego to the Martin Marietta facilities in Denver during 1995, just as the company was merging with Lockheed to become Lockheed Martin. In late 1995, the new company announced that it would fund significant upgrades to the Atlas vehicle. The new Atlas, initially designated IIAR, would have new engines, a single-stage booster (thus eliminating the characteristic stage and a half design) and a single-engine Centaur stage. Reductions in parts count, number of engines, and staging events are expected to increase reliability and reduce cost. The new RD-180 engine, the product of a joint venture between Pratt & Whitney and the Russian engine manufacturer NPO Energomash, was selected over the Rocketdyne MA-5D and the Russian NK-33 for the first stage. The new engine offers significantly higher thrust and specific impulse than its predecessors, and also incorporates throttling capability for the first time, thus reducing launch loads on the payload. In 1998, Lockheed Martin announced that the Atlas IIAR would instead be designated the Atlas IIIA, and that it would also develop the Atlas IIIB, with a longer dual engine Centaur stage for further performance increases. The Atlas IIIA was first launched in 2000, and the IIIB in 2002. However, attention was already shifting to the next step, the Atlas V.

The Atlas V was designed as part of the USAF Evolved Expendable Launch Vehicle (EELV) program, which began in 1995. In November 1996 the U.S. Air Force awarded four $30-million phase-one study contracts to Alliant Techsystems, The Boeing Company, Lockheed Martin Corporation, and McDonnell Douglas Aerospace. During the second phase, preengineering and manufacturing development, two $60-million, 17-month contracts were awarded to Boeing and Lockheed Martin to continue refining their system concepts and complete a detailed system design. The Lockheed Martin approach is based on an evolution of their existing Atlas II/III product line, with the Boeing design being an evolution of their Delta II/III products. Originally the U.S. Air Force planned a winner-take-all competition in which one contractor would provide EELV launch services to the government. At that time the U.S. Air Force believed that the total market was too small to sustain two competing launch systems. However, in reviewing the market with the industry-led Commercial Space Transportation Advisory Committee (COMSTAC), the U.S. Air Force was persuaded that the commercial market was large enough to support competition between the EELV winners. In November 1997 the U.S. Air Force announced that it intended to introduce competition across the lifespan of the EELV program, to encourage greater contractor investment and competition in the U.S. space launch industry, and to decrease the U.S. Air Force's overall development cost. The U.S. Air Force initially awarded Atlas V nine launches, while Delta IV was awarded nineteen launches. During the subsequent few years the launch market shrank significantly, making it difficult to justify the expense of fully developing both launch vehicle families. In 2000 Lockheed Martin's contract was restructured so that the company would not have to field the heavy lift configuration, or build a pad at VAFB. As a result, two planned West-Coast launches were transferred from Atlas V to Delta. The first Atlas V was successfully launched in August 2002, a few months ahead of the first Delta IV. As a result of increased orders, Lockheed Martin has decided to proceed with construction of a pad at VAFB, which could be operational by 2005.

CYCLONE



Courtesy Yuzhnoye.

The Cyclone is a small launch vehicle built in the Ukraine. It has been used since 1967 to carry spacecraft too large for the Kosmos 3M.

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General Description

Cyclone 2 and Cyclone 3

Cyclone 2

Cyclone 3

Summary

The Cyclone 2 and 3 are former Soviet launch systems built in the Ukraine with Russian participation. They are used primarily to carry military spacecraft that are larger than the capacity of the Kosmos 3M and too small for the Soyuz/Molniya vehicles. The Cyclone 2 is a two-stage launch vehicle derived directly from the R-36 (SS-9 Scarp) ICBM. The Cyclone 3 has a restartable third stage for improved performance to circular orbits. The Cyclone 2 is launched only from Baikonur, and the Cyclone 3 is operated from Plesetsk. Both launch sites have highly automated launch equipment.

Status

Cyclone 2: Operational. First launch in 1967.

Cyclone 3: Operational. First launch in 1977.

Origin

Ukraine and Russia

Key Organizations

Marketing Organizations	KB Yuzhnoye
Launch Service Provider	KB Yuzhnoye?
Prime Contractor	KB Yuzhnoye

Primary Missions

Russian military LEO satellites

Estimated Launch Price

$20–25 million (FAA, 2002)

Spaceport

Launch Site	Cyclone 2: Baikonur LC 90 Pad 20
Location	46.1° N, 63.0° E
Available Inclinations	51, 65, and 97 deg
Launch Site	Cyclone 3: Plesetsk LC 32 (two pads)
Location	62.7° N, 40.3° E
Available Inclinations	63, 73.5, and 82.5 deg

Performance Summary

200 km (108 nmi)	Cyclone 2, 51 deg: 3350 kg (7370 lbm)
	Cyclone 2, 65 deg: 2820 kg (6220 lbm)
	Cyclone 3, 66 deg: 4100 kg (9020 lbm)
200 km (108 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	
Sun-Sychronous Orbit: 800 km (432 nmi), 98.6 deg	
GTO: 185×35,786 km (100×19,323 nmi)	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

	Cyclone 2	Cyclone 3
Total Orbital Flights	103	120
Launch Vehicle Successes	103	113
Launch Vehicle Partial Failures	0	2
Launch Vehicle Failures	0	5

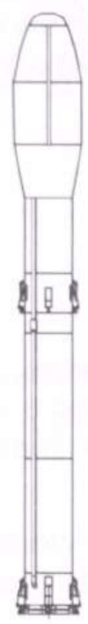
Flight Rate

Cyclone 2: 0–1 per year

Cyclone 3: 1 per year

General Description


Cyclone 2K


Cyclone 4

	Cyclone 2K	Cyclone 4

Summary

Cyclone 2K	Cyclone 4
The Cyclone 2K is a proposed upgrade that would add a new third stage to the first and second stages of Cyclone 2. It is marketed by United Start in the United States and Yuzhnoye in Ukraine.	The Cyclone 4 is a new derivative of the Cyclone family that would add a new liquid-fueled third stage and a larger diameter payload fairing. Rather than traditional launch sites at Plesetsk and Baikonur, Cyclone 4 is to be launched from the Brazilian spaceport at Alcântara. This would enable it to deliver small satellites into GTO.

Status

Cyclone 2K	Cyclone 4
In development. First launch planned 2004.	In development. First launch planned 2006.

Origin

Cyclone 2K	Cyclone 4
Russia and Ukraine	Ukraine

Key Organizations

	Cyclone 2K	Cyclone 4
Marketing organziations	United Start, Puskovie Uslugi	?
Launch service provider	Yuzhnoye SDO	Yuzhnoye SDO
Prime contractor	Yuzhnoye SDO	Yuzhnoye SDO

Primary Missions

Cyclone 2K	Cyclone 4
Small and medium LEO satellites	Small commercial GTO satellites

Estimated Launch Price

Cyclone 2K	Cyclone 4
?	?

Spaceport

	Cyclone 2K	Cyclone 4
Launch Site	Baikonur LC 90 Pad 20	Alcântara, Brazil
Location	46.1° N, 63.0° E	2.3° S, 44.4° W
Available Inclinations	65–98 deg	2.3–115 deg

Performance Summary

	Cyclone 2K	Cyclone 4
200 km (108 nmi)	65 deg: 2750 kg (6060 lbm)	2.3 deg: 5860 kg (12,920 lbm)
200 km (108 nmi), 90 deg	?	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	2650 kg (5840 lbm)	?
Sun-Sychronous Orbit: 800 km (432 nmi), 98.6 deg	1500 kg (3300 lbm)	3800 kg (8375 lbm)
GTO: 200×35,786 km (108×19,323 nmi), 2.3 deg	No capability	1560 kg (3440 lbm)
Geostationary Orbit	No capability	No capability

Flight Record (through 31 December 2003)

	Cyclone 2K	Cyclone 4
Total Flights	0	0

Flight Rate

Cyclone 2K	Cyclone 4
?	Up to 6 per year planned

Nomenclature

The name of the Cyclone launch vehicle is also commonly transliterated as Tsiklon or Tsyklon. These spellings are commonly used when referring to Russian launches of the traditional Tsiklon 2 and 3 vehicles. However, the spelling Cyclone is typically used when communicating to westerners. The preferred spelling for Cyclone 2K and Cyclone 4 is the Western spelling to distinguish them from earlier versions and because the vehicles are intended for use primarily by foreign customers

The Cyclone is a derivative of the R-36 ICBM, designated SS-9 Scarp by NATO. One version of the missile, the R-36-O, was capable of low orbital missions. Its orbital launches are therefore included in the Cyclone family launch record although it did not share the same name.

Cost

The FAA estimated in 2002 that commercial launch prices for the Cyclone 2 and 3 were in the range of $20–25 million. However these figures are uncertain because Cyclone has not yet had significant use in the commercial market.

Brazilian officials have stated that launches of Cyclone 4 may cost $30–50 million. They expect to spend about $50 million upgrading the infrastructure at the Alcântara launch site, including rail lines and the addition of a seaport, to support launches of Cyclone 4. The overall development of the Cyclone 4 program has been budgeted at $180 million, with Ukraine and Brazil each paying half. Although the project is being funded by the National Space Agency of Ukraine, private investment is also being sought to meet Ukraine's commitment.

Availability

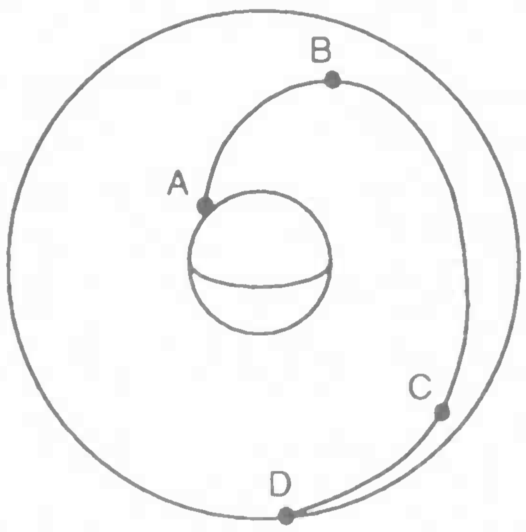
Since the breakup of the Soviet Union, the flight rate of the Cyclone has dropped from a high of 15 flights per year to 1–2 per year. In principle, Cyclone could sustain a rate of two Cyclone 2 (or 2K) launches and two Cyclone 3 launches per month. However, production of the Cyclone 2 and 3 ended in the early or mid 1990s and only a very limited number remain in storage. The Cyclone 2K is scheduled to perform its first launch in 2004. The Cyclone 4 is planned to launch in early 2006, and will be capable of six or more launches per year.

The Cyclone is unique among former Soviet launch vehicles in that it is the only one which has not been used commercially to launch Western payloads. United Start, which markets the Russian Kosmos and Start launch vehicles, announced in 2002 that it would begin marketing the Cyclone 2K vehicle in the West.

Performance

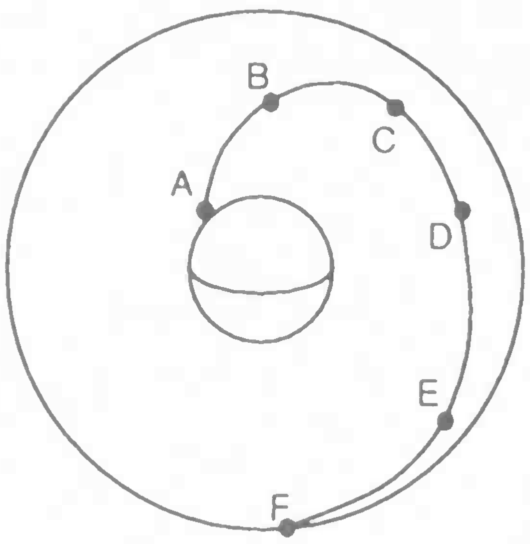
The Cyclone is launched from Baikonur (Cyclone 2) or Plesetsk (Cyclone 3). Because both sites are landlocked, Cyclone trajectories are constrained to specific launch azimuths that result in jettisoned stages landing in specified impact zones. The inclinations available from Baikonur are 51, 65, and 97 deg. The inclinations available from Plesetsk are 63, 73.5, and 82.5 deg. Cyclone does not have the capability to perform yaw-steering maneuvers to reach other inclinaions.

The basic Cyclone 2, lacking a restartable upper stage, can reach circular orbits only at very low altitudes—below approximately 400 km (215 nmi). The Cyclone 3 is used for higher orbits. It has an added C5M upper stage, which is significantly larger than the AKS. Cyclone 3 trajectories can follow one of two basic profiles. Typically, a single burn of the third-stage sustainer is used for injecting spacecraft into orbits ranging from 200–250 km (108–135 nmi) in altitude. For higher orbits Cyclone uses two burns of the third-stage sustainer engine separated by a long coast phase in an elliptical transfer orbit. The two profiles are depicted.



AB—Active segment of the flight trajectory of the first and second stages with a duration of approximately 280 s
BC—First passive segment of the third-stage flight trajectory with a duration of not less than 40 s
CD—Active segment of the third-stage flight trajectory with a duration of 5–118 s

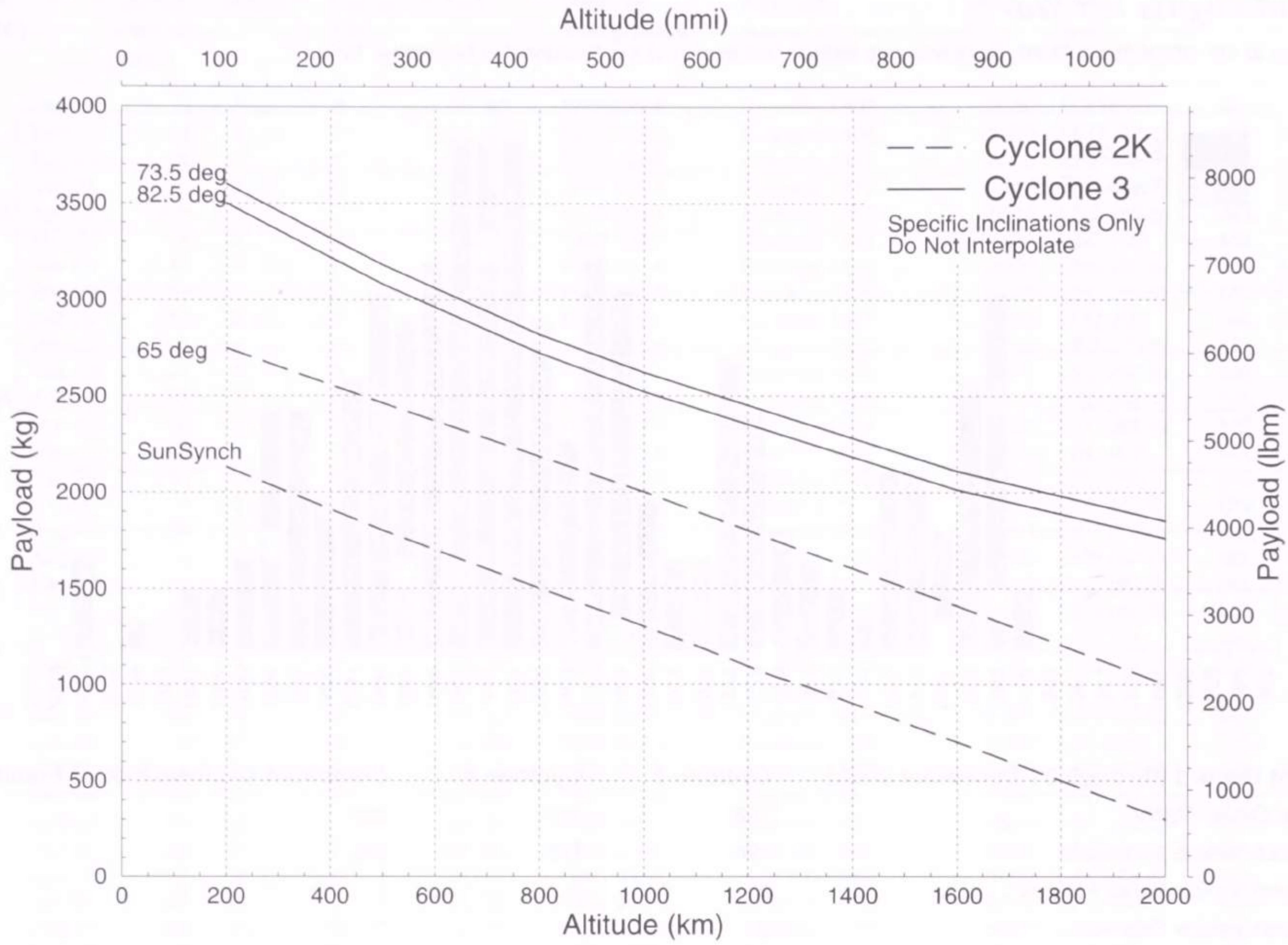
Cyclone 3 Single-Burn Third-Stage Flight Sequence



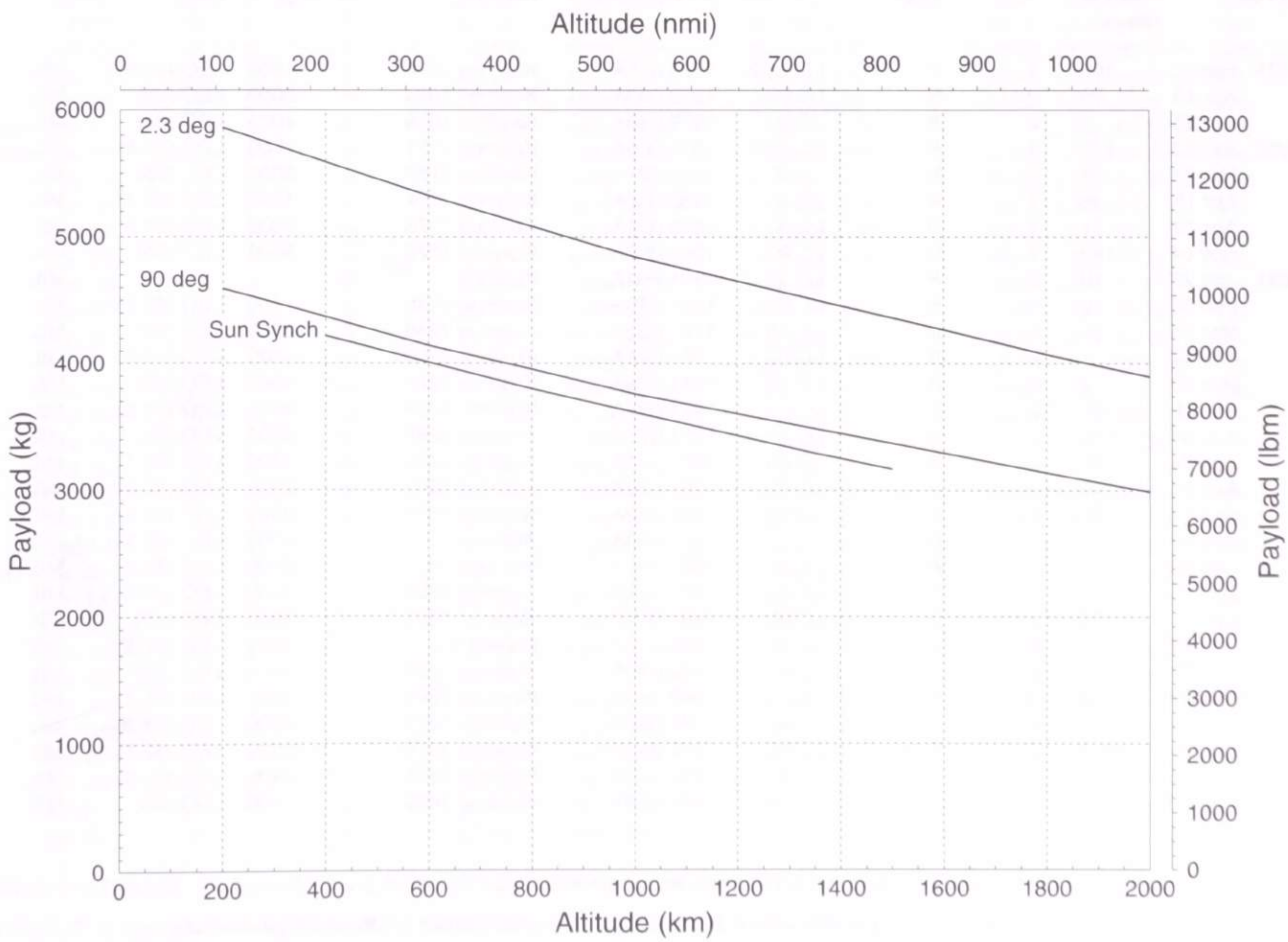
AB—Active segment of the flight trajectory of the first and second stages with a duration of approximately 280 s
BC—First passive segment of the third-stage flight trajectory with a duration of not less than 40 s
CD—First active segment of the third-stage flight trajectory with a duration of 5–113 s
DE—Second passive segment of the third-stage flight trajectory with a duration of not less than 150 s
EF—Second active segment of the third-stage flight trajectory with a duration of 5–113s

Cyclone 3 Double-Burn Third-Stage Flight Sequence

PERFORMANCE



Cyclone 2K and 3 Performance

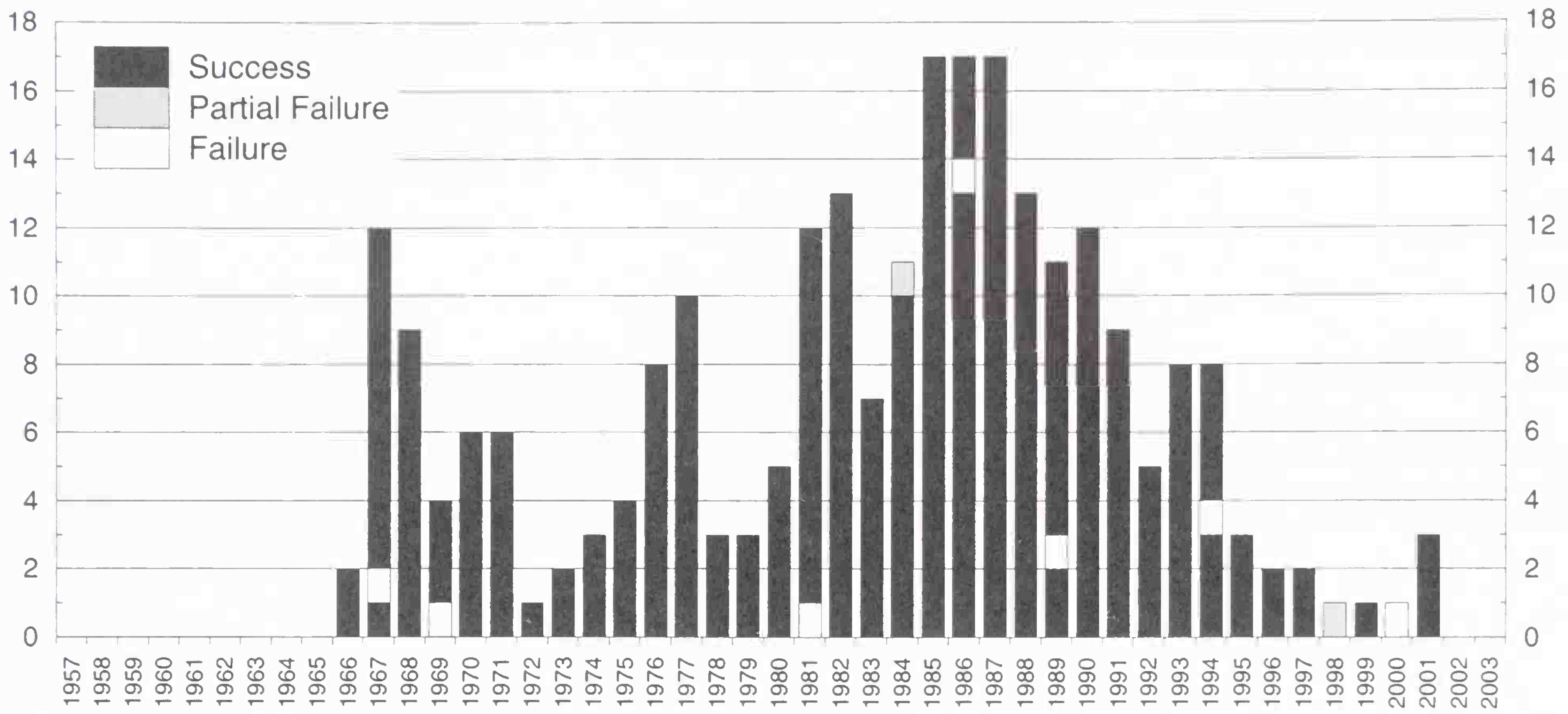


Cyclone 4 Performance

Flight History

Orbital Flights Per Year

In addition to the orbital flights listed, there was one ballistic test launch of the Cyclone 2 in November 1969.



Flight Record (through 31 December 2003)	Cyclone 2	Cyclone 3	Combined Cyclone/R-36-O Family
Total Orbital Flights	103	120	255
Launch Vehicle Successes	103	113	246
Launch Vehicle Partial Failures	0	2	2
Launch Vehicle Failures	0	5	7

		Date (UTC)		Launch Interval (days)	Model	Launch Site	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	53	1979	Feb 12	109	3	P	LC 32/2	1979 011A	Kosmos 1076	1900	LEO (82.5)	MIL	USSR
	54		Apr 18	65	2	B	LC 90	1979 033A	Kosmos 1094	3000	LEO (65)	MIL	USSR
	55		Apr 25	7	2	B	LC 90	1979 036A	Kosmos 1096	4000	LEO (65)	MIL	USSR
	56	1980	Jan 23	273	3	P	LC 32/1	1980 005A	Kosmos 1151	1900	LEO (82.5)	MIL	USSR
	57		Mar 14	51	2	B	LC 90	1980 021A	Kosmos 1167	3000	LEO (65)	MIL	USSR
	58		Apr 18	35	2	B	LC 90	1980 030A	Kosmos 1174	1400	LEO (66.1)	MIL	USSR
	59		Apr 29	11	2	B	LC 90	1980 034A	Kosmos 1176	3800	LEO (64.8)	MIL	USSR
	60		Nov 04	189	2	B	LC 90	1980 089A	Kosmos 1220	3000	LEO (65)	MIL	USSR
F	**61**	**1981**	**Jan 23**	**80**	**3**	**P**	**LC 32**	1981 F01A	Kosmos			MIL	USSR
	62		Feb 02	10	2	B	LC 90	1981 010A	Kosmos 1243	1400	LEO (65.8)	MIL	USSR
	63		Mar 05	31	2	B	LC 90	1981 021A	Kosmos 1249	3800	LEO (65)	MIL	USSR
	64		Mar 14	9	2	B	LC 90	1981 024A	Kosmos 1258	1400	LEO (65.8)	MIL	USSR
	65		Mar 20	6	2	B	LC 90	1981 028A	Kosmos 1260	3000	LEO (65)	MIL	USSR
	66		Apr 21	32	2	B	LC 90	1981 037A	Kosmos 1266	3800	LEO (64.8)	MIL	USSR
	67		Aug 04	105	2	B	LC 90	1981 072A	Kosmos 1286	3000	LEO (65)	MIL	USSR
	68		Aug 24	20	2	B	LC 90	1981 081A	Kosmos 1299	3800	LEO (65.1)	MIL	USSR
	69		Aug 24	0	3	P	LC 32	1981 082A	Kosmos 1300	2500	LEO (82.5)	MIL	USSR
	70		Sep 14	21	2	B	LC 90	1981 089A	Kosmos 1306	3000	LEO (64.9)	MIL	USSR
	71		Sep 21	7	3	P	LC 32	1981 094A	Aureole 3	1000	LEO (82.5)	CIV	USSR
	72		Sep 30	9	3	P	LC 32	1981 098A	Kosmos 1312	2200	LEO (82.6)	MIL	USSR
	73		Dec 03	64	3	P	LC 32	1982 117A	Kosmos 1328	2500	LEO (82.5)	MIL	USSR
S	74	1982	Feb 11	70	2	B	LC 90	1982 010A	Kosmos 1337	3000	LEO (65)	MIL	USSR
	75		Mar 25	42	3	P	LC 32	1982 025A	Meteor 2-8	1300	LEO (82.5)	CIV	USSR
	76		Apr 29	35	2	B	LC 90	1982 038A	Kosmos 1355	3000	LEO (65)	MIL	USSR
	77		May 14	15	2	B	LC 90	1982 043A	Kosmos 1365	3800	LEO (65.1)	MIL	USSR
	78		Jun 01	18	2	B	LC 90	1982 052A	Kosmos 1372	3800	LEO (64.9)	MIL	USSR
	79		Jun 10	9	3	P	LC 32	1982 059A	Kosmos 1378	2500	LEO (82.5)	MIL	USSR
	80		Jun 18	8	2	B	LC 90	1982 060A	Kosmos 1379	1400	LEO (65.8)	MIL	USSR
	81		Aug 30	73	2	B	LC 90	1982 084A	Kosmos 1402	3800	LEO (65)	MIL	USSR

B = Baikonur; P = Plesetsk; LC 90 Pad 20 is at Baikonur, LC 31 (with two pads) is at Plesetsk

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)		Launch Interval (days)	Model	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	82		Sep 04	5	2	B	LC 90	1982 088A		Kosmos 1405	3000	LEO (65)	MIL	USSR
	83		Sep 16	12	3	P	LC 32	1982 092A		Kosmos 1408	2500	LEO (82.6)	MIL	USSR
	84		Sep 24	8	3	P	LC 32	1982 096A		Kosmos 1410	2200	LEO (82.6)	MIL	USSR
	85		Oct 02	8	2	B	LC 90	1982 099A		Kosmos 1412	3800	LEO (64.8)	MIL	USSR
	86	1983	Apr 23	203	3	P	LC 32	1983 037A		Kosmos 1455	2500	LEO (82.5)	MIL	USSR
	87		May 07	14	2	B	LC 90	1983 044A		Kosmos 1461	3800	LEO (65)	MIL	USSR
	88		Jun 23	47	3	P	LC 32	1983 061A		Kosmos 1470	2500	LEO (82.5)	MIL	USSR
	89		Sep 28	97	3	P	LC 32/1	1983 099A		Kosmos 1500	1900	LEO (82.5)	MIL	USSR
	90		Oct 29	31	2	B	LC 90	1983 110A		Kosmos 1507	3000	LEO (65)	MIL	USSR
	91		Nov 24	26	3	P	LC 32	1983 115A		Kosmos 1510	2200	LEO (73.6)	MIL	USSR
	92		Dec 15	21	3	P	LC 32	1983 122A		Kosmos 1515	2500	LEO (82.5)	MIL	USSR
	93	1984	Feb 08	55	3	P	LC 32	1984 013A		Kosmos 1536	2500	LEO (82.4)	MIL	USSR
	94		Mar 15	36	3	P	LC 32	1984 027A		Kosmos 1544	2500	LEO (82.5)	MIL	USSR
	95		May 30	76	2	B	LC 90	1984 053A		Kosmos 1567	3000	LEO (65)	MIL	USSR
	96		Jun 29	30	2	B	LC 90	1984 069A		Kosmos 1579	3800	LEO (65)	MIL	USSR
	97		Jul 05	6	3	P	LC 32	1984 072A		Meteor 2-11	1300	LEO (82.5)	CIV	USSR
	98		Aug 07	33	2	B	LC 90	1984 083A		Kosmos 1588	3000	LEO (65)	MIL	USSR
	99		Aug 08	1	3	P	LC 32	1984 084A		Kosmos 1589	2200	LEO (82.6)	MIL	USSR
	100		Sep 28	51	3	P	LC 32/2	1984 105A		Kosmos 1602	1900	LEO (82.5)	MIL	USSR
	101		Oct 18	20	3	P	LC 32	1984 111A		Kosmos 1606	2500	LEO (82.5)	MIL	USSR
	102		Oct 31	13	2	B	LC 90	1984 112A		Kosmos 1607	3800	LEO (65)	MIL	USSR
P	**103**		**Nov 27**	**27**	**3**	**P**	**LC 32**	1984 120A		Kosmos 1612 (Meteor 3)	2215	LEO (82.6)	CIV	USSR
	104	1985	Jan 15	49	3	P	LC 32	1985 003A	M	Kosmos 1617–1622	6@225	LEO (82.6)	MIL	USSR
S	105		Jan 23	8	2	B	LC 90	1985 008A		Kosmos 1625	3000	LEO (65)	MIL	USSR
	106		Jan 24	1	3	P	LC 32	1985 009A		Kosmos 1626	2500	LEO (82.5)	MIL	USSR
	107		Feb 06	13	3	P	LC 32	1985 013A		Meteor 2-12	1300	LEO (82.5)	CIV	USSR
	108		Mar 05	27	3	P	LC 32	1985 020A		Kosmos 1633	2500	LEO (82.5)	MIL	USSR
	109		Apr 18	44	2	B	LC 90	1985 030A		Kosmos 1646	3000	LEO (65)	MIL	USSR
	110		Jun 14	57	3	P	LC 32	1985 047A		Kosmos 1660	3000	LEO (73.6)	MIL	USSR
	111		Jul 08	24	3	P	LC 32	1985 058A		Kosmos 1666	3000	LEO (82.5)	MIL	USSR
	112		Aug 01	24	2	B	LC 90	1985 064A		Kosmos 1670	3000	LEO (65)	MIL	USSR
	113		Aug 08	7	3	P	LC 32	1985 069A		Kosmos 1674	3000	LEO (82.5)	MIL	USSR
	114		Aug 23	15	2	B	LC 90	1985 075A		Kosmos 1677	3000	LEO (65)	MIL	USSR
	115		Sep 19	27	2	B	LC 90	1985 082A		Kosmos 1682	3000	LEO (65)	MIL	USSR
	116		Oct 09	20	3	P	LC 32	1985 094	M	Kosmos 1690–1695	3000	LEO (82.6)	MIL	USSR
	117		Oct 24	15	3	P	LC 32	1985 100A		Meteor 3-1	2215	LEO (82.5)	CIV	USSR
	118		Nov 22	29	3	P	LC 32	1985 108A		Kosmos 1703	2500	LEO (82.5)	MIL	USSR
	119		Dec 12	20	3	P	LC 32	1985 113A		Kosmos 1707	2500	LEO (82.5)	MIL	USSR
	120		Dec 26	14	3	P	LC 32	1985 119A		Meteor 2-13	1300	LEO (82.5)	CIV	USSR
	121	1986	Jan 17	22	3	P	LC32/2	1986 006A		Kosmos 1726		LEO (82.5)	MIL	USSR
	122	1986	Feb 11	47	3	P	LC 32	1986 015A		Kosmos 1732	2200	LEO (73.6)	MIL	USSR
	123		Feb 19	8	3	P	LC 32	1986 018A		Kosmos 1733	2500	LEO (82.5)	MIL	USSR
	124		Feb 27	8	2	B	LC 90	1986 021A		Kosmos 1735	3800	LEO (65)	MIL	USSR
	125		Mar 21	22	2	B	LC 90	1986 024A		Kosmos 1736	3800	LEO (65)	MIL	USSR
	126		Mar 25	4	2	B	LC 90	1986 025A		Kosmos 1737	3000	LEO (73.3)	MIL	USSR
	127		May 15	51	3	P	LC 32	1986 034A		Kosmos 1743	2500	LEO (82.6)	MIL	USSR
	128		May 27	12	3	P	LC 32	1986 039A		Meteor 2-14	1300	LEO (82.5)	CIV	USSR
	129		Jun 12	16	3	P	LC 32	1986 046A		Kosmos 1758	2500	LEO (82.5)	MIL	USSR
	130		Jul 28	46	3	P	LC 32/2	1986 055A		Kosmos 1766	1900	LEO (82.5)	MIL	USSR
	131		Aug 04	7	2	B	LC 90	1986 059A		Kosmos 1769	3000	LEO (65)	MIL	USSR
	132		Aug 20	16	2	B	LC 90	1986 062A		Kosmos 1771	3800	LEO (65)	MIL	USSR
	133		Sep 30	41	3	P	LC 32	1986 074A		Kosmos 1782	2500	LEO (82.5)	MIL	USSR
F	**134**		**Oct 15**	**15**	**3**	**P**	**LC 32**	1986 F06A		Kosmos			MIL	USSR
	135		Dec 02	48	3	P	LC 32	1986 094A		Kosmos 1803	2200	LEO (82.6)	MIL	USSR
	136		Dec 10	8	3	P	LC 32	1986 097A		Kosmos 1805	2500	LEO (82.5)	MIL	USSR
	137		Dec 18	8	3	P	LC 32	1986 101A		Kosmos 1809	700	LEO (82.5)	MIL	USSR
	138	1987	Jan 05	18	3	P	LC 32	1987 001A		Meteor 2-15	1300	LEO (82.5)	CIV	USSR
	139		Jan 14	9	3	P	LC 32	1987 003A		Kosmos 1812	2500	LEO (82.5)	MIL	USSR
	140		Feb 01	18	2	B	LC 90	1987 011A		Kosmos 1818	3800	LEO (65)	MIL	USSR
	141		Feb 20	19	3	P	LC 32	1987 020A		Kosmos 1823	835	LEO (73.6)	MIL	USSR
	142		Mar 03	11	3	P	LC 32	1987 024A		Kosmos 1825	2500	LEO (82.5)	MIL	USSR
	143		Mar 13	10	3	P	LC 32	1987 026	M	Kosmos 1827–1832	6@225	LEO (82.6)	MIL	USSR
	144		Apr 08	26	2	B	LC 90	1987 031A		Kosmos 1834	3000	LEO (65)	MIL	USSR
	145		Apr 27	19	3	P	LC 32	1987 038A		Kosmos 1842	2500	LEO (82.5)	MIL	USSR
	146		Jun 18	52	2	B	LC 90	1987 052A		Kosmos 1860	3800	LEO (65)	MIL	USSR
	147		Jul 01	13	3	P	LC 32	1987 055A		Kosmos 1862	1900	LEO (82.5)	MIL	USSR
	148		Jul 10	9	2	B	LC 90	1987 060A		Kosmos 1867	3800	LEO (65)	MIL	USSR

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

FLIGHT HISTORY

		Date (UTC)		Launch Interval (days)	Model	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	149		Jul 16	6	3	P	LC 32/2	1987 062A		Kosmos 1869	1600	LEO (82.5)	MIL	USSR
	150		Aug 18	33	3	P	LC 32	1987 068A		Meteor 2-16	1300	LEO (82.6)	CIV	USSR
	151		Sep 07	20	3	P	LC 32	1987 074	M	Kosmos 1875–1880	6@225	LEO (82.6)	MIL	USSR
	152		Oct 10	33	2	B	LC 90	1987 086A		Kosmos 1890	3000	LEO (65)	MIL	USSR
	153		Oct 20	10	3	P	LC 32	1987 088A		Kosmos 1892	2500	LEO (82.5)	MIL	USSR
	154		Dec 12	53	2	B	LC 90	1987 101A		Kosmos 1900	3800	LEO (66.1)	MIL	USSR
	155	1988	Jan 06	25	3	P	LC 32	1988 001A		Kosmos 1908	2500	LEO (82.5)	MIL	USSR
	156		Jan 15	9	3	P	LC 32	1988 002	M	Kosmos 1909–1914	6@225	LEO (82.6)	MIL	USSR
	157		Jan 30	15	3	P	LC 32	1998 005A		Meteor 2-17	1300	LEO (82.6)	CIV	USSR
	158		Mar 14	44	2	B	LC 90	1998 019A		Kosmos 1932	3800	LEO (65)	MIL	USSR
	159		Mar 15	1	3	P	LC 32	1998 020A		Kosmos 1933	2500	LEO (82.5)	MIL	USSR
	160		May 28	74	2	B	LC 90	1998 045A		Kosmos 1949	3000	LEO (65)	MIL	USSR
	161		May 30	2	3	P	LC 32	1998 046A		Kosmos 1950	700	LEO (73.6)	MIL	USSR
	162		Jun 14	15	3	P	LC 32	1998 050A		Kosmos 1953	2500	LEO (82.5)	MIL	USSR
	163		Jul 05	21	3	P	LC 32/1	1998 056A		Okean 1	1900	LEO (82.5)	CIV	USSR
	164		Jul 26	21	3	P	LC 32	1998 064A		Meteor 3-2	2215	LEO (82.5)	CIV	USSR
	165		Oct 11	77	3	P	LC 32	1988 093A		Kosmos 1975	2500	LEO (82.5)	MIL	USSR
	166		Nov 18	38	2	B	LC 90	1988 101A		Kosmos 1979	3000	LEO (65)	MIL	USSR
	167		Dec 23	35	3	P	LC 32	1988 113A		Kosmos 1985	3000	LEO (73.5)	MIL	USSR
	168	1989	Feb 10	49	3	P	LC 32	1989 009	M	Kosmos 1994–1999	6@225	LEO (82.6)	MIL	USSR
	169		Feb 28	18	3	P	LC 32	1989 018A		Meteor 2-18	1300	LEO (82.5)	CIV	USSR
F	**170**		**Jun 09**	**101**	**3**	**P**	**LC 32/2**	1989 F01A		Okean			CIV	USSR
	171		Jul 24	45	2	B	LC 90	1989 058A		Kosmos 2033	3000	LEO (65)	MIL	USSR
	172		Aug 28	35	3	P	LC 32	1989 068A		Kosmos 2037 (GEO 1K)	3000	LEO (73.5)	MIL	USSR
	173		Sep 14	17	3	P	LC 32	1989 074	M	Kosmos 2038–2043	3000	LEO (82.5)	MIL	USSR
	174		Sep 27	13	2	B	LC 90	1989 079A		Kosmos 2046	3000	LEO (65)	MIL	USSR
	175		Sep 28	1	3	P	LC 32/2	1989 080A		Interkosmos 24	1000	EEO (82.6)	CIV	USSR
S								1989 080B	A	Magion 2	50	EEO (82.5)	CIV	Czech.
	176		Oct 24	26	3	P	LC 32	1989 086A		Meteor 3-3	2215	LEO (82.5)	CIV	USSR
	177		Nov 24	31	2	B	LC 90	1989 092A		Kosmos 2051	3000	LEO (64.9)	MIL	USSR
	178		Dec 27	33	3	P	LC 32/2	1989 100A		Kosmos 2053	3000	LEO (73.5)	MIL	USSR
	179	1990	Jan 30	34	3	P	LC 32	1990 010A		Kosmos 2058	2500	LEO (82.5)	MIL	USSR
	180		Feb 28	29	3	P	LC 32/2	1990 018A		Okean 2	1900	LEO (82.5)	CIV	USSR
	181		Mar 14	14	2	B	LC 90	1990 022A		Kosmos 2060	3000	LEO (65)	MIL	USSR
	182		Jun 27	105	3	P	LC 32	1990 057A		Meteor 2-19	1300	LEO (82.5)	CIV	USSR
	183		Jul 30	33	3	P	LC 32	1990 066A		Kosmos 2088 (GEO 1K)	900	LEO (73.6)	MIL	USSR
	184		Aug 08	9	3	P	LC 32	1990 070	M	Kosmos 2090–2095	6@225	LEO (82.6)	MIL	USSR
	185		Aug 23	15	2	B	LC 90	1990 075A		Kosmos 2096	3000	LEO (65)	MIL	USSR
	186		Sep 28	36	3	P	LC 32	1990 086A		Meteor 2-20	1300	LEO (82.5)	CIV	USSR
	187		Nov 14	47	2	B	LC 90	1990 096A		Kosmos 2103	3000	LEO (65)	MIL	USSR
	188		Nov 28	14	3	P	LC 32	1990 104A		Kosmos 2106	3000	LEO (82.5)	MIL	USSR
	189		Dec 04	6	2	B	LC 90	1990 108A		Kosmos 2107	3000	LEO (65)	MIL	USSR
	190		Dec 22	18	3	P	LC 32	1990 114	M	Kosmos 2114–2119	6@225	LEO (82.6)	MIL	USSR
	191	1991	Jan 18	27	2	B	LC 90	1991 005A		Kosmos 2122	3000	LEO (65)	MIL	USSR
	192		Apr 24	96	3	P	LC 32	1991 030A		Meteor 3-4	2215	LEO (82.5)	CIV	USSR
	193		May 16	22	3	P	LC 32	1991 033A	M	Kosmos 2143–2148	6@225	LEO (82.6)	MIL	USSR
	194		Jun 04	19	3	P	LC 32/2	1991 039A		Okean 3	1900	LEO (82.5)	CIV	USSR
	195		Jun 14	10	3	P	LC 32	1991 042A		Kosmos 2151	2500	LEO (82.5)	MIL	USSR
	196		Aug 15	62	3	P	LC 32	1991 056A		Meteor 3-5/TOMS	2215	LEO (82.6)	CIV	USSR
	197		Sep 28	44	3	P	LC 32	1991 068	M	Kosmos 2157–2162	6@225	LEO (82.6)	MIL	USSR
	198		Nov 12	45	3	P	LC 32	1991 077	M	Kosmos 2165–2170	6@225	LEO (82.6)	MIL	USSR
	199		Dec 18	36	3	P	LC 32/2	1991 086A		Intercosmos 25	1000	EEO (82.6)	CIV	USSR
								1991 086E	A	Magion 3	52	EEO (82.6)	CIV	Czech.
	200	1992	Jul 13	208	3	P	LC 32	1992 042	M	Kosmos 2197–2202	6@225	LEO (82.6)	MIL	Russia
	201		Oct 20	99	3	P	LC 32	1992 068	M	Kosmos 2211–2216	6@225	LEO (82.6)	MIL	Russia
	202		Nov 24	35	3	P	LC 32	1992 080A		Kosmos 2221	2500	LEO (82.6)	MIL	Russia
	203		Dec 22	28	3	P	LC 32	1992 092A		Kosmos 2226	1000	LEO (82.6)	MIL	Russia
	204		Dec 25	3	3	P	LC 32	1992 094A		Kosmos 2228	2500	LEO (82.6)	MIL	Russia
	205	1993	Mar 30	95	2	B	LC 90	1993 018A		Kosmos 2238	3000	LEO (65)	MIL	Russia
	206		Apr 16	17	3	P	LC 32	1993 024A		Kosmos 2242	2500	LEO (82.6)	MIL	Russia
	207		Apr 28	12	2	B	LC 90	1993 029A		Kosmos 2244	3000	LEO (65)	MIL	Russia
	208		May 11	13	3	P	LC 32	1993 030	M	Kosmos 2245–2250	6@225	LEO (82.6)	MIL	Russia
	209		Jun 24	44	3	P	LC 32	1993 038	M	Kosmos 2252–2257	6@225	LEO (82.6)	MIL	Russia
	210		Jul 07	13	2	B	LC 90	1993 044A		Kosmos 2258	3000	LEO (65)	MIL	Russia
	211		Aug 31	55	3	P	LC 32	1993 055A		Meteor 2-21	1300	LEO (82.5)	CIV	Russia
								1993 055B	A	Temisat	42	LEO (82.5)	CML	Italy
	212		Sep 17	17	2	B	LC 90	1993 060A		Kosmos 2264	3000	LEO (65)	MIL	Russia
	213	1994	Jan 25	130	3	P	LC 32	1994 003A		Meteor 3-6	2215	LEO (82.5)	CIV	Russia
								1994 003B	A	TUBSat B	40	LEO (82.5)	NGO	Germany

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)		Launch Interval (days)	Model	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	214		Feb 12	18	3	P	LC 32	1994 011	M	Kosmos 2268–2273	6@225	LEO (82.5)	MIL	Russia
	215		Mar 02	18	3	P	LC 32/1	1994 014A		Koronas 1	2160	LEO (82.5)	CIV	Russia
F	**216**		**May 25**	**84**	**3**	**P**	**LC 32**	1994 F02A		Kosmos			MIL	Russia
	217		Oct 11	139	3	P	LC 32/2	1994 066A		Okean 4	2000	LEO (82.5)	CIV	Russia
	218		Nov 02	22	2	B	LC 90	1994 072A		Kosmos 2293	3000	LEO (65)	MIL	Russia
	219		Nov 29	27	3	P	LC 32	1994 078A		GEO 1K	900	LEO (73.6)	CIV	Russia
	220		Dec 26	27	3	P	LC 32	1994 086	M	Kosmos 2299–2304	6@225	LEO (82.6)	MIL	Russia
	221	1995	Jun 08	164	2	B	LC 90	1995 028A		Kosmos 2313	3000	LEO (65)	MIL	Russia
	222		Aug 31	84	3	B	LC32/2	1995 046A		Sich 1	1500	LEO (82.5)	CIV	Russia
S								1995 046B	A	FASat-Alfa	57		MIL	Chile
	223		Dec 20	111	2	B	LC 90	1995 071A		Kosmos 2326	3000	LEO (65)	MIL	Russia
	224	1996	Feb 19	61	3	P	LC 32/1	1996 009A	M	Gonets D1	225	LEO (82.6)	CML	Russia
								1996 009B	M	Gonets D2	225	LEO (82.6)	CML	Russia
								1996 009C	M	Gonets D3	225	LEO (82.6)	CML	Russia
								1996 009D	M	Kosmos 2328	225	LEO (82.6)	MIL	Russia
								1996 009E	M	Kosmos 2329	225	LEO (82.6)	MIL	Russia
								1996 009F	M	Kosmos 2330	225	LEO (82.6)	MIL	Russia
	225		Dec 11	296	2	B	LC 90L	1996 069A		Kosmos 2335	3000	LEO (65)	MIL	Russia
	226	1997	Feb 14	65	3	P	LC 32/1	1997 006A	M	Gonets D4	225	LEO (82.6)	CML	Russia
								1997 006B	M	Gonets D5	225	LEO (82.6)	CML	Russia
								1997 006C	M	Gonets D6	225	LEO (82.6)	CML	Russia
								1997 006D	M	Kosmos 2337	225	LEO (82.6)	MIL	Russia
								1997 006E	M	Kosmos 2338	225	LEO (82.6)	MIL	Russia
								1997 006F	M	Kosmos 2339	225	LEO (82.6)	MIL	Russia
	227		Dec 09	298	2	B	LC 90	1997 079A		Kosmos 2347	3000	LEO (65)	MIL	Russia
P	**228**	**1998**	**Jun 15**	**188**	**3**	**P**	**LC 32/1**	1998 036	M	Kosmos 2352–2357	6@225	LEO (82.6)	MIL	Russia
	229	1999	Dec 26		2	B	LC 90	1999 072A		Kosmos 2367		LEO (65)	MIL	Russia
F	**230**	**2000**	**Dec 28**		**3**	**P**	**LC 32**	2000 F03A	M	Gonets D-1	225	LEO (82.6)	CIV	Russia
								2000 F03B	M	Gonets D-1	225	LEO (82.6)	CIV	Russia
								2000 F03C	M	Gonets D-1	225	LEO (82.6)	CIV	Russia
								2000 F03D	M	Kosmos	225	LEO (82.6)	MIL	Russia
								2000 F03E	M	Kosmos	225	LEO (82.6)	MIL	Russia
								2000 F03F	M	Kosmos	225	LEO (82.6)	MIL	Russia
	231	2001	Jul 31		3	P	LC 32	2001 032A		Koronas F	2260	LEO (82.5)	CIV	Russia
	232		Dec 21		2	B	LC90	2001 057A		Kosmos 2383		LEO (65)	MIL	Russia
	233		Dec 28		3	P	LC 32	2001 058A	M	Gonets D1	225	LEO (82.6)	MIL	Russia
								2001 058B	M	Gonets D1	225	LEO (82.6)	MIL	Russia
								2001 058C	M	Gonets D1	225	LEO (82.6)	MIL	Russia
								2001 058D	M	Kosmos 2384	225	LEO (82.6)	MIL	Russia
								2001 058E	M	Kosmos 2385	225	LEO (82.6)	MIL	Russia
								2001 058F	M	Kosmos 2386	225	LEO (82.6)	MIL	Russia

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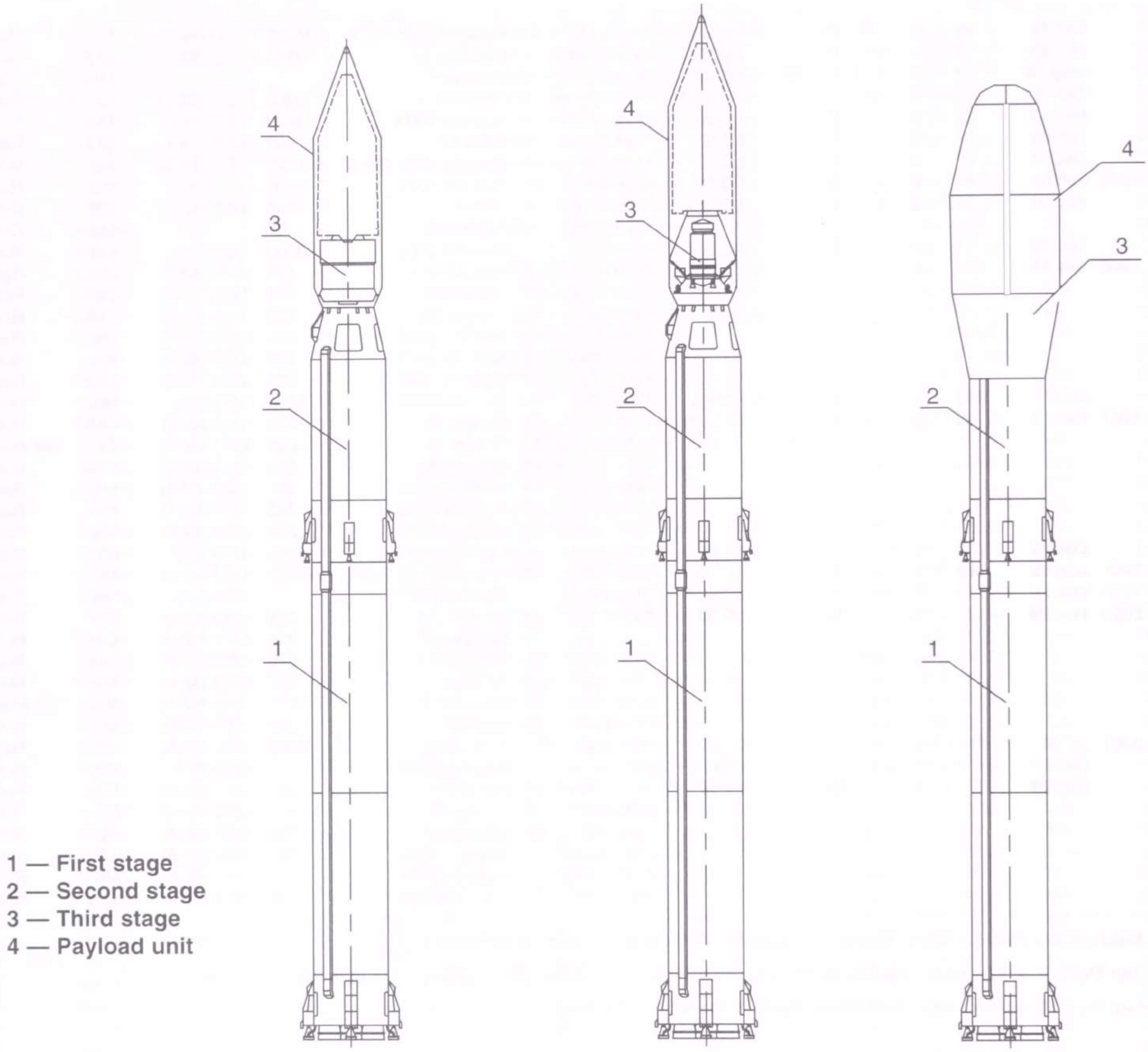
Failure Descriptions:

F	1966 Sep 17	R-36-O	1966 088	Vehicle fragmented into 53 tracked objects in orbit.
F	1967 Mar 22	R-36-O	1967 F02	Second-stage failure.
S	1969 Jan 25	Kosmos	1969 F02	Launch vehicle operated as planned, but malfunction of satellite propulsion system resulted in orbital injection failure.
S	1969 Aug 06	Kosmos 291	1969 066A	Satellite propulsion system failed.
S	1973 Apr 25	Kosmos	1973 F01	Launch vehicle operated as planned, but malfunction of satellite propulsion system resulted in orbital injection failure.
F	1981 Jan 23	Cyclone 3	1981 F01	The payload shroud failed to split and separate from the third-stage assembly.
S	1982 Feb 11	Kosmos 1337	1982 010A	Satellite propulsion or electronics system failed.
P	1984 Nov 27	Cyclone 3	1984 120	The third stage failed to restart for the circularization burn, stranding the spacecraft in an elliptical orbit with a low perigee. It was operated with limited utility, but reentered after only 2 months.
S	1985 Jan 23	Kosmos 1625	1985 008A	Satellite propulsion system failed; satellite decayed from orbit without operating.
F	1986 Oct 15	Cyclone 3	1986 F06	First-stage failure.
F	1989 Jun 09	Cyclone 3	1989 F01	Third stage failed to restart.
F	1994 May 25	Cyclone 3	1994 F02	Control system aboard the satellite failed to correctly transmit the command for the second and third stages to separate.
S	1995 Aug 31	FASat-Alfa	1995 046B	Failed to separate.
P	1998 Jun 15	Cyclone 3	1998 036	A prelaunch failure of the ground support equipment caused an anomaly during the third-stage burn that resulted in a roughly 1300×1875 km orbit, rather than the planned 1400-km near-circular orbit. The spacecraft was able to operate successfully in this orbit, and therefore the missions is considered a success by Yuzhnoye.
F	2000 Dec 28	Cyclone 3	2000 F03	Failure of a third-stage steering engine caused the control system to shut down the main engine at T+367 s. Vehicle crashed southeast of Wrangel Island. Failure was attributed to either a manufacturing defect or age—the rocket had been in storage for more than 10 years.

Vehicle Design

Overall Vehicle



	Cyclone 3	Cyclone 2K	Cyclone 4
Height	39.27 m (128.8 ft)	40.39 m (132.5 ft)	39.95 m (131.1 ft)
Gross Liftoff Mass	190 t (418 klbm) with 3.6-t (9340-lbm) spacecraft	184 t (405 klbm) with 2-t (4400-lbm) spacecraft	198 t (436 klbm) with 5-t (11-klbm) spacecraft
Thrust at Liftoff	2745 N (617 klbf)	2745 N (617 klbf)	2745 N (617 klbf)

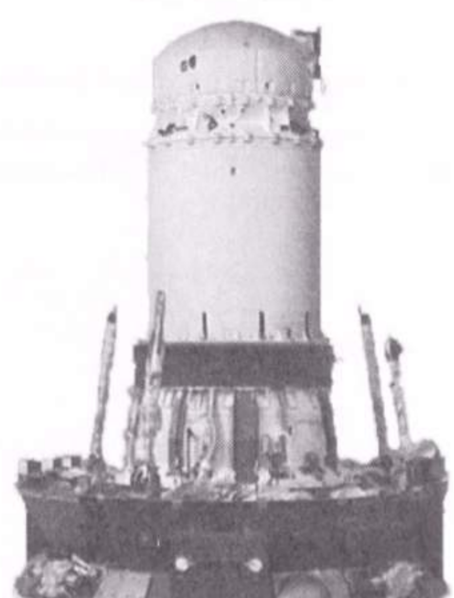
Stages

Cyclone configurations consist of two or three inline stages and a payload fairing. The Cyclone 2 is a basic two-stage configuration, while the Cyclone 3, 4, and 2K add different third stages inside the payload fairing. Propellants for all stages are storable N_2O_4 and UDMH, which are standard in ICBM-derived launch vehicles. The first stage consists of two cylindrical tanks with hemispherical bulkheads separated by an intertank structure that houses electrical systems. An aft compartment houses the propulsion system, consisting of three twin-chamber main engines and a four-chamber vernier engine mounted outside the circumference. Solid retrorockets are also attached here. The fuel tanks are arc-welded, while the intertank structure and aft compartment are riveted skin-stringer structures. The second stage is of similar construction, but the propellant tanks share a common bulkhead. The single main engine has two thrust chambers. Avionics are mounted on the forward end of the second stage.

Three different upper stages are used. The Cyclone 3 uses the C5M upper stage. Its pump-fed main engine can restart twice under weightless conditions, which permits spacecraft to be deployed in various orbits required for particular missions. The engine is enclosed by a single toroidal tank, which has an internal divider to separate the fuel and oxidizer. The stage is designed for long-term storage in a fueled state. The Cyclone 2K uses a new APM-600 (Apogee Propulsion Module) upper stage. This stage has a conical propulsion section with four engines and 16 ACS thrusters, a spherical propellant tank, and small cylindrical avionics section.

Vehicle Design

	Stage 1	Stage 2	Stage 3: Cyclone 3 (C5M)	Stage 3: Cyclone 4	Stage 3: Cyclone 2K (APM-610)
Dimensions					
Length	18.87 m (61.9 ft)	10.08 m (33.08 ft)	2.72 m (8.9 ft)	2.83 m (9.28 ft)	2.9 m (9.5 ft)
Diameter	3.0 m (9.8 ft)	3.0 m (9.8 ft)	2.4 m (7.9 ft)	3.96 m (13.0 ft)	2.7 m (8.9 ft)
Mass					
Propellant Mass	121 t (267 klbm)	49 t (108 klbm)	3.2 t (7050 lbm)	9.12 t (20.1 klbm)	600 kg (1320 lbm)
Inert Mass	6.3 t (13.9 klbm)	3.5 t (7.7 klbm) Cyclone 4: 2.6 t (5700 lbm)	1.4 t (3080 lbm)	1.34 t (2950 lbm)	1040 kg (2290 lbm)
Gross Mass	127.4 t (280.8 klbm)	52.5 t (115.7 klbm) Cyclone 4: 51.7 t (114 klbm)	4.6 t (10,140 lbm)	13.38 t (29.5 klbm)	1640 kg (3614 lbm)
Propellant Mass Fraction	0.95	0.93 Cyclone 4: 0.95	0.70	0.68	0.37
Structure					
Type	Skin-stringer	Skin-stringer	Monocoque	?	?
Material	Aluminum alloy	Aluminum alloy	Aluminum alloy	Aluminum alloy	Aluminum alloy
Propulsion					
Engine Designation	RD-251 + RD-855	RD-252 + RD-855	RD-861	RD-861K	APM-600
Number of Engines	1 RD-261 system comprising 3 RD-260 twin-chamber engines (6 chambers total), plus 1 RD-855 4-chamber steering engine	1 engine with 2 chambers and 4-chamber steering engine	1 engine	1	4
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Average Thrust	2745 kN (617 klbf) total	995 kN (223 klbf)	78.1 kN (17.6 klbf)	76.4 kN (17.2 klbf)	8826 kN (1984 klbf)
Isp	Sea level: 267.8 s	Vacuum: 315.3 s	Vacuum: 314.4 s	Vacuum: 325 s	Vacuum: 301 s
Chamber Pressure	86.1 bar (1250 psi)	92.2 bar (1337 psi)	91.7 bar (1330 klbf)	90.5 bar (1313 klbf)	?
Nozzle Expansion Ratio	14.7:1	46.1:1	112.4:1	115.8:1	?
Propellant Feed System	Turbopump	Turbopump	Turbopump	Turbopump	Gas generator
Mixture Ratio (O/F)	2.6:1	2.6:1	2.01:1	2.41:1	?
Throttling Capability	90–105%	Cyclone 2: 80–105% Cyclone 3: 90–105%	90–105%	100% only	?
Restart Capability	No	No	2 restarts	3 restarts	More than 2 restarts
Tank Pressurization	Gas generator	Gas generator	Helium	Helium	Gas generator
Attitude Control					
Pitch, Yaw, Roll	4-chamber steering engine	4-chamber steering engines	8 steering nozzles + 8 ACS thrusters	Main engine gimbal plus 10 ACS thrusters for roll control and 3-axis control during coast	16 ACS thrusters
Staging					
Nominal Burn Time	120 s	160 s	125 s	370 s	185 s
Shutdown Process	Command shutdown	Command shutdown	Command shutdown	Command shutdown	Command shutdown
Stage Separation	Retro-rockets	Retro-rockets	Spring ejection	Spring ejection	Spring ejection



Courtesy Puskovie Usulugi.
Cyclone 2K Upper Stage

Vehicle Design

Attitude Control System

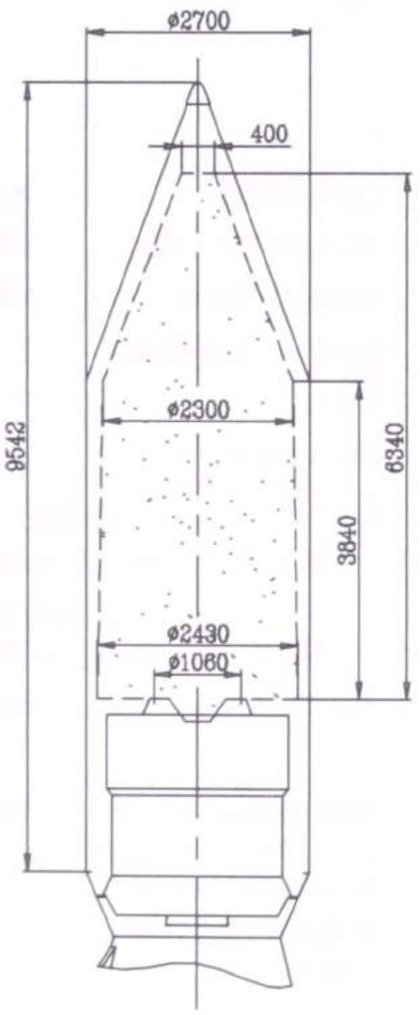
The first- and second-stage main engines are fixed. Steering is provided on each stage by a vernier engine, which has four small thrust chambers mounted 90 deg apart in shielded housings outside the diameter of the launch vehicle. The first-stage verniers can gimbal ±41 deg while the second-stage verniers gimbal ±30 deg. The Cyclone 3 and 2K upper stages control attitude using 16 thrusters. The Cyclone 4 will be able to gimbal the main-engine nozzle for thrust vector control.

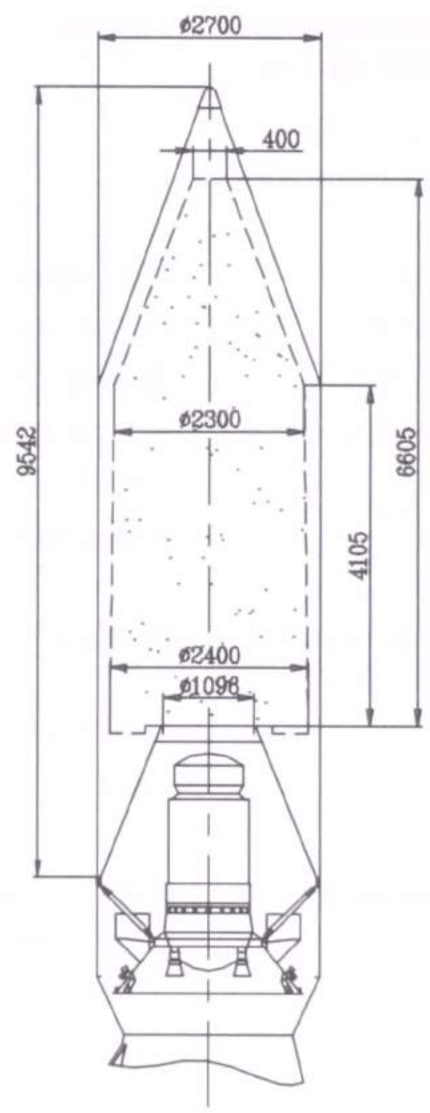
Avionics

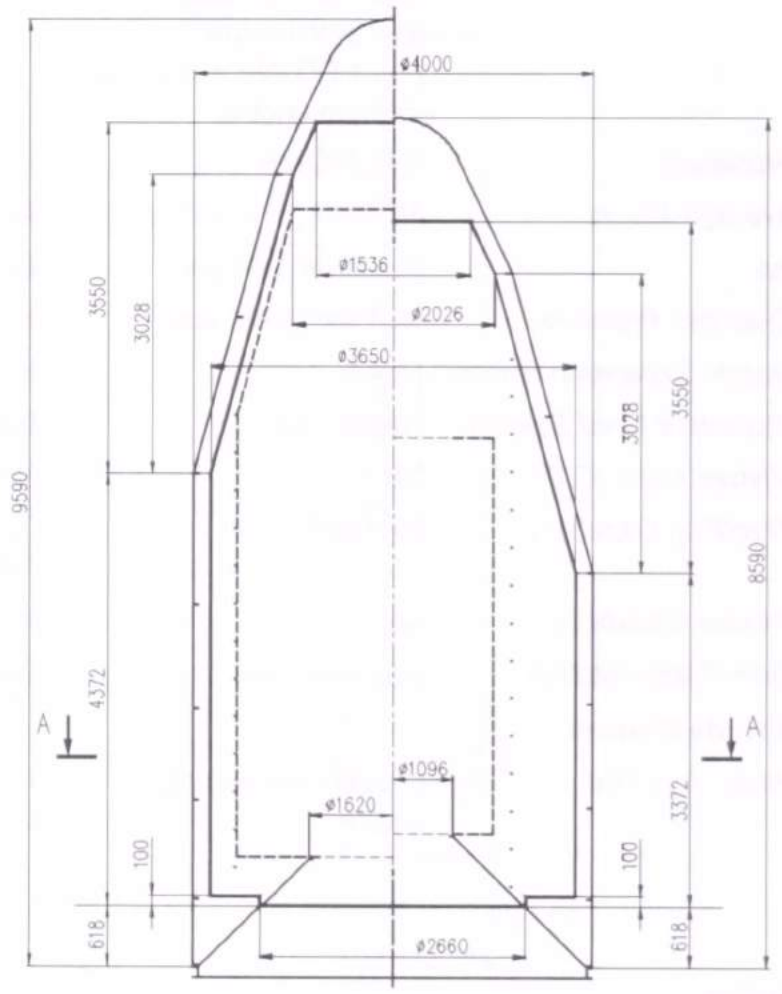
The lower two stages are controlled by an avionics system mounted on the second stage. Guidance is provided by a strapdown inertial guidance system, and the control system uses a two-out-of-three voting system for redundancy. Telemetry is provided by a multichannel radio link at 512 kbps. Each upper stage has its own avionics and control systems. Unlike other versions of the Cyclone, the Cyclone 4 will have a modernized control and flight safety system, and most of the launch vehicle avionics will be located on the third stage.

Payload Fairing

The same payload fairing is used for both Cyclone 3 and 2K. The fairing is held in place on the second stage by four pyrobolts, and the two halves are held together using mechanical latches. At separation, the segments are pushed apart by springs, and rotate on partial hinges through an angle of 65 deg before falling away from the vehicle. The payload fairing separation time can be selected based on payload sensitivity to aerodynamic heating. The Cyclone 4 will use a newly designed trisector payload fairing with a larger diameter to accommodate bigger spacecraft. It is available in two lengths—8.697 m (28.5 ft) and 9.697 m (31.8 ft).







	Cyclone 3	Cyclone 2K	Cyclone 4
Length	9.54 m (31.3 ft)	9.54 m (31.3 ft)	8.697 m (28.5 ft) or 9.697 m (31.8 ft)
Primary Diameter	2.7 m (8.9 ft)	2.7 m (8.9 ft)	4 m (13.1 ft)
Sections	2	2	3
Structure	Reinforced casing	Reinforced casing	Honeycomb
Material	Aluminum	Aluminum	Aluminum

Payload Accommodations

	Cylone 2	Cyclone 3 and 2K	Cylcone 4
Payload Compartment			
Maximum Payload Diameter	2400 mm (94.5 in)	2430 mm (95.6 in)	3650 mm (143.7 in)
Maximum Cylinder Length	3840 mm (151.2 in)	4105 mm (161.6 in)	3372 (132.8 in) or 4372 mm (172.1 in)
Maximum Cone Length	2500 mm (98.4 in)	2500 mm (98.4 in)	3550 mm (139.8 in)
Payload Adapter Interface Diameter	Bolted Interface: 1096 mm (43.1 in); Clamp Band Interface: 1194 mm (47.0 in)	1060 mm (41.7 in)	Bolted Interface: 1096 mm (43.1 in); Clamp Band Interface: 11,194 mm (47.0 in), 1566 (61.6 in), 1666 mm (65.6 in)
Payload Integration			
Nominal Mission Schedule Begins	18–24 months	18–24 months	18–24 months
Launch Window			
Last Countdown Hold Not Requiring Recycling	6 min	6 min	6 min
On-Pad Storage Capability	Fueled: 2 h Not fueled: 2 days	Fueled: 2 h Not fueled: 2 days	Fueled: 2 h Not fueled: 2 days
Last Access to Payload	T–23 min	T–30 min	T–30 min
Environment			
Maximum Axial Load	10–13 *g* depending on spacecraft mass	8–11 *g* depending on spacecraft mass	5.19–6.77 *g* depending on spacecraft mass
Maximum Lateral Load	±0.2 *g*	±0.2 *g*	±0.2 *g*
Minimum Lateral/Longitudinal Payload Frequency	10–20 Hz	10–20 Hz	15–25 / 8–10 Hz
Maximum Acoustic Level	131 dB at 125–160 Hz	131 dB at 125–160 Hz	116–139 dB at 250–8000 Hz
Maximum Flight Shock	1000 *g* from 1–2 kHz	1000 *g* from 1–2 kHz	?
Maximum Dynamic Pressure on Fairing	?	?	?
Maximum Aero-Heating Rate at Fairing Separation	< 3 kW/m2 (< 0.26 BTU/ft2/s)	< 3 kW/m2 (< 0.26 BTU/ft2/s)	< 3 kW/m2 (< 0.26 BTU/ft2/s)
Maximum Pressure Change in Fairing	2.0 kPa/s (0.3 psi/s)	2.0 kPa/s (0.3 psi/s)	2.0 kPa/s (0.3 psi/s)
Cleanliness Level in Fairing	Class 100,000	Class 100,000	Class 100,000
Payload Delivery			
Standard Orbit Injection Accuracy (3 sigma)		650 km (351 nmi) orbit: ±15 km (8.1 nmi) radius ±5 s period ±0.05 deg inclination	500 km (270 nmi) orbit: ±0.001 eccentricity ±4 s period ±0.2 deg inclination
	950 km (513 nmi) orbit: ±7–9 km (3.8–4.8 nmi) radius ±0.08 deg, 0.12 deg inclination	950 km (513 nmi) orbit: ±17 km (10.3 nmi) radius ±6 s period ±0.05 deg inclination	GTO: ±5 km (2.7 nmi) perigee altitude ±150 km (80 nmi) apogee altitude ±0.05 deg inclination
Attitude Accuracy (3 sigma)	Before separation: ±1.5 deg, ±1.0 deg/s After separation: ±3 deg, ±0.3 deg/s	Before separation: ±2 deg, ±0.3 deg/s After separation: ±3 deg, ±0.3 deg/s	Before separation: ±1.5 deg, ±0.3 deg/s After separation: ±3 deg, ±0.3 deg/s
Nominal Payload Separation Rate	1.25 ± 0.25 m/s (4.1 ft/s)	1.25 ± 0.25 m/s (4.1 ft/s)	0.5–1 m/s (1.6–3.3 ft)
Deployment Rotation Rate Available	?	< 3 deg/s	< 2 deg/s
Loiter Duration in Orbit	?	2.25 h	?
Maneuvers (Thermal/Collision Avoidance)	Yes	No	No
Multiple/Auxiliary Payloads			
Multiple or Comanifest	Cyclone 3 has routinely carried up to six Kosmos satellites per flight		
Auxiliary Payloads	Cyclone occasionally carries secondary payloads, but there is no standard interface. Mission-specific interfaces can be developed.		

Production and Launch Operations

Production

The Cyclone launch vehicles were designed by Yuzhnoye SDO of Dneipropetrovsk, Ukraine, and produced by its affiliate production organization, Yuzhmash PA. Production of Cyclone 2 and 3 ended in the mid 1990s. A number of vehicles were stockpiled for future use. If the Cyclone 4 program goes forward, the production line would be restarted and new vehicles built.

Organization	Responsibility
Yuzhnoye SDO	System design, design of stage 1 and 2 vernier engines and Cyclone 3 third stage main engines
Yuzhmash PA	Manufacturing of stages and engines
Energomash SPA	Stage 1 and 2 main engine design
Makeyev Design Bureau	Cyclone 2K stage 3

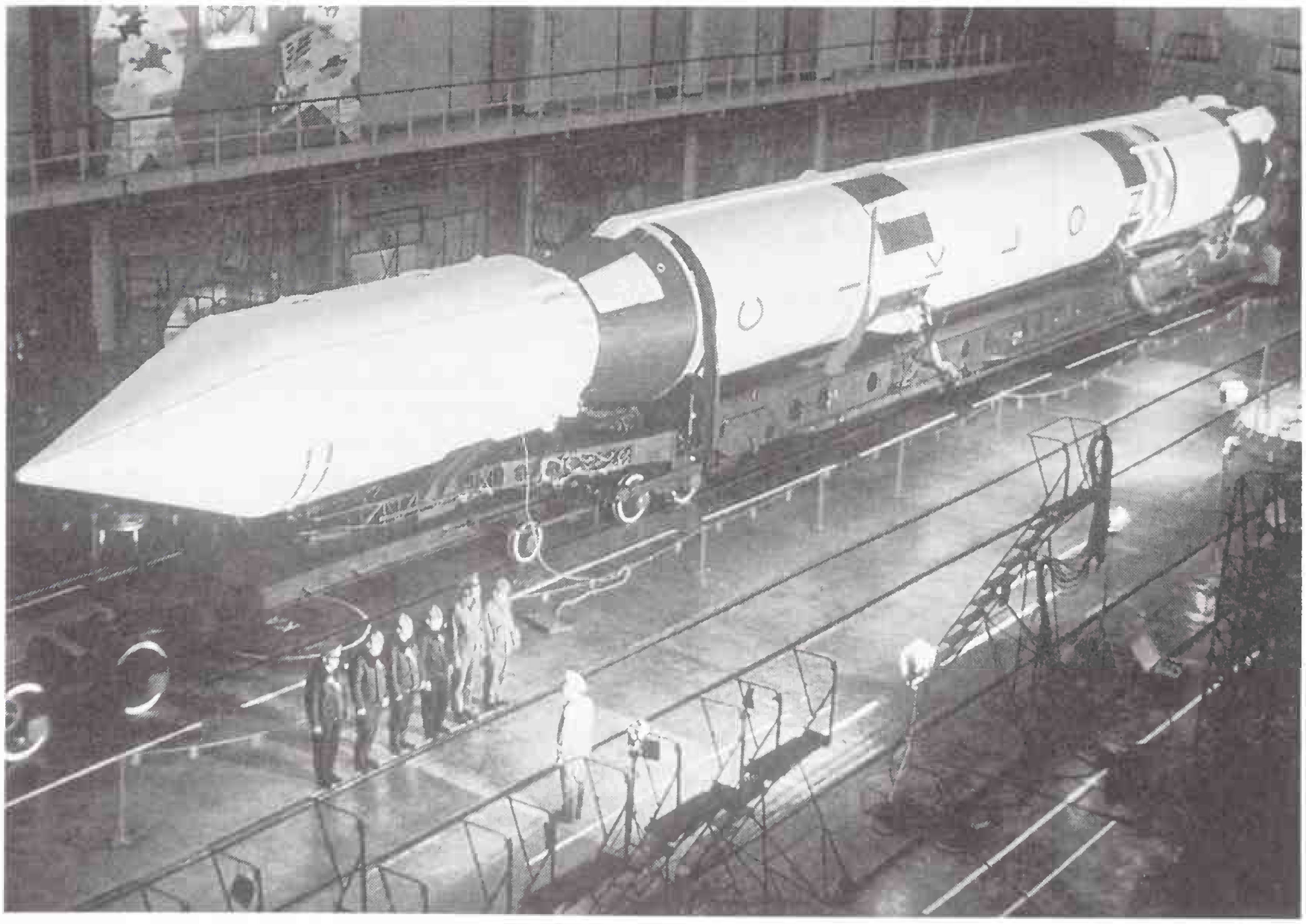
Launch Operations—Baikonur, Plesetsk

Although Cyclone vehicles are manufactured in Ukraine, they continue to be launched from the same sites in Russia and Kazakhstan as they were before the breakup of the Soviet Union, albeit at a much lower rate. An attitude of cooperation exists between the Ukrainian Space Agency, the Russian Space Agency, and the Russian Space Forces (the branch of the Russian Ministry of Defense that actually operates the launch facilities).

The Cyclone 2 is launched from the Baikonur Cosmodrome and Cyclone 3 from Plesetsk, Russia. For more information on both sites, refer to the Spaceports chapter. The Cyclone 2 launch complex, LC 90, is located in the northwest corner of Baikonur, near the Proton facilities. LC 90 has one operational launch pad, Pad 20. Although only the Cyclone 2 is currently launched from Baikonur, it has been speculated that, because of the similarities between the two vehicles, the Cyclone 3 could be launched from Baikonur with little or no facilities modifications.

The Plesetsk Cosmodrome is situated in a heavily wooded area close to the Arctic Circle near the town of Plesetsk, which is on the railway line from Moscow to Archangelsk. Historically, it has been the world's most used launch site. Both the Cyclone 2 and 3 can be accommodated at Plesetsk, although currently only the Cyclone 3 is launched from this site. The Cyclone facilities at LC 32 include two launch pads.

The Cyclone is assembled and tested in a horizontal orientation on a railcar. Beginning with launch vehicle assembly, about 12 working days are required to inspect, test, and prepare the vehicle for launch. Once integrated and assembled, a launch can occur within 24 h of the go-ahead command, in any weather conditions. The vehicle is transported by rail to the launch site and erected into the vertical position. Once vertical, there is no further access to the payload. After the vehicle has been connected to the umbilicals, all further pad operations are automated and can be controlled remotely from an underground bunker located between the two pads at the launch complex. Fueling of the lower two stages is performed automatically with no staff on the pad for safety reasons. The third stage is fueled offline before vehicle integration. Following the launch, the pad can be prepared for a second flight in as little as 10 h.



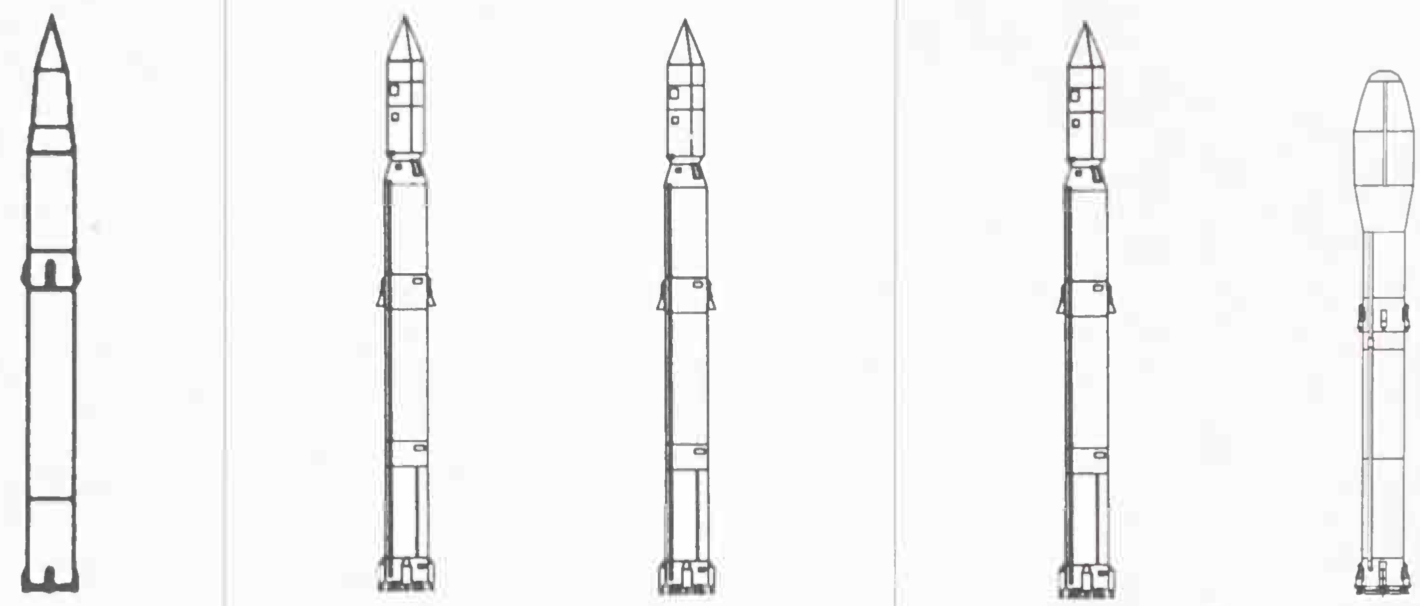
Courtesy Space Commerce Corporation

The Cyclone is a small launch vehicle built in the Ukraine. It has been used since 1967 to carry spacecraft too large for the Kosmos 3M.

Vehicle History

Vehicle Evolution

Retired | Operational | In Development



Vehicle	**R-36-O**	**Cyclone 2**	**Cyclone 3**	**Cyclone 2K**	**Cyclone 4**
Period of Service	1966–1971	1967–Present	1977–Present	Planned 2004	Planned 2006
Payload	?	51 deg: 3350 kg (7370 lbm) 65 deg: 2820 kg (6220 lbm)	66 deg: 4100 kg (9020 lbm)	2750 kg (6060 lbm)	5860 kg (12,920 lbm)

Vehicle Description

•R-36-O	F-1-r	SL-11	Orbit-capable configuration of R-36 (SS-9 Scarp) ICBM used for fractional orbit bombardment system
•Cyclone 2	F-1-m	SL-11	Two-stage derivative of R-36 for space launch
•Cyclone 3	F-2	SL-14	Similar to Cyclone 2 with added third stage
•Cyclone 2K			Similar to Cyclone 3 with new small third stage
•Cyclone 4			Derived from Cyclone 3 using new 4-m-diam third stage and payload fairing. Planned launch from Alcantara, Brazil.

Historical Summary

The Cyclone launch vehicles (also known as Tsyklon and Tsiklon) are derived from the Russian R-36 ICBM (designated SS-9 Scarp by NATO). The R-36, along with its subsequent derivative space launchers, has been produced by NPO Yuzhnoye, located in Dniepropetrovsk, Ukraine, since 1966.

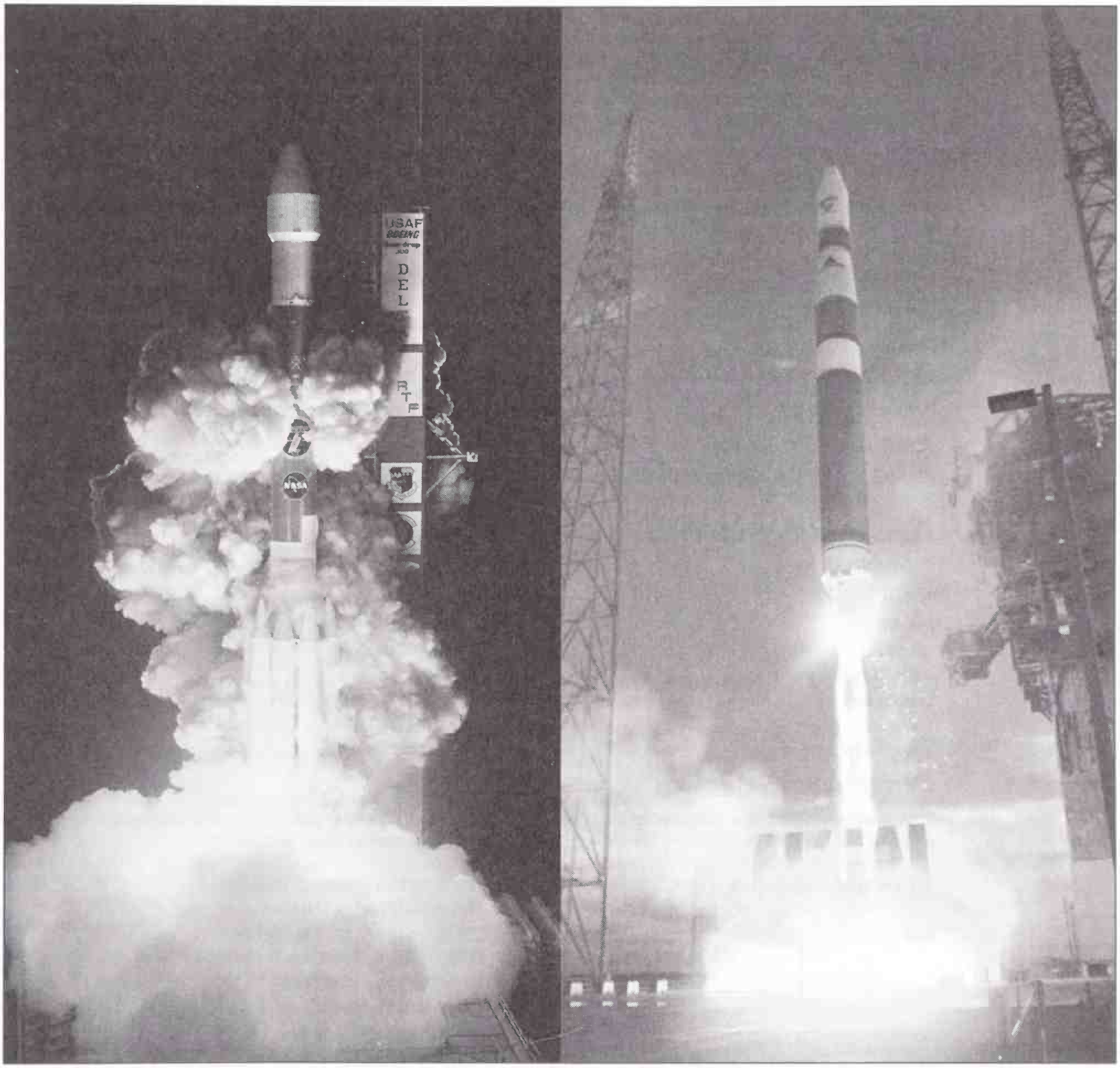
Since it entered service, the R-36 has been described as having both intercontinental and orbital capability. A variant of the R-36, the R-36-O (F-1-r), was used for tests of the Fractional Orbital Bombardment System (FOBS), so described because the payloads reached orbit, but reentered over their targets before completing one full revolution. Many of these flights were assigned COSPAR designations and provided with Kosmos numbers, even though they were in unstable orbits. The missiles were launched directly south, flying over the southern hemisphere, coming around from behind the U.S. early warning radar lines, approaching the U.S. from the Gulf of Mexico. These tests ran from 1965 through 1971. When the Strategic Arms Limitation Treaty was signed, this system was abandoned. This FOBS version of the R-36 is referred to in the Sheldon naming system as F-1-r, with the "r" standing for a retro-rocket stage, which was actually part of the payload.

The Cyclone-2 (F-1-m, SL-11) entered service 27 October 1967 with a launch from Baikonur Cosmodrome. The Cyclone-2 antisatellite interceptor and ocean reconnaissance launcher was referred to as F-1-m, the "m" standing for maneuverable stage, which is actually part of the payload. Cyclone-2 has only launched from Baikonur and has been used primarily for military purposes.

The Cyclone-3, (F-2, SL-14), which is the Cyclone-2 with a restartable third stage, was introduced 24 June 1977. A large number of missions previously flown on Kosmos and Vostok launchers have migrated to the Cyclone-3. These include communications, meteorology, remote-sensing, science, geodesy, electronic intelligence, and minor military missions. The Cyclone-3 has only been launched at Plesetsk from a highly automated launch complex into orbits at inclinations of 73.5 deg and 82.5 deg.

After the end of the Cold War, Russian space organizations entered partnerships with U.S. or European companies to market launch vehicles such as Proton, Soyuz, and Zenit. There were similar attempts to commercialize Cyclone, but none have been successful to date. Rockwell's Space Systems Division announced in 1996 that it was marketing the existing Cyclone 2 and 3 vehicles for launches from their established launch sites. Rockwell later withdrew without selling any launches. In 1997 several European companies, including DASA, Matra Marconi, and Arianespace, considered investing in an upgraded vehicle called Cyclone 3K, which would have Western avionics hardware and launch from Kourou. In 1997 Fiat Avio entered an agreement with Yuzhnoye SDO to develop and market jointly a new vehicle called Cyclone 4 with a larger third stage and payload fairing. This partnership ended around 2000, coinciding with ESA's decision to fund Fiat Avio's Vega program for a European small launch vehicle. Yuzhnoye has continued the Cyclone 4 program. After considering several low laititude launch sites, including Cape Canaveral, the Brazilian launch site at Alcantara was selected, and Brazil has agreed to cooperate in the development of launch site infrastructure for the vehicle. A new attempt at commercialization began in 2002 when United Start announced that it would market the Cyclone 2K, another derivative of the Cyclone family.

DELTA



Courtesy Boeing.

The second Delta II Heavy lifts off from CCAFS carrying the NASA Space Infrared Telescope Facility spacecraft (left). The Delta IV, (shown right) on its third mission, is on its way to a successful delivery of a USAF Defense Satellite Communications System (DSCS III B6) spacecraft.

Contact Information

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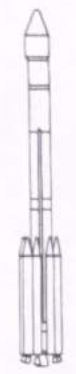
Web site: www.boeing.com/launch

General Description

Delta II



Delta II

Summary

The Delta II series was developed following the failures of the *Challenger* and several ELVs in 1986 to provide launches for USAF payloads—primarily GPS satellites. The initial Delta II 6925 configuration was phased out in 1992 in favor of the current 7925 family. In addition to USAF payloads, Delta II built a reputation as a reliable launcher for medium-sized NASA payloads and small commercial payloads headed for GTO. The newer 7420 and 7320 series can launch smaller payloads by using fewer strap-on motors. The newest configuration, the 7925H, uses larger strap-on motors for additional performance.

Status

Operational. 7920 first launch in 1990. 7420 first launch in 1998. 7326 first launch in 1998. 7925H first launch in 2003.

Origin

United States

Key Organizations

Marketing Organization	Boeing Launch Services
Launch Service Provider	Boeing Expendable Launch Systems
Prime Contractor(s)	Boeing Expendable Launch Systems

Primary Missions

Three-Stage: GPS satellites and deep-space missions
Two-stage: Remote sensing, research, and LEO communications satellites

Estimated Launch Price

Price negotiable

Spaceports

Launch Site	Cape Canaveral AFS, SLC-17A & B
Location	28.5° N, 81.0° W
Available Inclinations	28.5–60 deg
Launch Site	Vandenberg AFB, SLC-2W
Location	34.7° N, 120.6° W
Available Inclinations	63–145 deg

Performance Summary

Performance shown below accounts for the 2.9-m (9.5-ft) payload fairing and appropriate payload adapters. Only the largest and smallest Delta II configurations are shown. See Performance section for more info.

185 km (100 nmi), 28.7 deg	7320: 2776 kg (6120 lbm) 7920: 5058 kg (11,150 lbm)
185 km (100 nmi), 90 deg	7320: 2049 kg (4517 lbm) 7920: 3792 kg (8360 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	7320: 2294 kg (5058 lbm) 7920: 4104 kg (9048 lbm)
Sun-Synchronous Orbit: 833 km (450 nmi), 98.7 deg	7320: 1652 kg (3643 lbm) 7920: 3182 kg (7016 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 28.7 deg	7325: 929 kg (2048 lbm) 7925: 1832 kg (4038 lbm)
Geostationary Orbit	No capability

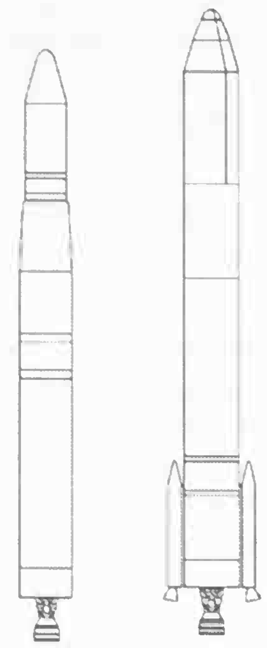
Flight Record (through 31 December 2003)

Total Orbital Flights	110
Launch Vehicle Successes	108
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	1

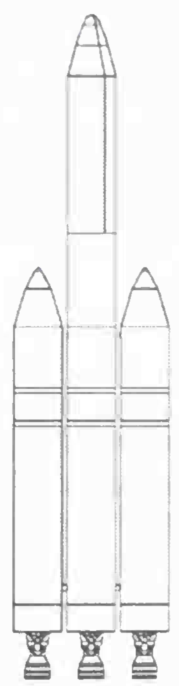
Flight Rate

3–12 per year

General Description



Delta IV Medium and Medium Plus



Delta IV Heavy

	Delta IV Medium and Medium Plus	Delta IV Heavy

Summary

Delta IV Medium and Medium Plus	Delta IV Heavy
The Delta IV Medium is a large, modular launch vehicle that can launch medium-class payloads. Both core stages are powered by hydrogen and oxygen, making the new design quite different from previous Delta configurations. The new 5-m diameter first stage is referred to as the Common Booster Core. The "Medium Plus" vehicles add two or four solid motors to increase capability to heavier launch payloads. Payload fairings are available in 4-m diameter and 5-m diameter versions.	The Delta IV Heavy is a Titan IV class heavy launch vehicle. It consists of a Delta IV core vehicle with two additional Common Booster Core stages used as liquid strap-on boosters. The Delta IV Heavy is capable of delivering spacecraft directly to geostationary orbit as well as earth escape missions.

Status

Operational. First launch in 2002.	In development. First launch in 2004.

Origin

United States	United States

Key Organizations

Marketing Organization	Boeing Launch Services	Boeing Launch Services
Launch Service Provider	Boeing Expendable Launch Systems	Boeing Expendable Launch Systems
Prime Contractor(s)	Boeing Expendable Launch Systems	Boeing Expendable Launch Systems
Primary Missions	Medium to heavy payloads to LEO and GTO	Heavy payloads to GTO, LEO, and earth escape

Estimated Launch Price

DIV M and DIVM+: price negotiable	DIV Heavy: price negotiable

Primary Missions

same as Delta II	Heavy payloads to GTO, LEO, and earth escape missions.

Spaceports

Launch Site	Cape Canaveral AFS, SLC-37	Cape Canaveral AFS, SLC-37
Location	28.5° N, 81.0° W	28.5° N, 81.0° W
Available Inclinations	28.5–51 deg	28.5–51 deg
Launch Site	Vandenberg AFB, SLC-6	Vandenberg AFB, SLC-6
Location	34.7° N, 120.6° W	34.7° N, 120.6° W
Available Inclinations	63–120 deg	63–120 deg

Performance Summary

Performance shown below is separated spacecraft mass, accounting for appropriate payload adapters. Only the smallest and largest Delta IVM configurations are shown. See the Performance section for information on other configurations.

185 km (100 nmi), 28.7 deg	DIV M: 9144 kg (20,158 lbm) DIV M+(5,4): 13,701 kg (30,205 lbm)	23,975 kg (52,855 lbm)
185 km (100 nmi), 90 deg	DIV M: 7840 kg (17,284 lbm) DIV M+(5,4): 12,021 kg (26,501 lbm)	22,184 kg (48,908 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	DIV M: 8501 kg (18,741 lbm) DIV M+(5,4): 12,894 kg (28,425 lbm)	21,892 kg (48,264 lbm)
Sun-Synchronous Orbit: 833 km (450 nmi), 98.7 deg	DIV M: 6593 kg (14,536 lbm) DIV M+(5,4): 10,463 kg (23,068 lbm)	19,265 kg (42,473 lbm)
GTO: 185×35,768 km (100×19,323 nmi), 27 deg	DIV M: 4231 kg (9327 lbm) DIV M+(5,4): 6822 kg (15,039 lbm)	12,757 kg (18,124 lbm)
Geostationary Orbit	DIV M: 1138 kg (2508 lbm) DIV M+(5,4): 2786 kg (6142 lbm)	6276 kg (13,837 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	3	0
Launch Vehicle Successes	3	0
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	0	0

Flight Rate

5–6 per year planned	2 per year planned

Nomenclature

A four digit designator has been used to distinguish between Delta II configurations since the early 1970s. However, Boeing uses a new designation system for the Delta IV vehicles. The Delta IV is available in two basic types, Medium and Heavy. These are designated DIV-M and DIV-H. The Medium can be enhanced ("Medium Plus") with a larger fairing and strap-on boosters. These configurations are designated with a digit for the fairing diameter in meters, and digit for the number of strap-on boosters. Thus, the DIV-M+ (4,2) has a 4-m fairing and two strap-on boosters, while the DIV-M+ (5,4) has a 5-m fairing and four strap-on boosters.

Delta/Delta II

Example — **Delta 7925-10**

First Digit

First-stage type, engine type, strap-on motor type

0: Long tank, MB-3 engine, Castor II

1: Extended long tank, MB-3 engine, Castor II

2: Extended long tank, RS-27 engine, Castor II

3: Extended long tank, RS-27 engine, Castor IV

4: Extended long tank, MB-3 engine, Castor IVA

5: Extended long tank, RS-27 engine, Castor IVA

6: Extra extended long tank, RS-27 engine, Castor IVA

7: Extra extended long tank, RS-27A engine, GEM-40

8: Delta III shortened first stage, RS-27A engine, GEM-46

Second Digit

Number of strap-on motors (3,4,6, or 9)

Third Digit

Second-stage engine

0: AJ-10-118 (Aerojet)

1: TR-201 (TRW)

2: AJ-10-118K (Aerojet)

3: RL10B-2 (Pratt & Whitney)

Fourth Digit

Third-stage motor

0: No third stage or unspecified

3: TE-364-3

4: TE-364-4

5: PAM-D derivative Star 48B

6: Star 37FM

Dash Number

Fairing type (optional)

-8: 8-ft fairing

-10: 10-ft composite fairing

-10L: 10-ft stretched composite fairing

Delta IV

Example — **Delta IV M+(5,4)**

M+

Denotes Delta IV Medium Plus launch vehicle

First digit

Payload fairing (PLF) diameter (meters)

Second digit

Number of graphite-epoxy motors (GEM)

Cost

The 1998 launch of Deep Space 1 cost NASA $43.5 million for Delta II 7326. The 1999 launch of Stardust on a Delta II 7426 cost $45 million. NASA's total launch services budget for the 2003 launch of Mars Exploration Rover B on a Delta II 7925H configuration was $68 million. In 1998, NASA's Discovery Program published estimated Delta II launch prices for purposes of evaluating spacecraft proposals. They listed the cost of a Delta II 7925 at $55 million. In 2002, NASA signed a contract for up to 19 Delta II launches between 2006 and 2009, worth a total of $1.2 billion. This is about $63 million per launch, though it is difficult to compare this to previous prices because of the unknown mix of vehicle configurations and the likely use of future year dollar values. Boeing executives said in the mid 1990s that the Delta II strap-on motors cost only about $1 million each. However, the cost difference between a Delta II 7490 and a Delta II 7920 is higher than the cost of the five additional strap-on motors alone would suggest, because the price is based on the value of the increased performance rather than the cost of hardware.

The Delta III is no longer being marketed, and therefore has no current market price. However, in the late 1990s the list price for a Delta III launch was $85 million. Boeing's 2000 annual report indicates that the company spent $78 million on the Delta III demonstration flight that year.

Boeing is believed to have spent about $1.5 billion of its own funds to develop Delta IV. The Air Force provided $500 million in "Other Transaction Agreements" when it selected Delta IV for the EELV program in 1998. In addition, the U.S. Air Force also provided Delta IV and each of its competitors with $30 million in 1995 and $60 million in 1996 for two phases of early study contracts. The Delta IV production plant cost around $450 million. The construction of launch facilities at CCAFS was budgeted at $250 million. Launch facilities at VAFB were expected to cost only $120 million. Development of the RS-68 engine cost about $500 million. Boeing's initial launch services contract under the EELV program is worth $1.38 billion for 19 launches. The U.S. Air Force is also paying $141 million for a demonstration launch of the first Delta IV Heavy launch vehicle. In 2000, the web newsletter "Spacelift Washington" quoted a Boeing spokesperson as saying that commercial launches of the Delta IV Heavy would cost $148–160 million.

Availability

The Delta II 7920/5 configuration has been in operation since 1990. Like all Delta configurations it is available on the international commercial launch services market. (In fact, an earlier Delta version performed the first commercially licensed space launch in the United States.) The smaller Delta II 7420/5 and 7320/5 configurations became operational in 1998. Delta II vehicles have demonstrated a peak flight rate of 12 per year. The maximum surge capability is about 15 per year. Despite the arrival of Delta IV, the Delta II will not be phased out any time soon. In late 2002, NASA awarded Boeing a contract for 12 firm Delta II launches plus options, to be exercised through 2009.

The Delta IV Medium Plus configuration first flew in 2002. The first two launches of the Medium were successfully completed in 2003. The Heavy configuration is scheduled for launch in 2004. The Delta factory is capable of producing 40 Common Booster Cores per year, which can be combined into a mix of Medium and Heavy launch vehicles. Up to 17 launches per year can be conducted from CCAFS, and as many as 9 per year from VAFB.

Performance

The Delta II series vehicles are available in two-stage and three-stage configurations. The two-stage configuration is used for LEO missions, while the three-stage configuration is used for high-energy missions, such as GPS transfer orbits, GTO, and Earth-escape trajectories. From CCAFS, launch azimuths between 65 deg and 110 deg are available, corresponding to orbit inclinations between 28.5 deg and 60 deg. The standard inclination for GTO missions is 28.7 deg rather than 28.5 deg because of range safety concerns along the coast of West Africa. From the SLC-2W pad at VAFB, Delta is limited to launch azimuths between 190 deg and 200 deg, because launch azimuths below 190 deg would overfly facilities on the southern portion of the base. However, Delta can reach inclinations as low as 63.4 deg with modest performance penalties through yaw steering.

Delta IV performance can be tailored by adding strap-on boosters to the common booster core first stage. Two or four solid boosters can be added in the Medium Plus configurations, or two additional common booster cores can be added as liquid strap-on boosters to create the Delta IV Heavy configuration. With minor modifications to the second stage, Delta IV can deliver payloads directly to GEO.

Performance shown below includes the following assumptions:

Separated spacecraft mass is shown, accounting for appropriate adapter mass for each mission.

Flight Performance Reserve provides 99.7% probability of guidance command shutdown for Delta II and 99.865% for Delta IV.

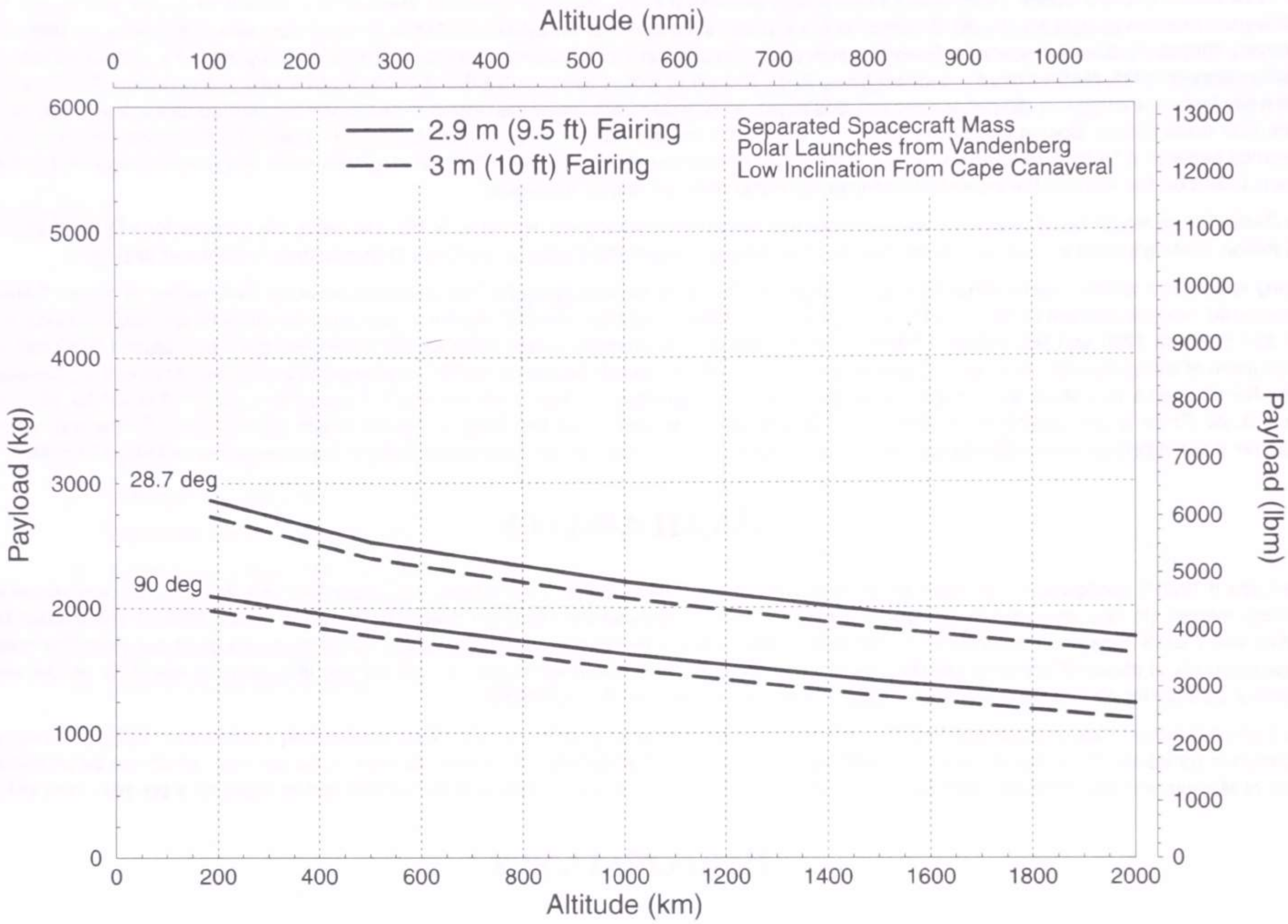
Delta II performance is for the 9.5-ft diameter payload fairing.

Vehicle	Circular: 185 km (100 nmi), 28.7 deg	Circular: 185 km (100 nmi), 90 deg	Space Station Orbit: 407 km (220 nmi), 51.6 deg	Sun-Synchronous Orbit: 833 km (450 nmi), 98.7 deg	GTO: 185×35,786 km (100×19,323 nmi)*	Geostationary Orbit
Delta II 732x	2776 kg (6120 lbm)	2049 kg (4517 lbm)	2294 kg (5058 lbm)	1652 kg (3643 lbm)	929 kg (2048 lbm)	No Capability
Delta II 742x	3185 kg (7022 lbm)	2446 kg (5392 lbm)	2648 kg (5838 lbm)	1998 kg (4405 lbm)	1060 kg (2338 lbm)	No Capability
Delta II 792x	5058 kg (11,150 lbm)	3792 kg (8360 lbm)	4104 kg (9048 lbm)	3182 kg (7016 lbm)	1832 kg (4038 lbm)	No Capability
Delta II 792xH	6107 kg (13,464 lbm)	No Capability	4343 kg (9574 lbm)	No Capability	2180 kg (4807 lbm)	No Capability
Delta IV M	9144 kg (20,158 lbm)	7840 kg (17,284 lbm)	8501 kg (18,741 lbm)	6593 kg (14,536 lbm)	4231 kg (9327 lbm)	1138 kg (2508 lbm)
Delta IV M+(4,2)	12,120 kg (26,719 lbm)	10,689 kg (23,566 lbm)	11,455 kg (25,254 lbm)	9181 kg (20,242 lbm)	5941 kg (13,098 lbm)	2036 kg (4489 lbm)
Delta IV M+(5,2)	10,667 kg (23,516 lbm)	9120 kg (20,106 lbm)	9782 kg (21,565 lbm)	7707 kg (16,992 lbm)	4869 kg (10,733 lbm)	1686 kg (3717 lbm)
Delta IV M+(5,4)	13,701 kg (30,205 lbm)	12,021 kg (26,501 lbm)	12,894 kg (28,425 lbm)	10,463 kg (23,068 lbm)	6822 kg (15,039 lbm)	2786 kg (6142 lbm)
Delta IV Heavy	23,975 kg (52,855 lbm)	22,184 kg (48,908 lbm)	21,892 kg (48,264 lbm)	19,265 kg (42,473 lbm)	12,757 kg (28,124 lbm)	6276 kg (13,837 lbm)

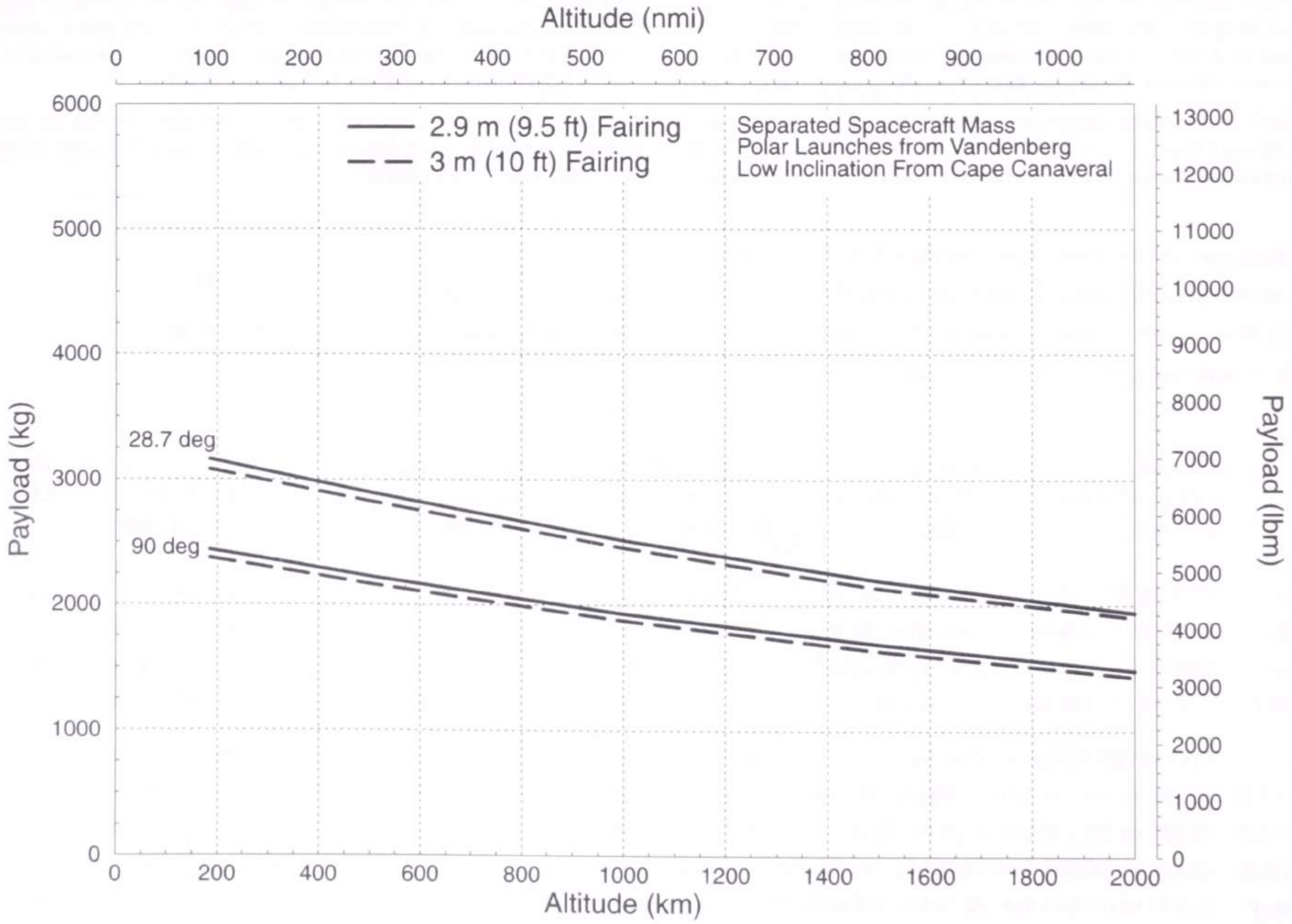
Note: Delta II GTO and geostationary performance are for three-stage configuration. All others are two-stage configurations.

**GTO inclination is 28.7 deg for Delta II and 27.0 deg for Delta IV.*

Performance

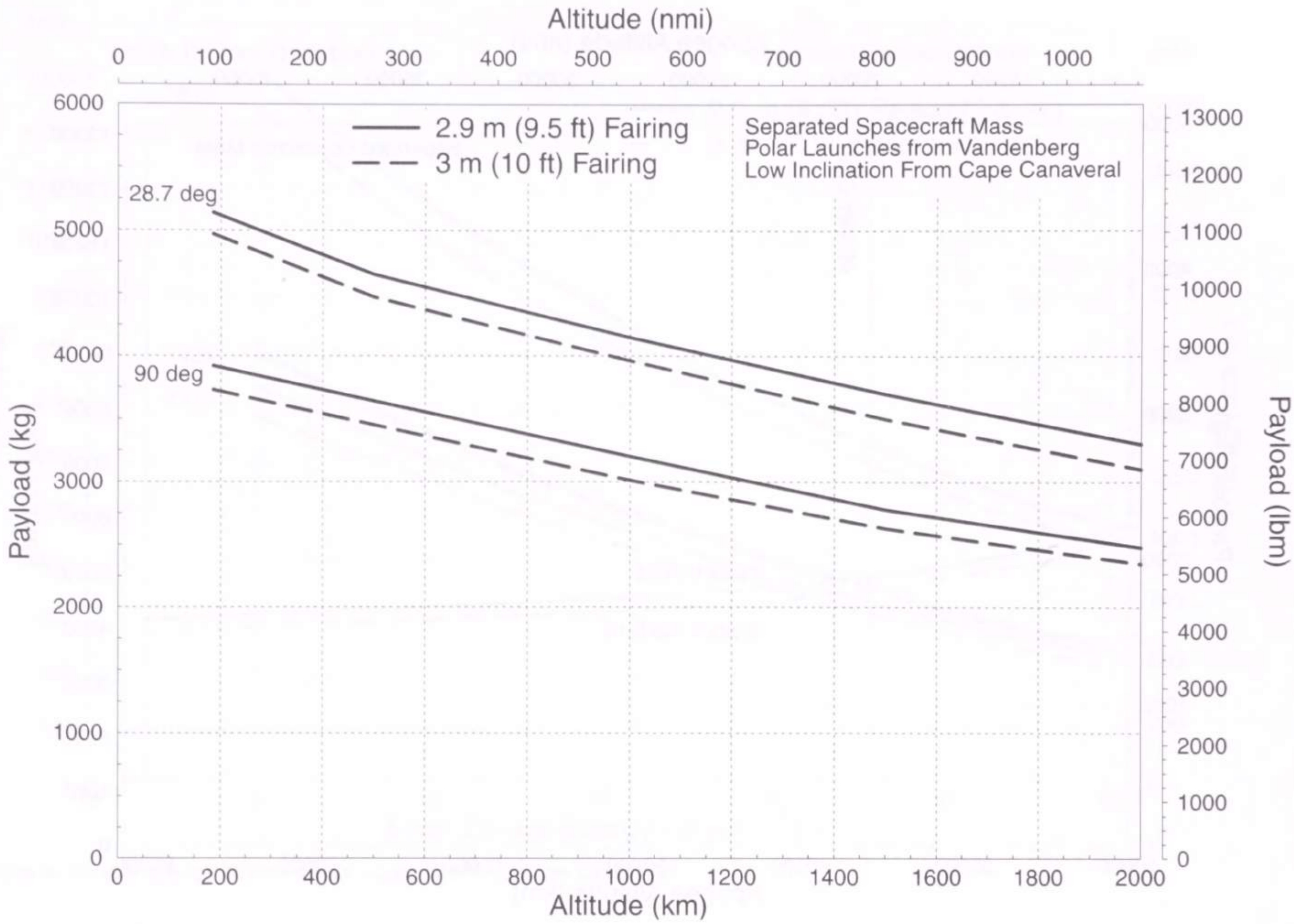


Delta II 7320: LEO Performance

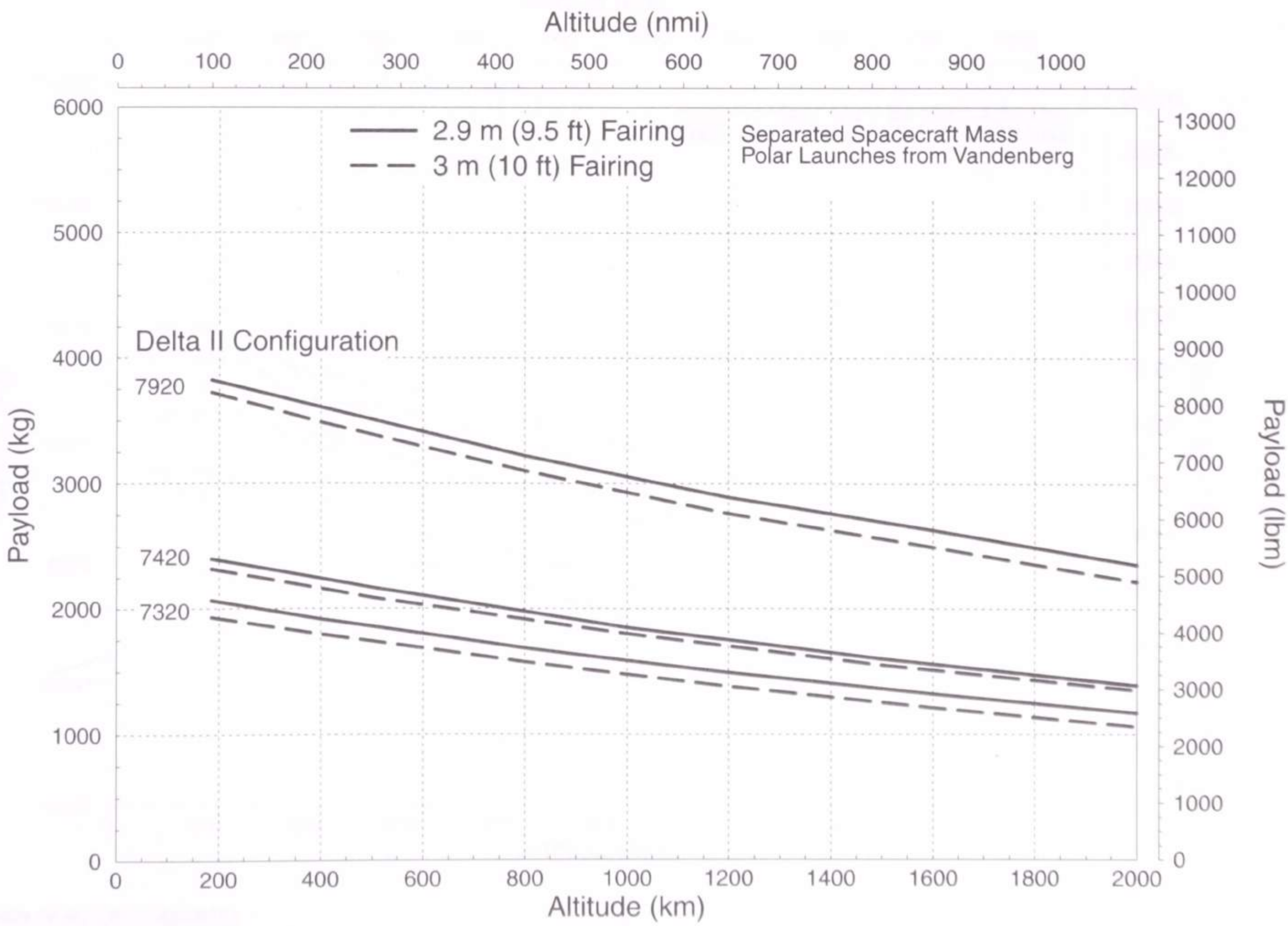


Delta II 7420: LEO Performance

PERFORMANCE

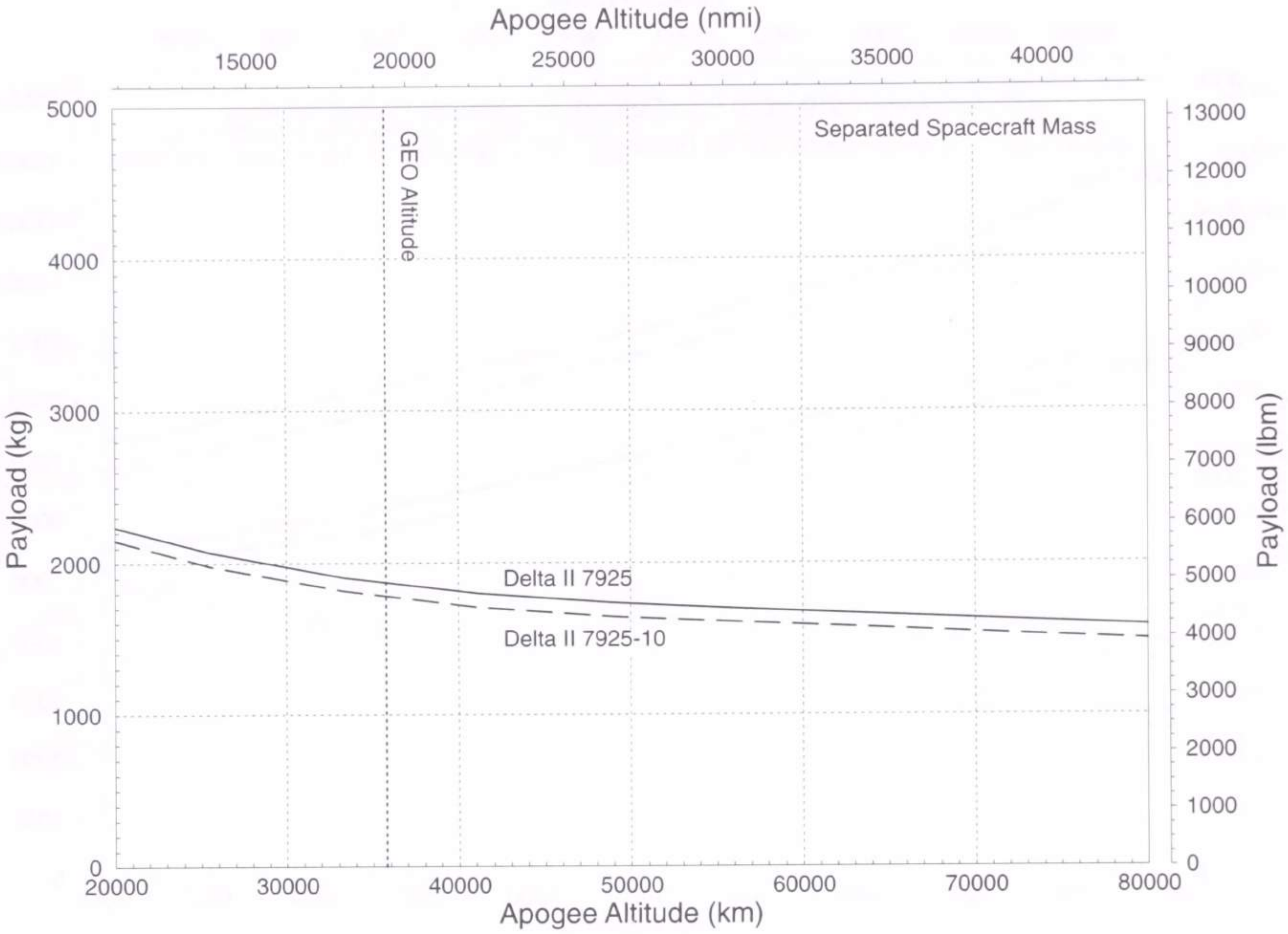


Delta II 7920: LEO Performance



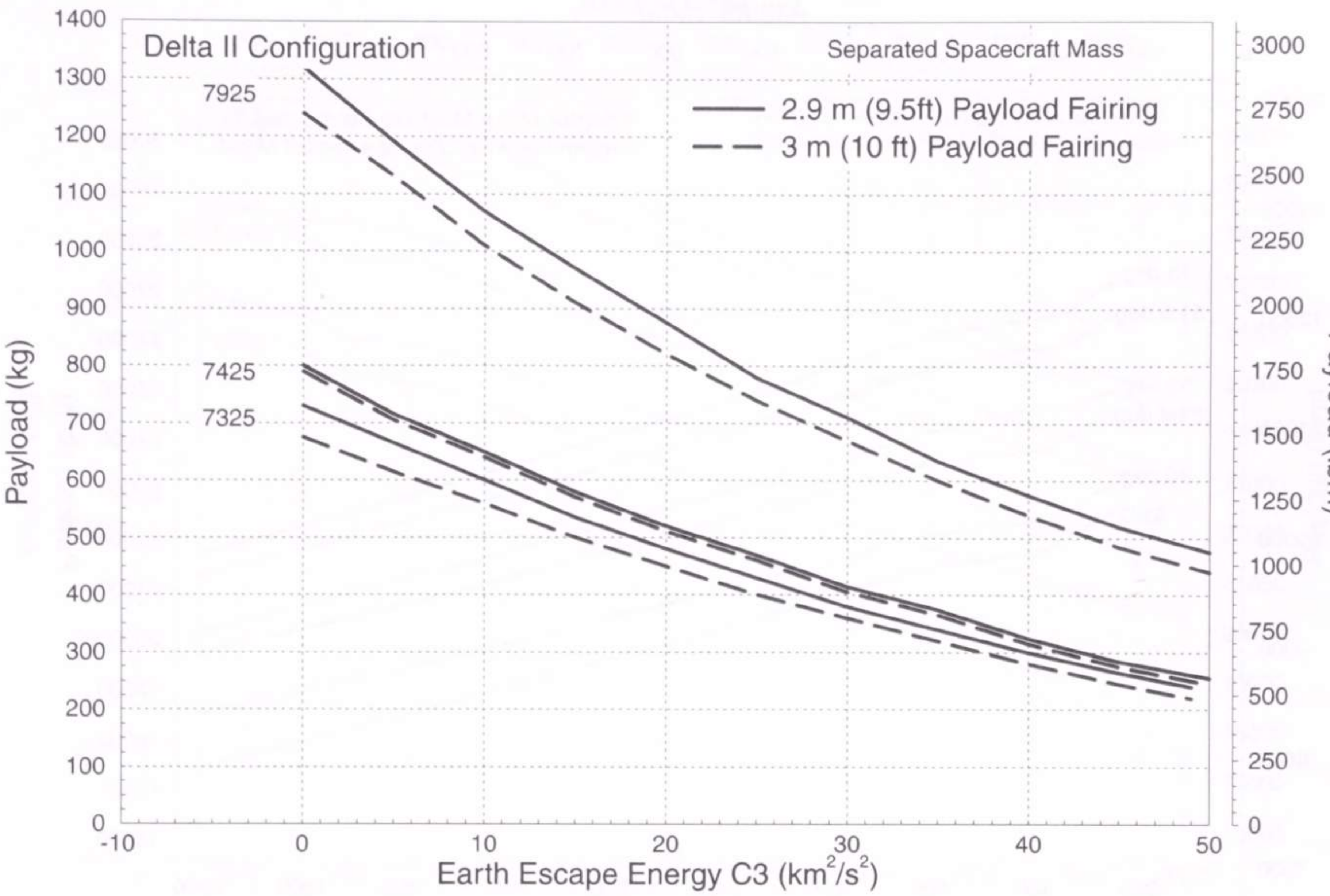
Delta II: Sun-Synchronous Orbit Capability

Performance

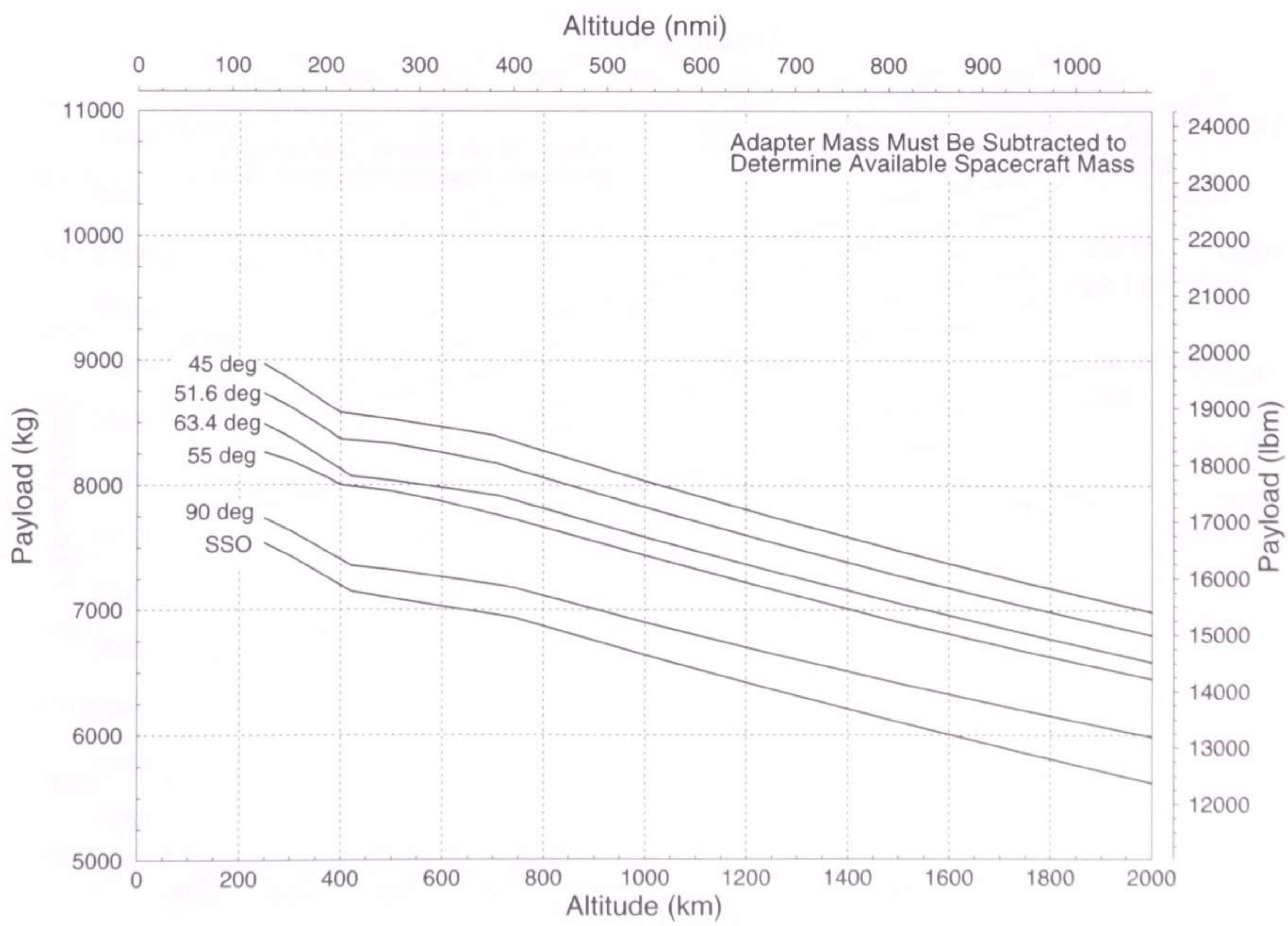


Delta II: GTO Capability, Launched from Cape Canaveral

Performance

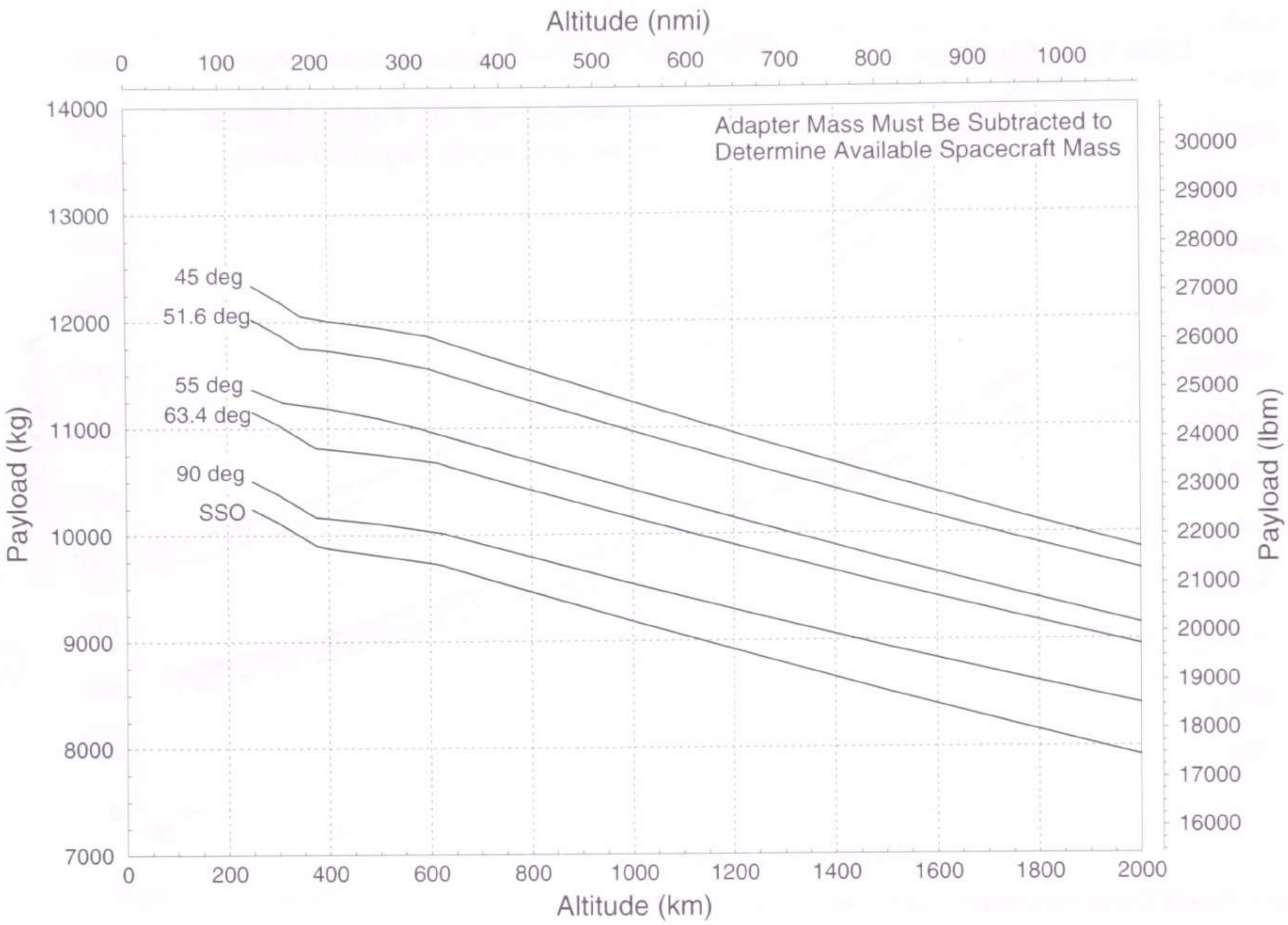


Delta II: Escape Energy Performance, Launched from Cape Canaveral

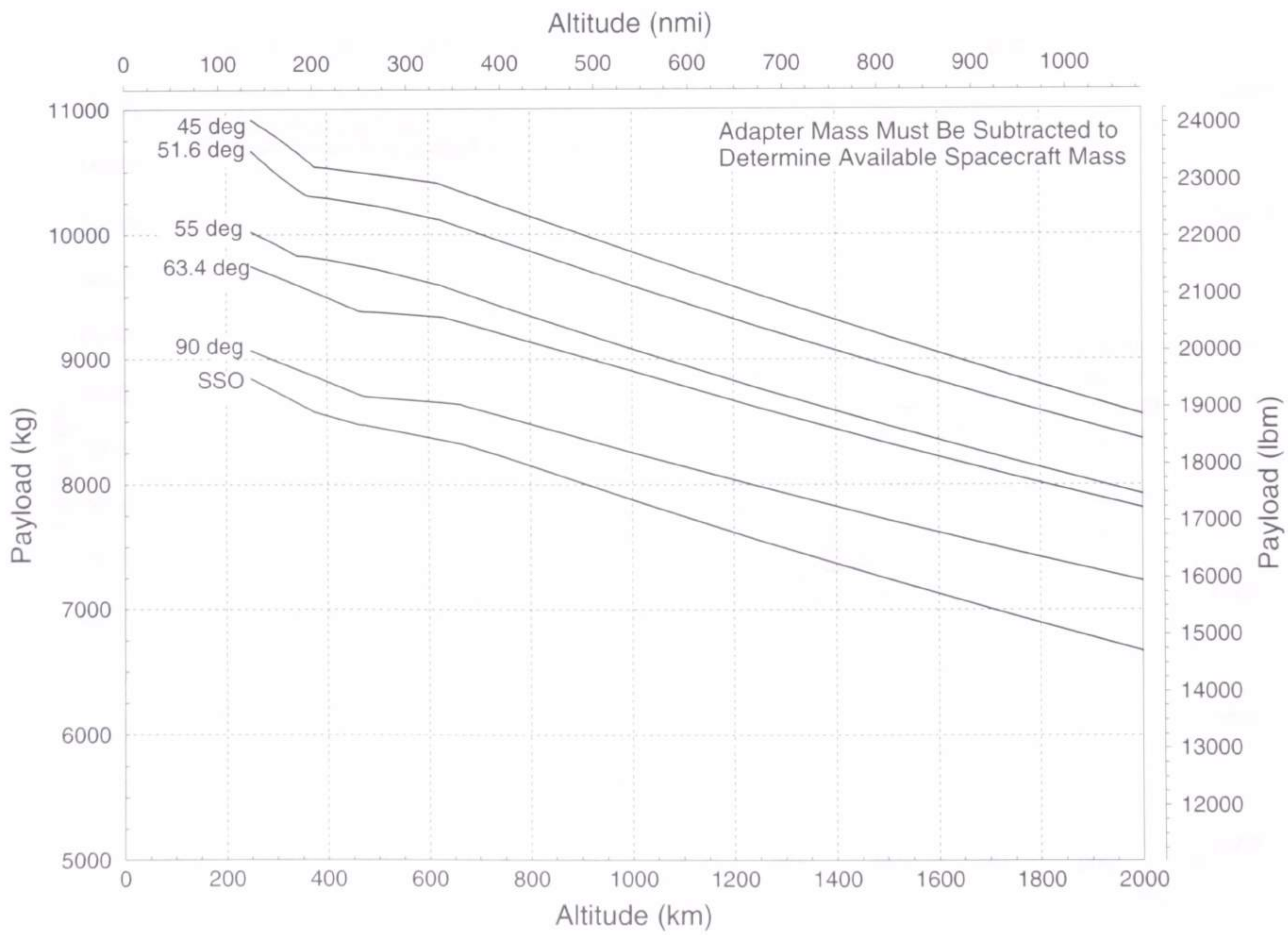


Delta IV M: LEO Capability

PERFORMANCE

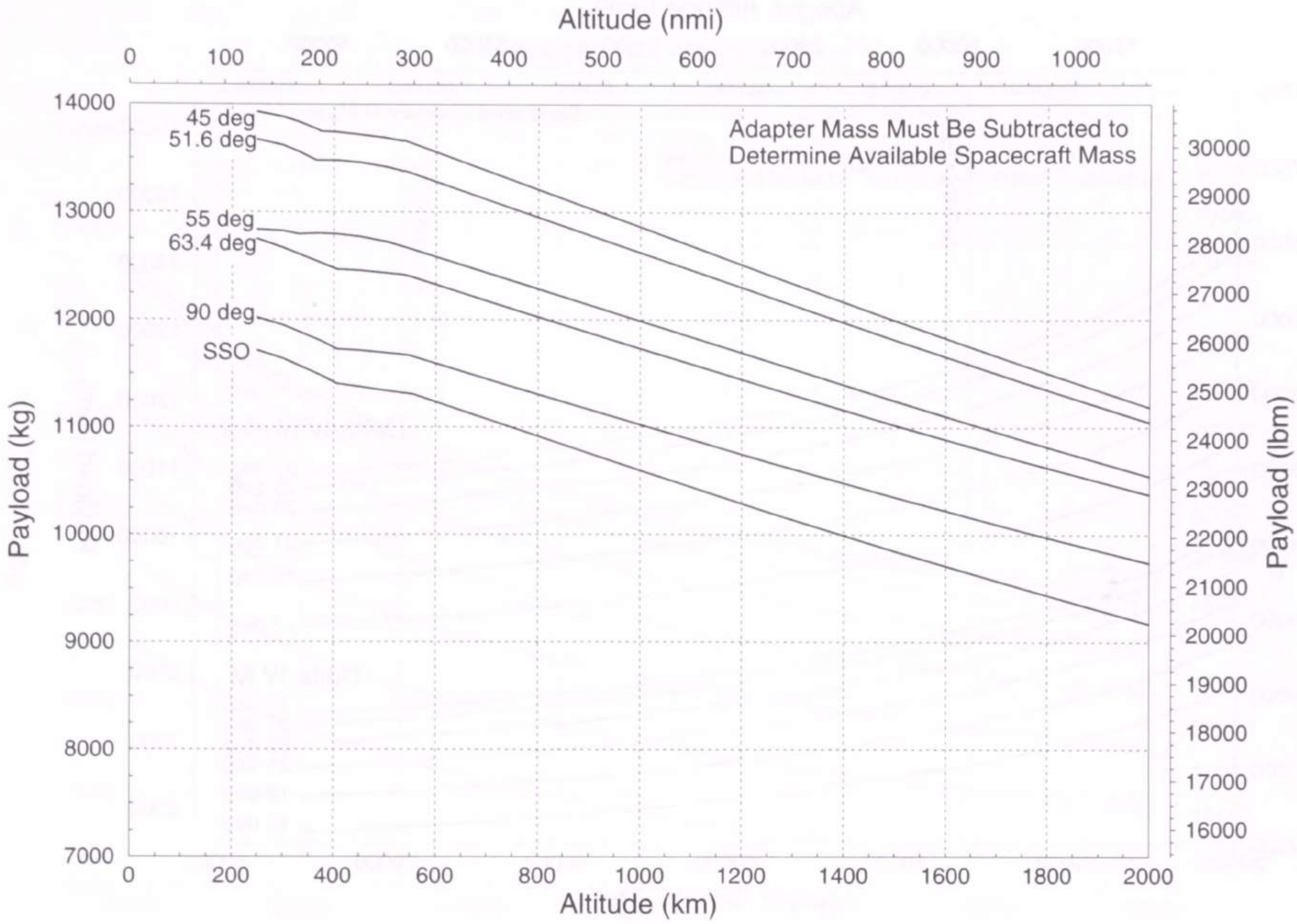


Delta IV M+(4,2): LEO Capability

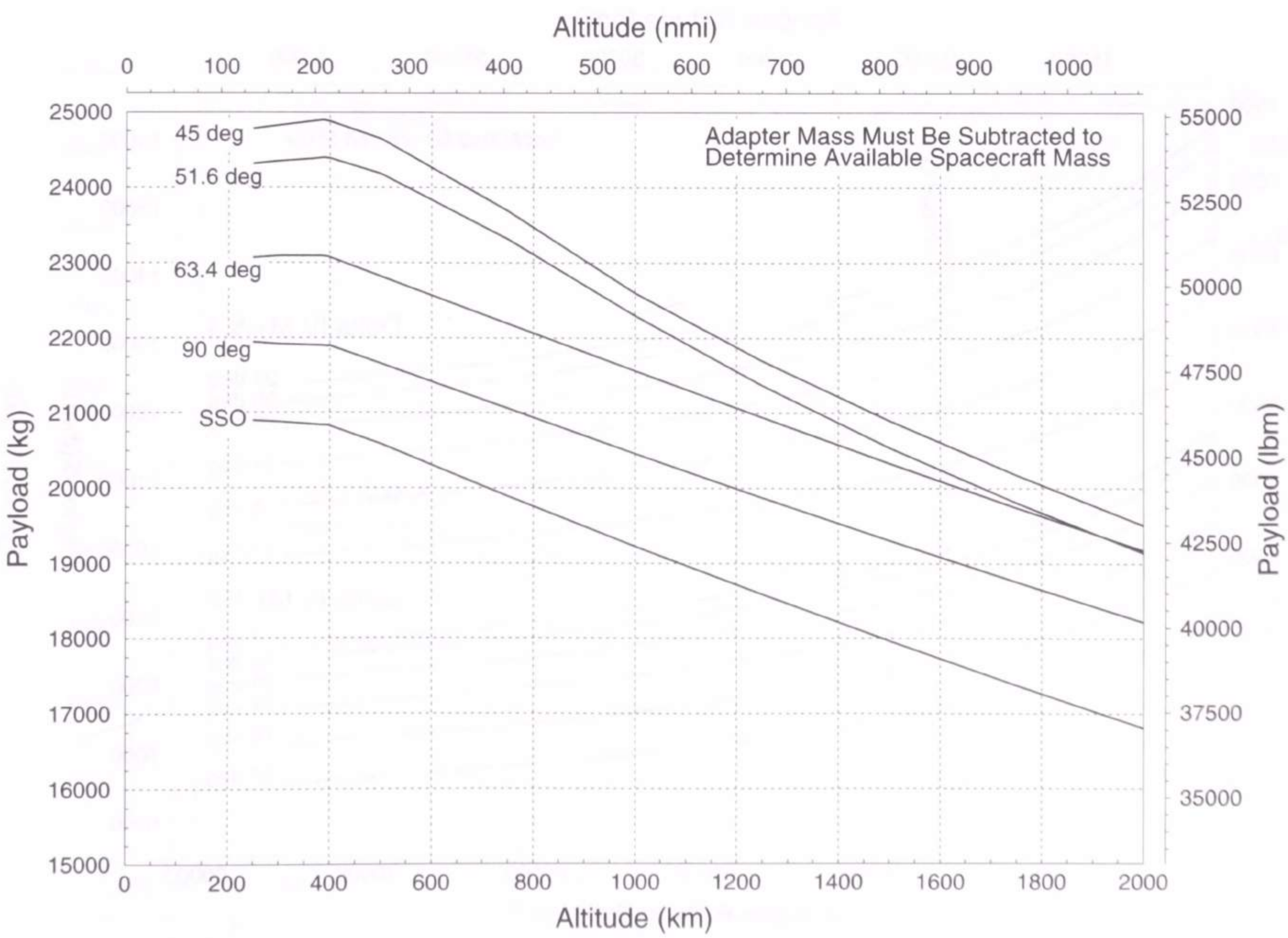


Delta IV M+(5,2): LEO Capability

Performance

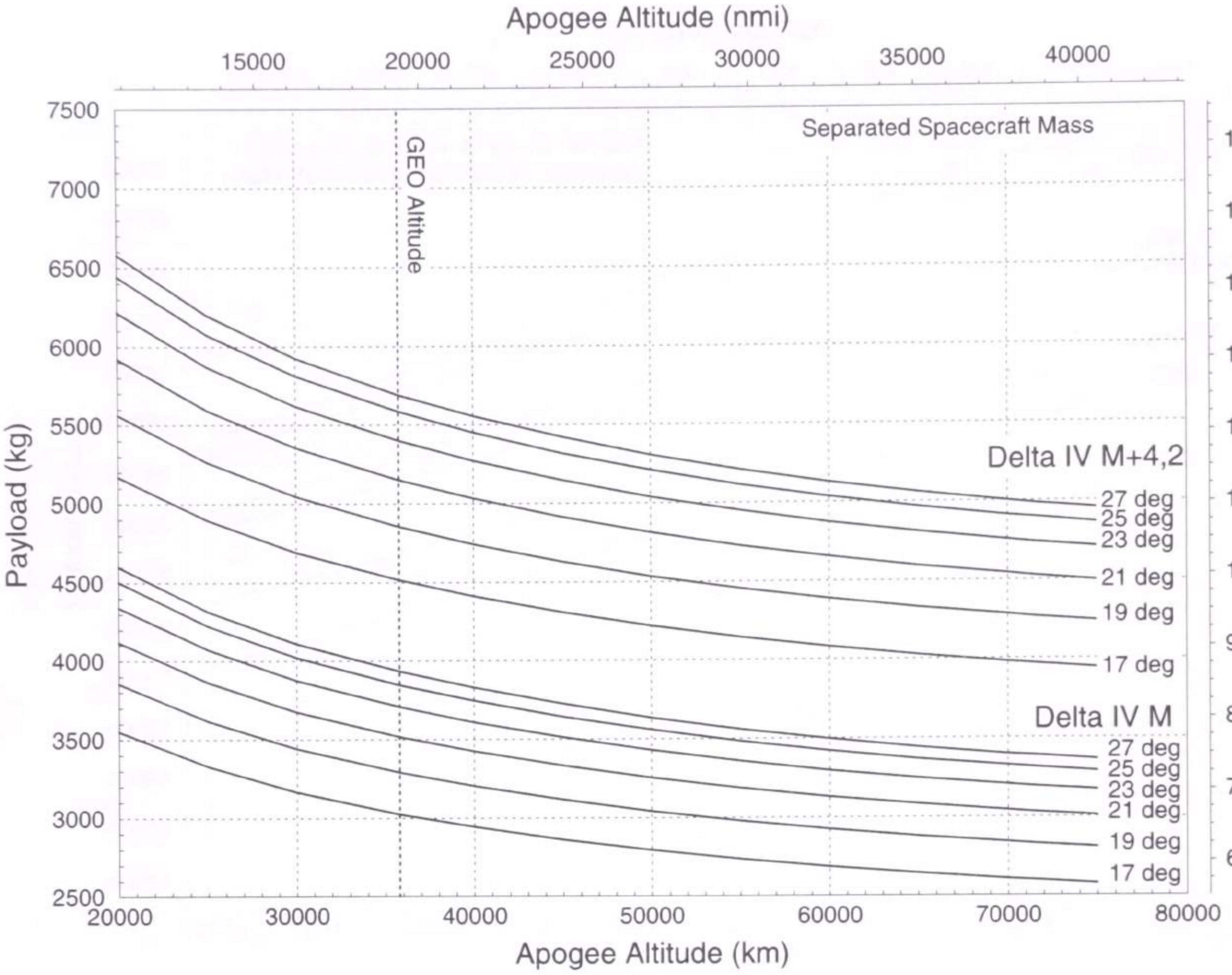


Delta IV M+(5,4): LEO Capability

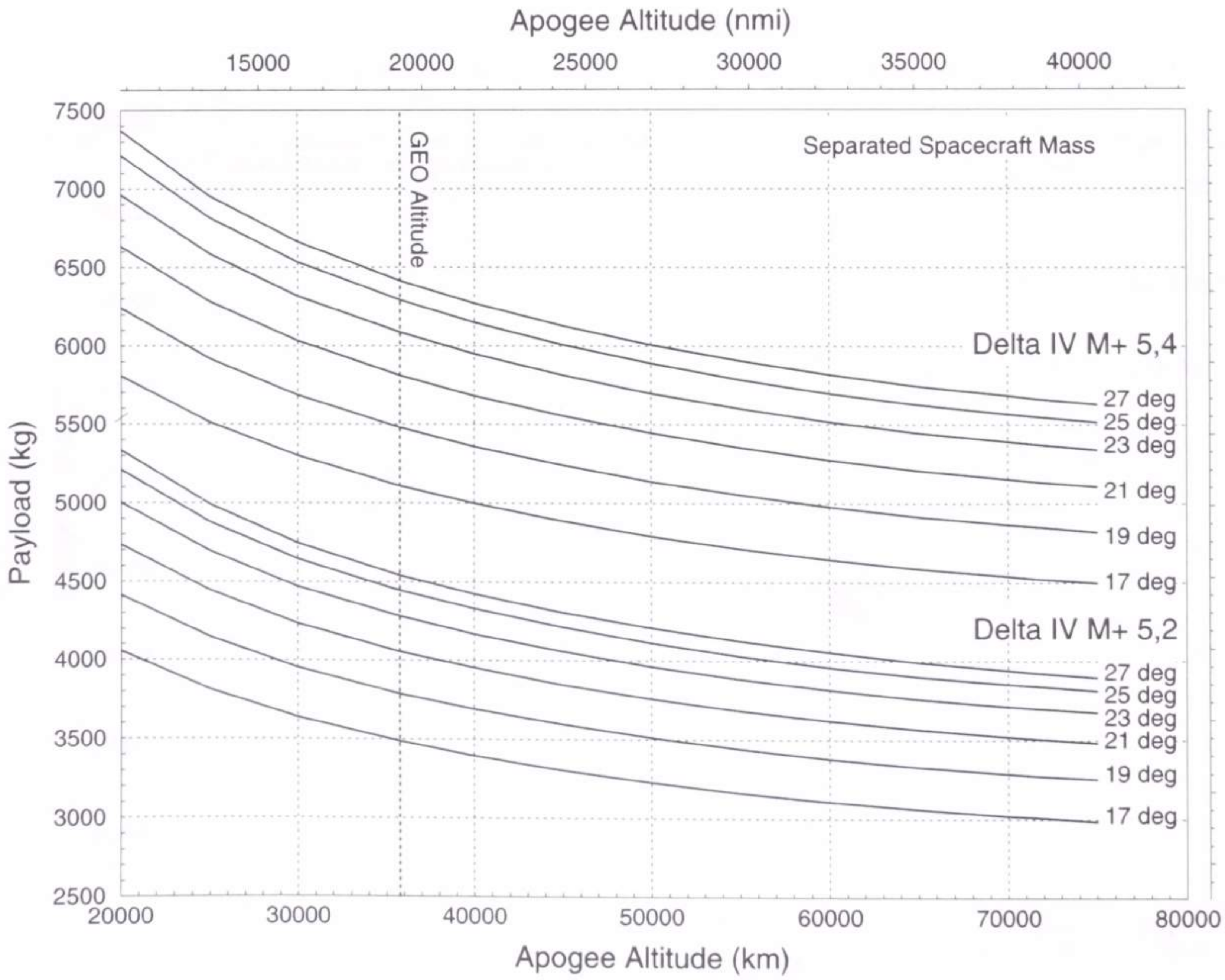


Delta IVHeavy: LEO Capability

PERFORMANCE

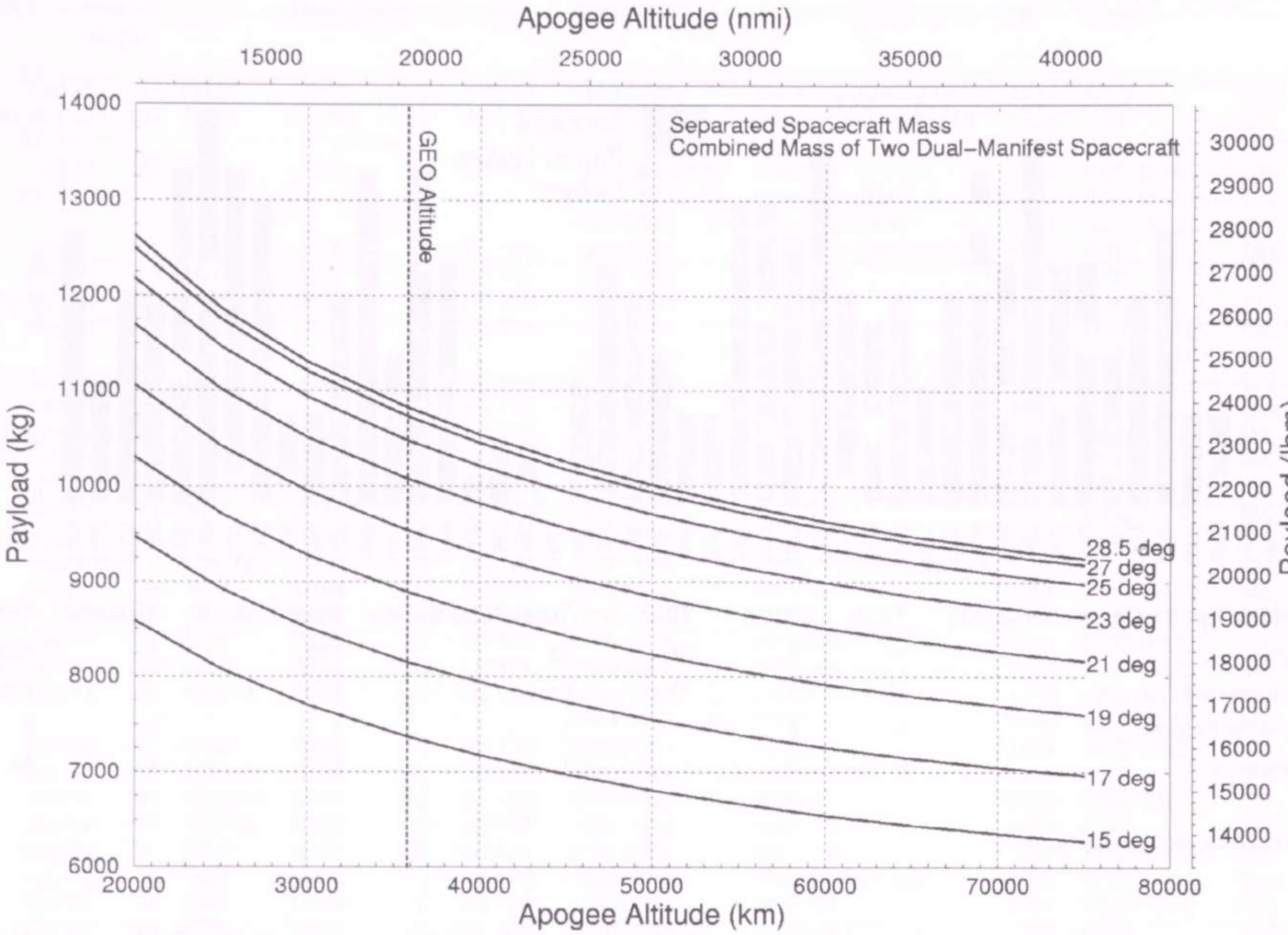


Delta IV M and Delta IV M+(4,2) GTO Capability

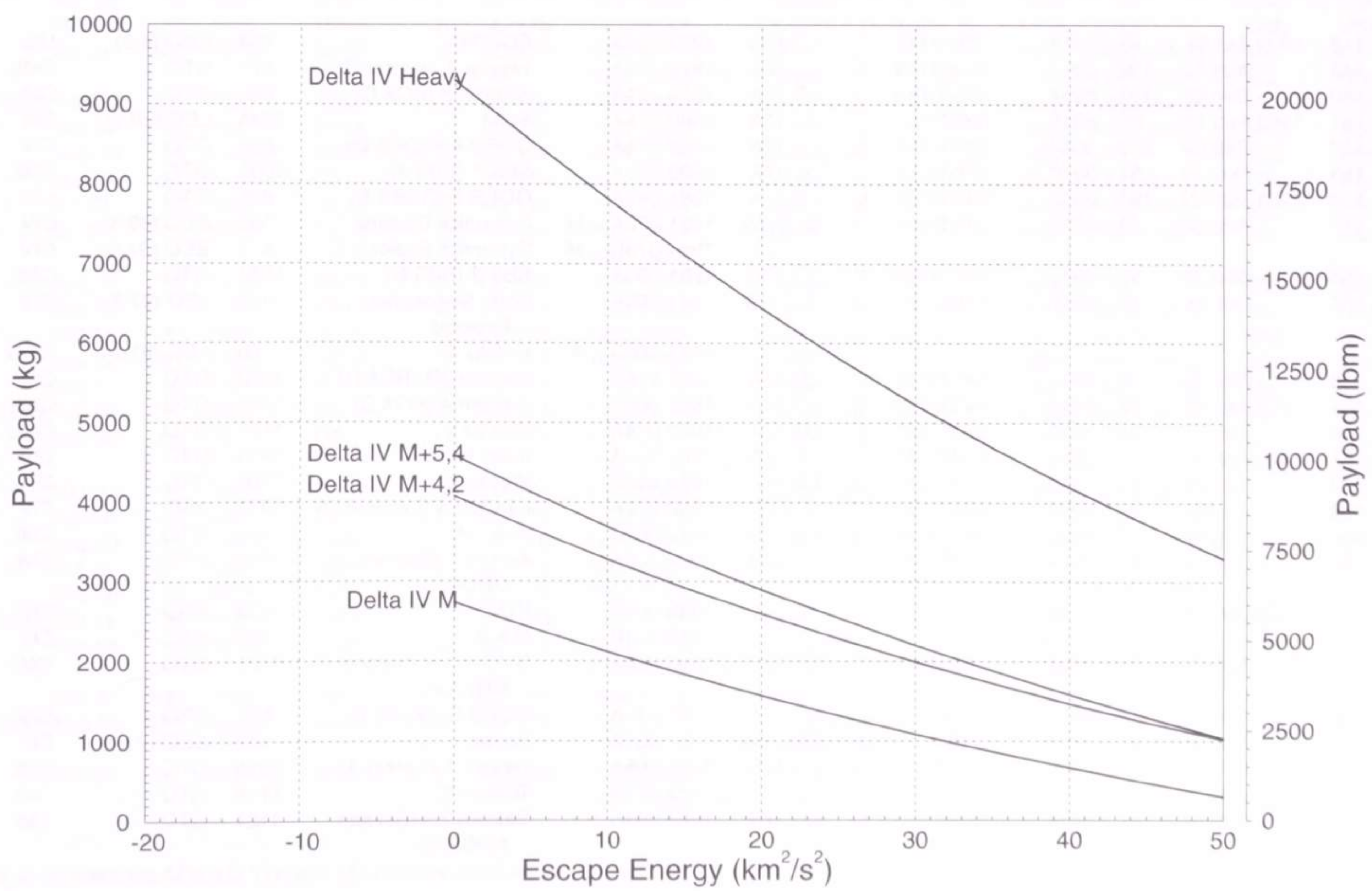


Delta IV M+(5,2) and Delta IV M+(5,4) GTO Capability

Performance



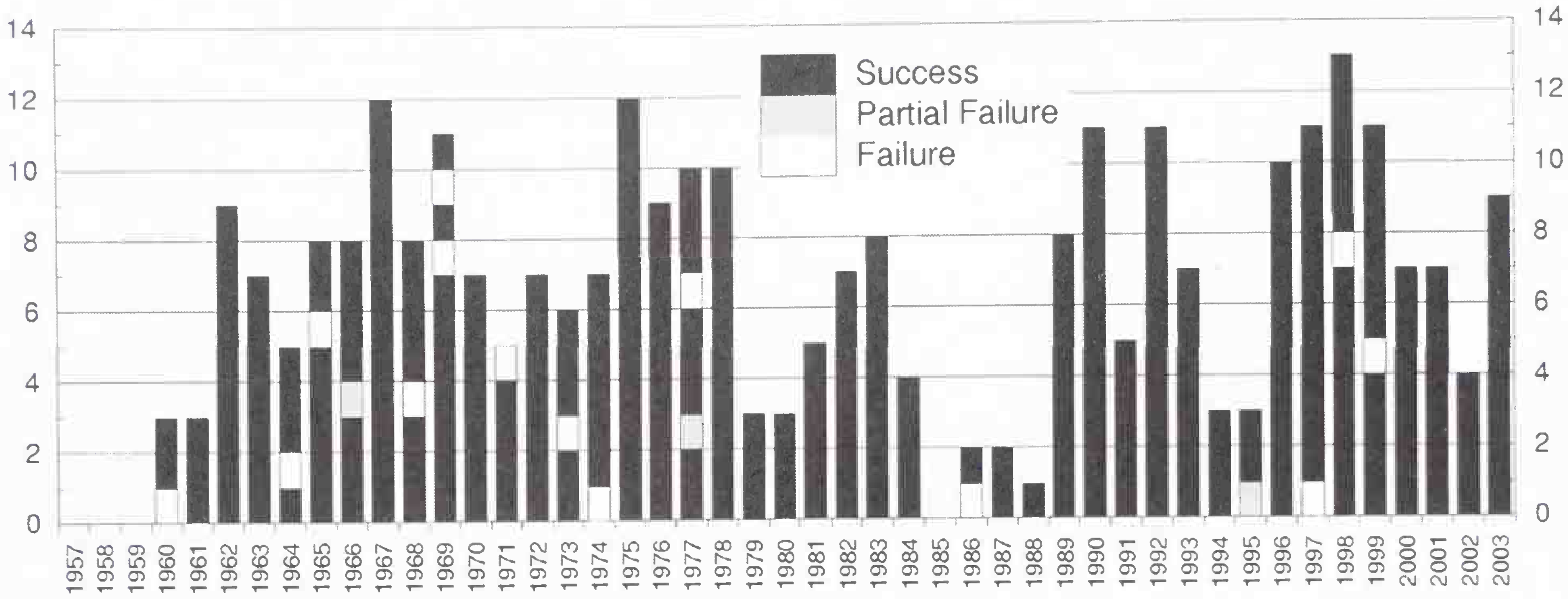
Delta IV Heavy GTO Capability



Delta IV Escape Energy Performance, Launched From Cape Canaveral

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	742X	732X	792X	Total 7000 Series	Total Delta II	Delta IV	Total Delta Family
Total Orbital Flights	12	7	74	93	110	3	302
Launch Vehicle Successes	12	7	72	91	108	3	285
Launch Vehicle Partial Failures	0	0	1	1	1	0	3
Launch Vehicle Failures	0	0	1	1	1	0	14

Delta II is Delta 7000 + Delta 6000 series vehicles

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	148	1979 Jan 30	45	2914	629/D148	C	LC 17B	1979 007A		SCATHA	659	EEO (4.9)	MIL	USA
	149	Aug 10	192	2914	638/D149	C	LC 17A	1979 072A		Westar 3 (Westar C)	574	GTO	CML	USA
S	150	Dec 07	119	3914	622/D150	C	LC 17A	1979 101A		Satcom 3 (RCA C)	895	GTO	CML	USA
	151	1980 Feb 14	69	3910	635/D151	C	LC 17A	1980 014A		SMM	2248	LEO (28.5)	CIV	USA
	152	Sep 09	208	3914	637/D152	C	LC 17A	1980 074A		GOES 4 (GOES D)	833	GTO	CIV	USA
	153	Nov 15	67	3910	636/D153	C	LC 17A	1980 091A		SBS 1 (SBS A)	1083	GTO	CML	USA
	154	1981 May 22	188	3914	645/D154	C	LC 17A	1981 049A		GOES 5 (GOES E)	835	GTO	CIV	USA
	155	Aug 03	73	3913	642/D155	V	SLC 2W	1981 070A	M	Dynamics Explorer 1	403	EEO (89.5)	CIV	USA
								1981 070B	M	Dynamics Explorer 2	415	EEO (97.6)	CIV	USA
	156	Sep 24	52	3910	641/D156	C	LC 17A	1981 096A		SBS 2 (SBS B)	1085	GTO	CML	USA
	157	Oct 06	12	2310	639/D157	V	SLC 2W	1981 100A		Solar Mesosphere Explorer	416	LEO (97.7)	CIV	USA
								1981 100B	A	UoSAT 1	60	LEO (97.7)	NGO	UK
	158	Nov 20	45	3910	640/D158	C	LC 17A	1981 114A		Satcom 3R (RCA D)	1082	GTO	CML	USA
	159	1982 Jan 16	57	3910	643/D159	C	LC 17A	1982 004A		Satcom 4 (RCA C)	1082	GTO	CML	USA
	160	Feb 26	41	3910	644/D160	C	LC 17A	1982 014A		Westar 4	1108	GTO	CML	USA
S	161	Apr 10	43	3910	647/D161	C	LC 17A	1982 031A		Insat 1A	1153	GTO	CIV	India
	162	Jun 09	60	3910	649/D162	C	LC 17A	1982 058A		Westar 5	1108	GTO	CML	USA
	163	Jul 16	37	3920	648/D163	V	SLC 2W	1982 072A		Landsat 4 (Landsat D)	1972	SSO	CIV	USA
	164	Aug 26	41	3920	651/D164	C	LC 17B	1982 082A		Anik D1	1238	GTO	CML	Canada
	165	Oct 28	63	3924	652/D165	C	LC 17B	1982 105A		Aurora 1 (Satcom 5/RCA E)	1102	GTO	CML	USA
	166	1983 Jan 26	90	3910	650/D166	V	SLC 2W	1983 004A		IRAS	1073	SSO	CIV	USA
								1983 004B		PIX 2	350	SSO	CIV	USA
	167	Apr 11	75	3924	653/D167	C	LC 17B	1983 030A		Satcom 1R (Satcom 6/RCA F)	1121	GTO	CML	USA
	168	Apr 28	17	3914	D168	C	LC 17A	1983 041A		GOES 6 (GOES F)	835	GTO	CIV	USA
	169	May 26	28	3914	D169	V	SLC 2W	1983 051A		Exosat	510	EEO (71.4)	CIV	Europe
	170	Jun 28	33	3920	D170	C	LC 17B	1983 065A		Galaxy 1 (Galaxy A)	1218	GTO	CML	USA
	171	Jul 28	30	3920	D171	C	LC 17A	1983 077A		Telstar 301	1218	GTO	CML	USA
	172	Sep 08	42	3924	D172	C	LC 17B	1983 094A		Satcom 2R (Satcom 7/RCA G)	1121	GTO	CML	USA

SLC 2W is at Vandenberg AFB; LC 17A and LC 17B are at CCAFS

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	173	Sep 22	14	3920	D173	C	LC 17A	1983 098A		Galaxy 2 (Galaxy B)	1218	GTO	CML	USA
	174	1984 Mar 01	161	3920	D174	V	SLC 2W	1984 021A		Landsat 5 (Landsat D)	1938	SSO	CIV	USA
								1984 021B	A	UoSAT 2	52	SSO	NGO	UK
	175	Aug 16	168	3924	D175	C	LC 17A	1984 088A	M	Ampte 1	242	EEO (3.8)	CIV	USA
								1984 088B	M	Ampte 2 (IRM)	605	EEO (27)	CIV	Germany
								1984 088C	M	Ampte 3 (UKS)	77	EEO (26.9)	CIV	UK
	176	Sep 21	36	3920	D176	C	LC 17B	1984 101A		Galaxy 3 (Galaxy C)	1218	GTO	CML	USA
	177	Nov 14	54	3914	D177	C	LC 17A	1984 115A		NATO 3D	760	GTO	MIL	Europe
F	**178**	**1986 May 03**	**535**	**3914**	**D178**	**C**	**LC 17A**	1986 F04A		GOES G	838	GTO	CIV	USA
	179	Sep 05	125	3920	D180	C	LC 17B	1986 069A		USA 19	2495	LEO (39.1)	MIL	USA
								1986 069B		USA 19 Target		LEO (22.8)	MIL	USA
	180	1987 Feb 26	174	3914	D179	C	LC 17A	1987 022A		GOES 7 (GOES H)	841	GTO	CIV	USA
	181	Mar 20	22	3920	D182	C	LC 17B	1987 029A		Palapa BP2	1244	GTO	CML	Indonesia
	182	1988 Feb 08	325	3910	D181	C	LC 17B	1988 008A		USA 30	1574	LEO (28.6)	MIL	USA
	183	1989 Feb 14	372	6925	D184	C	LC 17A	1989 013A		GPS 2-1	1657	MEO (55.1)	MIL	USA
	184	Mar 24	38	3920–8	D183	C	LC 17B	1989 026A		USA 36	2637	LEO (47.7)	MIL	USA
	185	Jun 10	78	6925	D185	C	LC 17A	1989 044A		GPS 2-2	1657	MEO (54.7)	MIL	USA
	186	Aug 18	69	6925	D186	C	LC 17A	1989 064A		GPS 2-3	1664	MEO (55)	MIL	USA
	187	Aug 27	9	4925–8	D187	C	LC 17B	1989 067A		Sirius 1 (Marcopolo 1)	1233	GTO	CML	UK
	188	Oct 21	55	6925	D188	C	LC 17A	1989 085A		GPS 2-4	1664	MEO (54.7)	MIL	USA
	189	Nov 18	28	5920–8	D189	V	SLC 2W	1989 089A		COBE	2203	SSO	CIV	USA
	190	Dec 11	23	6925	D190	C	LC 17B	1989 097A		GPS 2-5	1664	MEO (54.9)	MIL	USA
	191	1990 Jan 24	44	6925	D191	C	LC 17A	1990 008A		GPS 2-6	1664	MEO (54.6)	MIL	USA
	192	Feb 14	21	6920–8	D192	C	LC 17B	1990 015A	C	LACE	1430	LEO (43.1)	MIL	USA
								1990 015B	C	RME	1040	LEO (43.1)	MIL	USA
	193	Mar 26	40	6925	D193	C	LC 17A	1990 025A		GPS 2-7	1664	MEO (55.2)	MIL	USA
	194	Apr 13	18	6925–8	D194	C	LC 17B	1990 034A		Palapa B2R	1241	GTO	CML	Indonesia
	195	Jun 01	49	6920–10	D195	C	LC 17A	1990 049A		ROSAT	2440	LEO (53)	CIV	Germany
	196	Jun 12	11	4925–8	D196	C	LC 17B	1990 051A		Insat 1D	1293	GTO	CIV	India
	197	Aug 02	51	6925	D197	C	LC 17A	1990 068A		GPS 2-8	1664	MEO (54.7)	MIL	USA
	198	Aug 18	16	6925	D198	C	LC 17B	1990 074A		Thor 1 (Marcopolo 2)	1233	GTO	CML	UK
	199	Oct 01	44	6925	D199	C	LC 17A	1990 088A		GPS 2-9	1664	MEO (54.9)	MIL	USA
	200	Oct 30	29	6925	D200	C	LC 17B	1990 093A		Inmarsat 201	1370	GTO	CML	Int'l
	201	Nov 26	27	7925	D201	C	LC 17A	1990 103A		GPS 2-10	1882	MEO (54.9)	MIL	USA
	202	1991 Jan 08	43	7925	D202	C	LC 17B	1991 001A		NATO 4A	1434	GTO	MIL	Europe
	203	Mar 08	59	6925	D203	C	LC 17B	1991 018A		Inmarsat 202	1385	GTO	CML	USA
	204	Apr 13	36	7925	D204	C	LC 17B	1991 028A		Spacenet 4 (ASC 2)	1358	GTO	CML	USA
	205	May 29	46	7925	D205	C	LC 17B	1991 037A		Aurora 2 (Satcom C5)	1338	GTO	CML	USA
	206	Jul 04	36	7925	D206	C	LC 17A	1991 047A		GPS 2-11	1882	MEO (55.3)	MIL	USA
								1991 047B	A	Losat X	75	LEO (40)	MIL	USA
	207	1992 Feb 23	234	7925	D207	C	LC 17B	1992 009A		GPS 2-12	1882	MEO (54.5)	MIL	USA
	208	Apr 10	47	7925	D208	C	LC 17B	1992 019A		GPS 2-13	1882	MEO (55.3)	MIL	USA
	209	May 14	34	7925–8	D209	C	LC 17B	1992 027A		Palapa B4	1252	GTO	CML	Indonesia
	210	Jun 07	24	6920–10	D210	C	LC 17A	1992 031A		EUVE	3249	LEO (28.4)	CIV	USA
	211	Jul 07	30	7925	D211	C	LC 17B	1992 039A		GPS 2-14	1882	MEO (55)	MIL	USA
	212	Jul 24	17	6925	D212	C	LC 17A	1992 044A		Geotail	1008	EEO (18)	CIV	Japan
								1992 044B		DUVE				
	213	Aug 31	38	7925	D213	C	LC 17B	1992 057A		Satcom C4	1402	GTO	CML	USA
	214	Sep 09	9	7925	D214	C	LC 17A	1992 058A		GPS 2-15	1882	MEO (54.5)	MIL	USA
	215	Oct 12	33	7925	D215	C	LC 17B	1992 066A		DFS Kopernikus 3	1411	GTO	CML	Germany
	216	Nov 22	41	7925	D216	C	LC 17A	1992 079A		GPS 2-16	1882	MEO (54.8)	MIL	USA
	217	Dec 18	26	7925	D217	C	LC 17B	1992 089A		GPS 2-17	1882	MEO (54.8)	MIL	USA
	218	1993 Feb 03	47	7925	D218	C	LC 17A	1993 007A		GPS 2-18	1882	MEO (54.8)	MIL	USA
	219	Mar 30	55	7925	D219	C	LC 17A	1993 017A		GPS 2-19	1882	MEO (55)	MIL	USA
								1993 017B	A	SEDS 1	25		NGO	USA
	220	May 13	44	7925	D220	C	LC 17A	1993 032A		GPS 2-20	1882	MEO (55.1)	MIL	USA
	221	Jun 26	44	7925	D221	C	LC 17A	1993 042A		GPS 2-21	1882	MEO (54.6)	MIL	USA
	222	Aug 30	65	7925	D222	C	LC 17B	1993 054A		GPS 2-22	1882	MEO (54.9)	MIL	USA
	223	Oct 26	57	7925	D223	C	LC 17B	1993 068A		GPS 2-23	1882	MEO (55.1)	MIL	USA
	224	Dec 08	43	7925	D224	C	LC 17B	1993 076A		NATO 4B	1434	GTO	MIL	Europe
	225	1994 Feb 19	73	7925–8	D225	C	LC 17B	1994 013A		Galaxy 1R	1400	GTO	CML	USA
	226	Mar 10	19	7925	D226	C	LC 17A	1994 016A		GPS 2-24	1882	MEO (55)	MIL	USA
								1994 016B	A	SEDS 2	38	LEO (32.3)	NGO	USA
	227	Nov 01	236	7925	D227	C	LC 17B	1994 071A		WIND	1250	EEO (28.7)	CIV	USA
P	**228**	**1995 Aug 05**	**277**	**7925**	**D228**	**C**	**LC 17B**	1995 041A		Mugunghwa 1 (Koreasat 1)	1447	GTO	CML	Korea
	229	Nov 04	91	7920–10	D229	V	SLC 2W	1995 059A		Radarsat 1	2713	SSO	CIV	Canada
								1995 059B	A	Surfsat	55	SSO	NGO	USA

SLC 2W is at Vandenberg AFB; LC 17A and LC 17B are at CCAFS

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	230	Dec 30	56	7920A–10	D230	C	LC 17A	1995 074A		Rossi XTE	3030	LEO (23)	CIV	USA
	231	1996 Jan 14	15	7925	D231	C	LC 17B	1996 003A		Mugungwha 2 (Koreasat 2)	1457	GTO	CML	Korea
	232	Feb 17	34	7925–8	D232	C	LC 17B	1996 008A		NEAR	806	Asteroid Eros	CIV	USA
	233	Feb 24	7	7925–10	D233	V	SLC 2W	1996 013A		Polar	1301	EEO (86.3)	CIV	USA
	234	Mar 28	33	7925	D234	C	LC 17B	1996 019A		GPS 2-25	1884	MEO (54.7)	MIL	USA
	235	Apr 24	27	7920–10	D235	V	SLC 2W	1996 024A		MSX	2800	SSO	MIL	USA
	236	May 24	30	7925	D236	C	LC 17B	1996 033A		Galaxy 9	1395	GTO	CML	USA
	237	Jul 16	53	7925	D237	C	LC 17A	1996 041A		GPS 2-26	1881	MEO (55.1)	MIL	USA
	238	Sep 12	58	7925	D238	C	LC 17A	1996 056A		GPS 2-27	1881	MEO (54.7)	MIL	USA
	239	Nov 07	56	7925	D239	C	LC 17A	1996 062A		Mars Global Surveyor	1060	Mars	CIV	USA
	240	Dec 04	27	7925	D240	C	LC 17B	1996 068A		Mars Pathfinder/ Sojourner	990	Mars	CIV	USA
F	**241**	**1997 Jan 17**	**44**	**7925**	**D241**	**C**	**LC 17A**	1997 F01A		GPS 42		MEO (55)	MIL	USA
	242	May 05	108	7920–10C	D242	V	SLC 2W	1997 020A	M	Iridium 008	690	LEO (87)	CML	USA
								1997 020B	M	Iridium 007	690	LEO (87)	CML	USA
								1997 020C	M	Iridium 006	690	LEO (87)	CML	USA
								1997 020D	M	Iridium 005	690	LEO (87)	CML	USA
								1997 020E	M	Iridium 004	690	LEO (87)	CML	USA
	243	May 20	15	7925	D243	C	LC 17A	1997 025A		Thor 2	1248	GTO	CML	Norway
	244	Jul 09	50	7920–10C	D244	V	SLC 2W	1997 034A	M	Iridium 015	690	LEO (87)	CML	USA
								1997 034B	M	Iridium 017	690	LEO (87)	CML	USA
								1997 034C	M	Iridium 018	690	LEO (87)	CML	USA
								1997 034D	M	Iridium 020	690	LEO (87)	CML	USA
S								1997 034E	M	Iridium 021	690	LEO (87)	CML	USA
	245	Jul 23	14	7925	D245	C	LC 17A	1997 035A		GPS 43	2028	MEO (55)	MIL	USA
	246	Aug 21	29	7920–10C	D246	V	SLC 2W	1997 043A	M	Iridium 026	690	LEO (87)	CML	USA
								1997 043B	M	Iridium 025	690	LEO (87)	CML	USA
S								1997 043C	M	Iridium 024	690	LEO (87)	CML	USA
								1997 043D	M	Iridium 023	690	LEO (87)	CML	USA
								1997 043E	M	Iridium 022	690	LEO (87)	CML	USA
	247	Aug 25	4	7920–8	D247	C	LC 17A	1997 045A		ACE	765	L1 Halo	CIV	USA
	248	Sep 27	33	7920–10C	D248	V	SLC 2W	1997 056A	M	Iridium 019	690	LEO (87)	CML	USA
								1997 056B	M	Iridium 037	690	LEO (87)	CML	USA
								1997 056C	M	Iridium 036	690	LEO (87)	CML	USA
								1997 056D	M	Iridium 035	690	LEO (87)	CML	USA
								1997 056E	M	Iridium 034	690	LEO (87)	CML	USA
	249	Nov 06	40	7925	D249	C	LC 17A	1997 067A		GPS 38	1881	MEO (55)	MIL	USA
	250	Nov 09	3	7920–10C	D250	V	SLC 2W	1997 069A	M	Iridium 038	690	LEO (87)	CML	USA
								1997 069B	M	Iridium 039	690	LEO (87)	CML	USA
								1997 069C	M	Iridium 040	690	LEO (87)	CML	USA
								1997 069D	M	Iridium 041	690	LEO (87)	CML	USA
								1997 069E	M	Iridium 043	690	LEO (87)	CML	USA
	251	Dec 20	41	7920–10C	D251	V	SLC 2W	1997 082A	M	Iridium 045	690	LEO (87)	CML	USA
								1997 082B	M	Iridium 046	690	LEO (87)	CML	USA
								1997 082C	M	Iridium 047	690	LEO (87)	CML	USA
								1997 082D	M	Iridium 048	690	LEO (87)	CML	USA
								1997 082E	M	Iridium 049	690	LEO (87)	CML	USA
	252	1998 Jan 10	21	7925	D252	C	LC 17B	1998 002A		Skynet 4D	1500	GTO	MIL	UK
	253	Feb 18	39	7920–10C	D254	V	SLC 2W	1998 010A	M	Iridium 050	690	LEO (87)	CML	USA
								1998 010B	M	Iridium 052	690	LEO (87)	CML	USA
								1998 010C	M	Iridium 053	690	LEO (87)	CML	USA
								1998 010D	M	Iridium 054	690	LEO (87)	CML	USA
								1998 010E	M	Iridium 055	690	LEO (87)	CML	USA
	254	Mar 30	40	7920–10C	D255	V	SLC 2W	1998 019A	M	Iridium 055	690	LEO (87)	CML	USA
								1998 019B	M	Iridium 057	690	LEO (87)	CML	USA
								1998 019C	M	Iridium 058	690	LEO (87)	CML	USA
								1998 019D	M	Iridium 059	690	LEO (87)	CML	USA
								1998 019E	M	Iridium 060	690	LEO (87)	CML	USA
	255	Apr 14	15	7420–10C	D253	C	LC 17A	1998 008A	M	Globalstar 1	450	LEO (52)	CML	USA
								1998 008B	M	Globalstar 2	450	LEO (52)	CML	USA
								1998 008C	M	Globalstar 3	450	LEO (52)	CML	USA
								1998 008D	M	Globalstar 4	450	LEO (52)	CML	USA
	256	Apr 24	10	7420–10C	D256	C	LC 17A	1998 023A	M	Globalstar 6	450	LEO (52)	CML	USA
								1998 023B	M	Globalstar 8	450	LEO (52)	CML	USA
								1998 023C	M	Globalstar 14	450	LEO (52)	CML	USA
								1998 023D	M	Globalstar 15	450	LEO (52)	CML	USA

SLC 2W is at Vandenberg AFB; LC 17A and LC 17B are at CCAFS

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	257	May 17	23	7920–10C	D257	V	SLC 2W	1998 032A	M	Iridium 070	690	LEO (87)	CML	USA
								1998 032B	M	Iridium 072	690	LEO (87)	CML	USA
								1998 032C	M	Iridium 073	690	LEO (87)	CML	USA
								1998 032D	M	Iridium 074	690	LEO (87)	CML	USA
								1998 032E	M	Iridium 075	690	LEO (87)	CML	USA
	258	Jun 10	24	7925–9.5	D258	C	LC 17A	1998 035A		Thor 3	1450	GTO	CML	Norway
F	**259**	**Aug 27**	**78**	**8930**	**D3-1**	**C**	**LC 17B**	1998 F03A		Galaxy 10	3876	GTO	CML	USA
	260	Sep 08	12	7920–10C	D259	V	SLC 2W	1998 051A	M	Iridium 082	690	LEO (87)	CML	USA
								1998 051B	M	Iridium 081	690	LEO (87)	CML	USA
								1998 051C	M	Iridium 080	690	LEO (87)	CML	USA
S								1998 051D	M	Iridium 079	690	LEO (87)	CML	USA
								1998 051E	M	Iridium 077	690	LEO (87)	CML	USA
	261	Oct 24	46	7325–9.5	D261	C	LC 17A	1998 061A		Deep Space 1	486	asteroid Braille	CIV	USA
								1998 061B	A	SEDSat	34	LEO (31.4)	NGO	USA
	262	Nov 06	13	7920–10C	D262	V	SLC 2W	1998 066A	M	Iridium 002	690	LEO (86)	CML	USA
								1998 066B	M	Iridium 086	690	LEO (86)	CML	USA
								1998 066C	M	Iridium 085	690	LEO (86)	CML	USA
								1998 066D	M	Iridium 084	690	LEO (86)	CML	USA
								1998 066E	M	Iridium 083	690	LEO (86)	CML	USA
	263	Nov 22	16	7925	D263	C	LC 17B	1998 068A		Bonum 1	1452	GTO	CML	Russia
	264	Dec 11	19	7425	D264	C	LC 17A	1998 073A		Mars Climate Orbiter	629	Mars	CIV	USA
	265	1999 Jan 03	23	7425–9.5	D265	C	LC 17B	1999 001A		Mars Polar Lander	574	Mars	CIV	USA
										Deep Space 2		Mars	CIV	USA
	266	Feb 07	35	7426	D266	C	LC 17A	1999 003A		Stardust	380	Comet Wild	CIV	USA
	267	Feb 23	16	7920	D267	V	SLC 2W	1999 008A		ARGOS	2491	SSO	MIL	USA
								1999 008B	A	Ørsted	62	SSO	CIV	Denmark
								1999 008C	A	SUNSat	64	SSO	NGO	South Africa
	268	Apr 15	51	7920–10	D268	V	SLC 2W	1999 020A		Landsat 7	2200	SSO	CIV	USA
F	**269**	**May 05**	**20**	**8930**	**D3-2**	**C**	**LC 17B**	1999 024A		Orion 3	4300	GTO	CML	USA
	270	Jun 10	36	7420–10	D270	C	LC 17B	1999 031A	M	Globalstar 52	450	LEO (52)	CML	USA
								1999 031B	M	Globalstar 49	450	LEO (52)	CML	USA
								1999 031C	M	Globalstar 25	450	LEO (52)	CML	USA
								1999 031D	M	Globalstar 47	450	LEO (52)	CML	USA
	271	Jun 24	14	7320–10	D271	C	LC 17A	1999 035A		FUSE	1360	LEO (25)	CIV	USA
	272	Jul 10	16	7420 - 10	D272	C	LC 17B	1999 037A	M	Globalstar 32	450	LEO (52)	CML	USA
								1999 037B	M	Globalstar 30	450	LEO (52)	CML	USA
								1999 037C	M	Globalstar 35	450	LEO (52)	CML	USA
								1999 037D	M	Globalstar 51	450	LEO (52)	CML	USA
	273	Jul 25	15	7420 - 10	D273	C	LC 17A	1999 041A	M	Globalstar 26	450	LEO (52)	CML	USA
								1999 041B	M	Globalstar 28	450	LEO (52)	CML	USA
								1999 041C	M	Globalstar 43	450	LEO (52)	CML	USA
								1999 041D	M	Globalstar 48	450	LEO (52)	CML	USA
	274	Aug 17	23	7420 - 10	D274	C	LC 17B	1999 043A	M	Globalstar 24	450	LEO (52)	CML	USA
								1999 043B	M	Globalstar 27	450	LEO (52)	CML	USA
								1999 043C	M	Globalstar 53	450	LEO (52)	CML	USA
								1999 043D	M	Globalstar 54	450	LEO (52)	CML	USA
	275	Oct 07	51	7925	D275	C	LC 17A	1999 055A		GPS 2R-3	1075	MEO (55)	MIL	USA
	276	2000 Feb 08	124	7420 - 10	D276	C	LC 17B	2000 008A	M	Globalstar 60	450	LEO (52)	CML	USA
								2000 008B	M	Globalstar 62	450	LEO (52)	CML	USA
								2000 008C	M	Globalstar 63	450	LEO (52)	CML	USA
								2000 008D	M	Globalstar 64	450	LEO (52)	CML	USA
	277	Mar 25	46	7326	D277	V	SLC 2W	2000 017A		IMAGE	494	EEO (93.9)	CIV	USA
	278	May 11	47	7925	D278	C	LC 17A	2000 025A		GPS 2R-4	2032	MEO (55)	MIL	USA
	279	Jul 16	66	7925	D279	C	LC 17A	2000 040A		GPS 2R-5	2032	MEO (55)	MIL	USA
T	280	Aug 23	38	8930	D280	C	LC 17B	2000 048A		DM-F3	4348	GTO	CML	USA
	281	Nov 10	79	7925	D281	C	LC 17A	2000 071A		GPS 2R-6	2032	MEO (55)	MIL	USA
	282	Nov 21	11	7320 - 10	D282	V	SLC 2W	2000 075A	C	EO 1	529	SSO	CIV	USA
								2000 075B	C	SAC C	425	SSO	CIV	Argentina
								2000 075C	A	Munin	7.5	SSO	CIV	Sweden
	283	2001 Jan 30	70	7925	D283	C	LC 17A	2001 004A		GPS 2R-7 (54)	2032	MEO (55)	MIL	USA
	284	Apr 07	67	7925	D284	C	LC 17A	2001 013A		Mars Odyssey	758	Mars	CIV	USA
	285	May 18	41	7925	D285	C	LC 17B	2001 020A		GeoLITE	1800	GTO	MIL	USA
	286	Jun 30	43	7425 - 10	D286	C	LC 17B	2001 027A		MAP	840	L2	CIV	USA
	287	Aug 08	39	7326	D287	C	LC 17A	2001 034A		Genesis	636	L1	CIV	USA
	288	Oct 18	71	7320 - 10	D288	V	SLC 2W	2001 047A		QuickBird	950	SSO	CML	USA

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Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	289	Dec 07	50	7920 - 10	D289	V	SLC 2W	2001 055A		Jason 1	495	LEO (66)	CIV	USA
		2000				V	SLC 2W	2001 055B	A	TIMED	587	LEO (74)	CIV	USA
	290	2002 Feb 11	66	7920 - 10	D290	V	SLC 2W	2002 005A	M	Iridium 90	725	LEO (86.4)	CML	USA
					D290	V	SLC 2W	2002 005B	M	Iridium 91	725	LEO (86.4)	CML	USA
					D290	V	SLC 2W	2002 005C	M	Iridium 94	725	LEO (86.4)	CML	USA
					D290	V	SLC 2W	2002 005D	M	Iridium 95	725	LEO (86.4)	CML	USA
					D290	V	SLC 2W	2002 005E	M	Iridium 96	725	LEO (86.4)	CML	USA
	291	May 04	82	7920 - 10L	D291	V	SLC 2W	2002 022A		Aqua	2935	SSO	CIV	USA
S	292	Jul 03	60	7425	D292	C	LC 17A	2002 034A		CONTOUR	967	Earth Escape	CIV	USA
	293	Nov 20	140	4 M+ (4,2)	D293	C	SLC 37B	2002 051A		Eutelsat W1	3163	GTO	CML	France
	294	2003 Jan 12	53	7320-10C	D294	V	SLC 2W	2003-002A	M	ICEsat	959	LEO (94)	CVL	USA
								2003-002B	M	CHIPsat	85	LEO (94)	CVL	USA
	295	Jan 29	17	7925	D295	C	LC 17B	2003-005A		GPS IIR-8	2032	MEO (55)	MIL	USA
								2003-005B	A	XSS-10	28	LEO (40)	MIL	USA
	296	Mar 10	40	Medium	D296	C	SLC 37B	2003-008A		DSCS III A3	1244	GTO	MIL	USA
	297	Mar 31	21	7925	D297	C	LC 17A	2003-010A		GPS IIR-9	2032	MEO (55)	MIL	USA
	298	Jun 10	71	7925	D298	C	LC 17A	2003-027A		MER-A (Spirit)	1062	Mars	CVL	USA
	299	Jul 7	27	7925H	D299	C	LC 17B	2003-032A		MER-B (Opportunity)	1062	Mars	CVL	USA
	300	Aug 25	49	7920H	D300	C	LC 17B	2003-038A		SIRTF	924	SSO	CVL	USA
	301	Aug 29	4	Medium	D301	C	SLC 37B	2003-040A		DSCS III B6	2377	GTO	MIL	USA
	302	Dec 21	114	7925	D302	C	LC 17A	2003-058A		GPS IIR-10	2032	MEO (55)	MIL	USA

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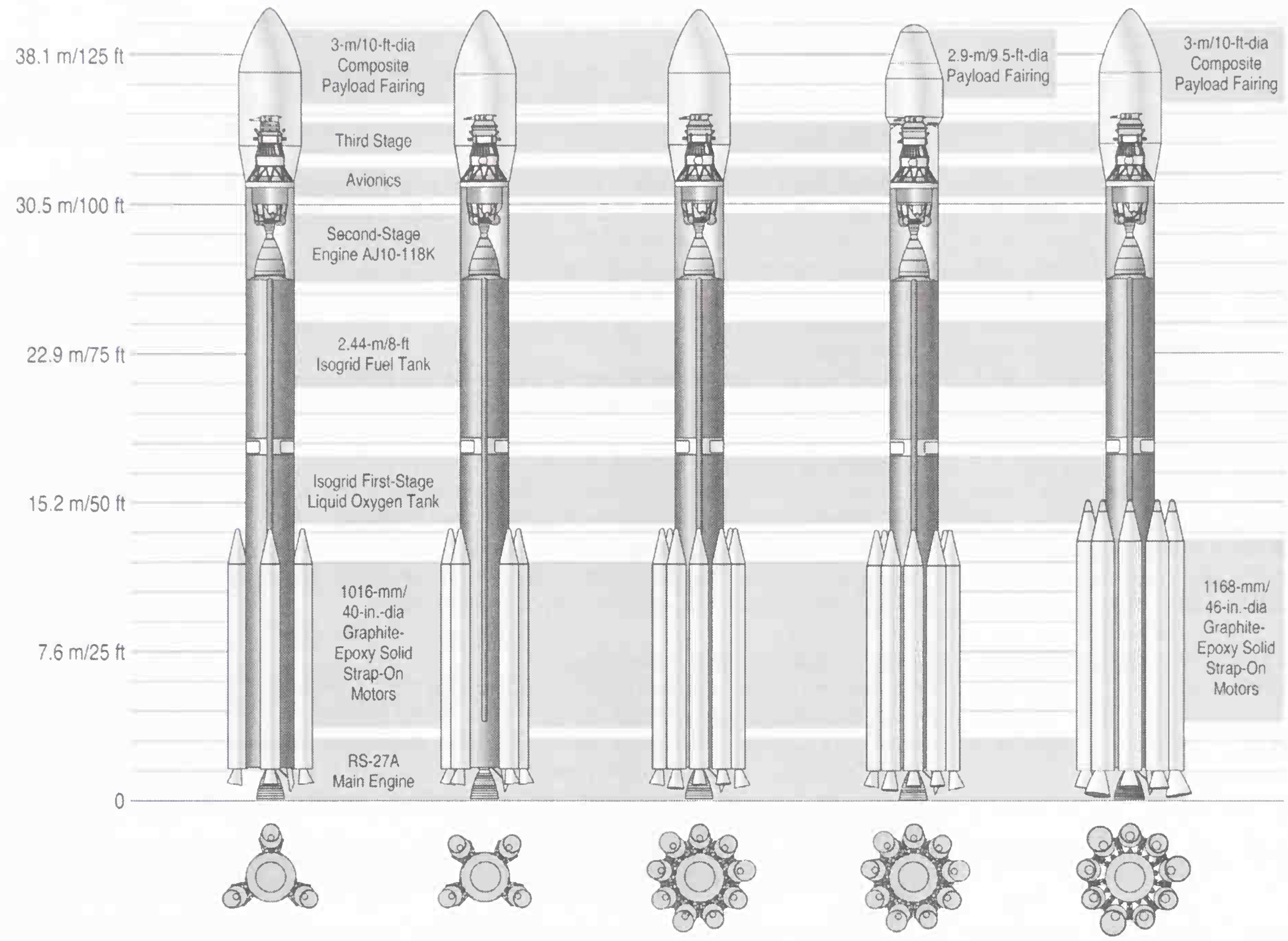
Flight History

Failure Descriptions:

F	1960 May 13	144/D1	1960 F07	Second-stage attitude control malfunction.
S	1963 Feb 14	Syncom 1	1963 004A	Spacecraft communications system failed when satellite reached GEO.
F	1964 Mar 19	391/D24	1964 F02	Insufficient third-stage thrust.
F	1965 Aug 25	434/D33	1965 F09	Third stage ignited before separation because the ignition signal bypassed a delay train. Stage was not stabilized, resulting in failure to reach orbit.
P	1966 Jul 01	467/D39	1966 058A	Excessive second-stage velocity prevented spacecraft from reaching lunar orbit. Experiments were carried out in elliptical orbit instead.
S	1966 Oct 26	Intelsat 201	1966 096A	Failed to achieve geosynchronous orbit because of satellite apogee motor malfunction.
F	1968 Sep 19	529/D59	1968 F08	First-stage rate gyro malfunctioned, causing pitch oscillation beginning at T+20 s, loss of control at T+60 s, and destruction by range safety officer at T+108 s.
F	1969 Jul 26	547/D71	1969 064A	Third-stage malfunction placed satellite in incorrect orbit.
F	1969 Aug 27	540/D73	1969 F12	First-stage hydraulics failure forced destruction at T+483 s.
S	1970 Jul 23	Intelsat 308	1970 055A	Apogee motor failure.
S	1970 Aug 19	Skynet 1B	1970 062A	Apogee motor failure.
F	1971 Oct 21	572/D86	1971 091A	Second-stage nitrogen gas ACS thruster malfunctioned, causing loss of stability.
F	1973 Jul 16	578/D96	1973 F06	Failure of the second-stage hydraulic pump motor caused loss of attitude control.
F	1974 Jan 19	587/D100	1974 002A	Failure of second stage attitude control electronics caused spacecraft to be delivered to low orbit and subsequently reenter the atmosphere.
P	1977 Apr 20	617/D130	1977 029A	Spacecraft was delivered to incorrect orbit due to a malfunction during third-stage spin-up. Orbit was partly corrected, allowing some mission goals to be completed.
F	1977 Sep 13	619/D134	1977 F04	A burn through on one of the strap-on boosters caused the first-stage to explode at T+55 s. Investigation revealed that SRM case properties did not meet requirements.
S	1979 Dec 07	Satcom 3 (RCA C)	1979 101A	Satellite ceased functioning after firing apogee kick motor.
S	1982 Apr 10	Insat 1A	1982 031A	Spacecraft failed to become operational.
F	1986 May 03	D178	1986 F04	First-stage engine suffered a premature shutdown at T+71 s because of an electrical short. Destroyed by range safety at T+91 s.
P	1995 Aug 05	D228	1995 041A	One of the air-ignited SRM GEMs failed to separate, because of overheated explosive lines in the separation system. The extra mass caused the launch vehicle to deliver the payload to a lower than planned orbit. The spacecraft achieved GSO using onboard thrusters and station-keeping propellant, causing its planned lifetime to be cut in half.
F	1997 Jan 17	D241	1997 F01	At T+12 s the No. 2 GEM strap-on motor suffered a structural failure of its composite case (not a motor burn through), resulting in a long vertical crack along the side of the motor. The vehicle self destructed.
S	1997 Jul 09	Iridium 21	1997 034E	One of the five spacecraft failed to become operational. The root cause of the spacecraft failure has never been identified conclusively, however, the launch parameters were nominal. Originally falling fairing blanket material was cited as a cause, this was ruled out.
S	1997 Aug 21	Iridium 024	1997 043C	Spacecraft failed in orbit.
F	1998 Aug 27	D3-1	1998 F03	At T+55 the rocket began a normal 4 Hz roll oscillation. Because the control software design had not accounted for the oscillation, the vehicle used up all the hydraulic fluid in the strap-on booster nozzle TVC system attempting to correct the roll. Once the hydraulic fluid was exhausted, attitude control was lost, the vehicle pitched over at T+72 s and began to break up because of aerodynamic forces, causing autodestruct.
S	1998 Sep 08	Iridium 079	1998 051D	Spacecraft failed shortly after launch.
F	1999 May 05	D3-2	1999 024A	Second-stage RL10 engine shut down immediately after start of second burn because of structural failure of the combustion chamber, stranding spacecraft in low orbit. Structural failure occurred because of poor brazing process in combustion chamber fabrication.
S	2003 Jul 03	Contour	2002 034A	Spacecraft fired its integral solid motor to leave Earth orbit and broke up toward the end of the burn. The aft end of the spacecraft was built too close to the nozzle exit plane, allowing it to overheat.

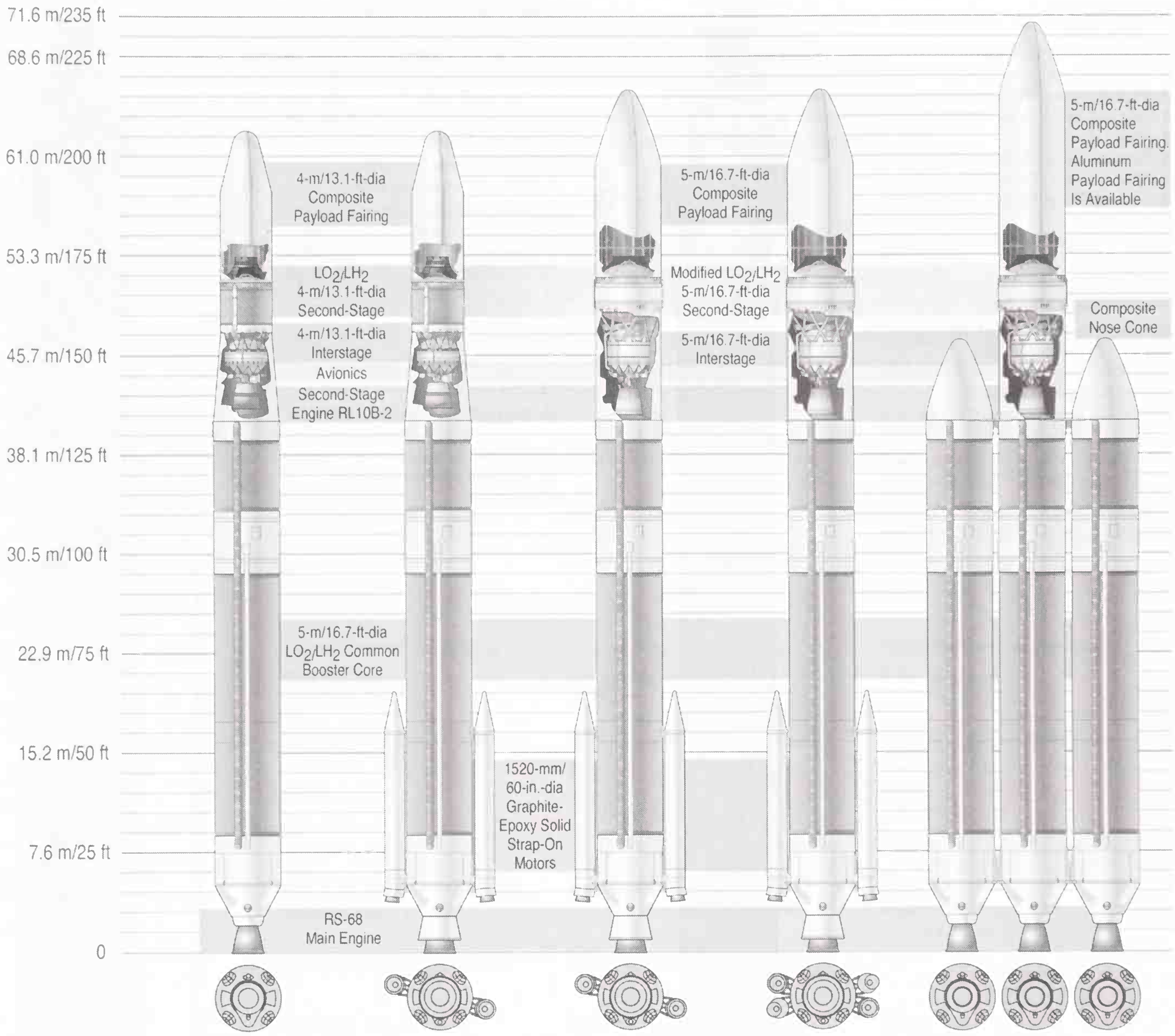
Vehicle Design

Overall Vehicle



	Delta II 7326-10	Delta II 7425-10	Delta II 7925-10	Delta II 7925-9.5	Delta II 7925H-10
Height	38.2–38.9 m (125.2–126.5 ft)	38.2–38.9 m (125.2–126.5 ft)	38.2–38.9 m (125.2–126.5 ft)	38.2–38.9 m (125.2–126.5 ft)	38.2–38.9 m (125.2–126.5 ft)
Gross Liftoff Mass	164 t (361.1 klbf)	165.7 t (364.6 klbf)	233.3 t (513.2 klbf)	233.0 t (512.6 klbf)	286.1 t (629.5 klbf)
Thrust at Liftoff	2157 kN (484.9 klbf)	2140 kN (488.8 klbf)	2844 kN (639.3 klbf)	2844 kN (639.3 klbf)	3900 kN (876.8 klbf)

Vehicle Design



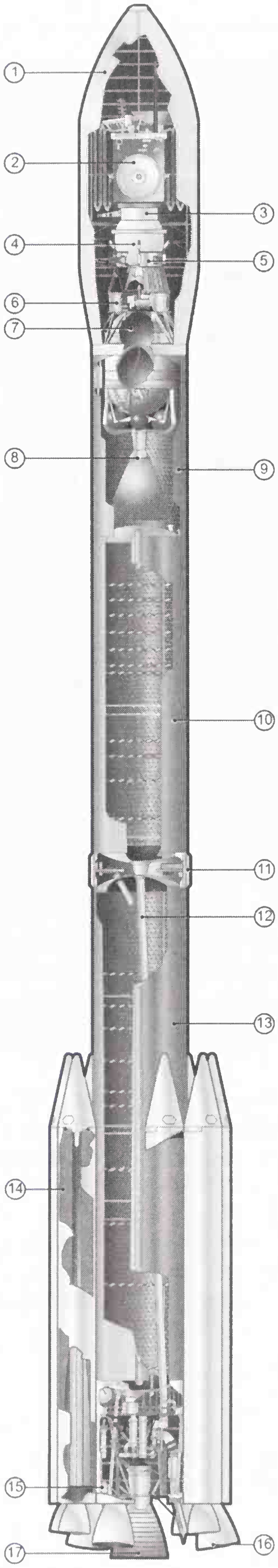
	Delta IV Medium	Delta IV Medium+ (4,2)	Delta IV Medium+ (5,2)	Delta IV Medium+ (5,4)	Delta IV Heavy
Height	~63.0 m (206.7 ft)	~63.0–66.2 m (206.7–217.2 ft)	~63.0–66.2 m (206.7–217.2 ft)	~63.0–66.2 m (206.7–217.2 ft)	~70.7 m (232.0 ft)
Gross Liftoff Mass	1257 t (571.4 klbf)	1616.5 t (734.8 klbf)	1640.8 t (745.8 klbf)	1980 t (900.2 klbf)	3585.3 t (1629 klbf)
Thrust at Liftoff	2879 kN 657,650 (klbf)	4859 kN (1109 klbf)*	4859 kN (1109 klbf)*	6817 kN (1557 klbf)*	8638 kN 1973 klbf)

** thrust measured 1.0 second after GEM60 ignition*

Vehicle Design

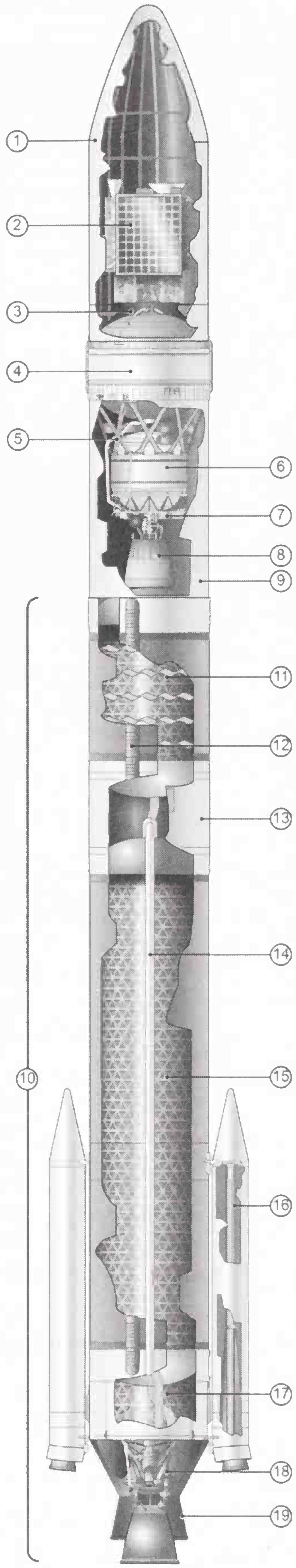
Delta II 7925-10

1. Fairing 3-m (10-ft) diam
2. Spacecraft (typical)
3. Payload Attach Fitting (PAF)*
4. Star 48B Solid Rocket Motor*
5. Spin Table*
6. Second-Stage Guidance System
7. Second-Stage Fuel Tank
8. Second-Stage Engine (AJ10-118K)
9. Interstage
10. First-Stage Fuel Tank
11. Centerbody
12. Fuel Transfer Tube
13. First-Stage LO_2 Tank
14. Graphite-Epoxy Solid Strap-on Motor (GEM 40)
15. Engine Section
16. Extended Air-Lit Nozzle
17. First-Stage Engine (RS-27A)



Delta IV Medium+ (5,2)/(5,4)

1. Composite Fairing (5-m/16.7-ft diam by 14.34-m/27-ft length
2. Spacecraft (typical)
3. Payload Attach Fitting/Dispenser
4. Second-Stage LH_2 Tank
5. Second-Stage Intertank
6. Second-Stage LO_2 Tank
7. Second-Stage Equipment Shelf
8. Second-Stage Engine, RL 10B-2
9. Interstage
10. Common Booster Core
11. LO_2 Tank
12. Utility Tunnel
13. Centerbody
14. LO_2 Feedline
15. LH_2 Tank
16. Graphite-Epoxy Motor, GEM 60 (strap-on solid rocket motor)
17. Engine Section
18. First-Stage Engine, RS-68
19. Aeroskirt



Vehicle Design

Solid Rocket Boosters

The solid rocket motor (SRM) used on Delta is the graphite–epoxy motor (GEM) built by Alliant Techsystems. This motor replaces the smaller, steel-cased Castor family of motors used on Deltas since the mid 1960s. Three different sizes of GEMs are available, designated by their diameter in inches. The Delta II uses the original GEM 40. In the standard Delta II configuration with nine SSRMs, six motors are ignited at liftoff, and three are ignited after the first set burns out. In 1995, extended nozzles were added to the air-lit motors for improved performance. Some Delta II configurations use three or four motors rather than the normal nine. The Delta II Heavy uses nine GEM 46 motors. The basic Delta IV M vehicle does not use any strap-on motors. However, the larger Delta IV M+ vehicles use either 2 or 4 GEM 60 motors for increased performance.

	Delta II GEM 40	Delta Heavy II GEM 46	Delta IV GEM 60
Dimensions			
Length	13.0 m (42.5 ft)	14.7 m (48.1 ft)	16.2 m (53.0 ft)
Diameter	1.0 m (3.3 ft)	1.17 m (3.8 ft)	1.52 m (5.0 ft)
Mass			
Propellant Mass (each)	11,765 kg (25,940 lbm)	17,010 kg (37,500 lbm)	29,904 kg (65,926 lbm)
Inert Mass (each)	1315 kg (2900 lbm)	2035 kg (4490 lbm)	Fixed nozzle: 3250 kg (7165 lbm) With TVC: 3849 kg (8485 lbm)
Gross Mass (each)	13,080 kg (28,840 lbm)	19,040 kg (41,990 lbm)	Fixed nozzle: 33,199 kg (73,191 lbm) With TVC: 33,798 kg (74,511 lbm)
Propellant Mass Fraction	0.90	0.89	Fixed nozzle: 0.90 With TVC: 0.89
Structure			
Type	Filament-wound monocoque	Filament-wound monocoque	Filament-wound monocoque
Material	Graphite–epoxy	Graphite–epoxy	Graphite–epoxy
Propulsion			
Motor Designation	GEM 40 (Alliant Tech Systems)	GEM 46 (Alliant Tech Systems)	GEM 60 (Alliant Tech Systems)
Number of Motors	3, 4, or 9	9	2 or 4
Propellant	HTPB	HTPB	HTPB
Number of Segments	1	1	1
Average Thrust (each)			
Ground-lit	Sea level: 446.0 kN (100.3 klbf) Vacuum: 499.2 kN (112.2 klbf)	Sea level: 537.7 kN (120.8 klbf) Vacuum: 608.1 kN (136.7 klbf)	Vacuum: Fixed nozzle: 851.5 kN (191,416 lbf) TVC: 826.9 kN (185,886 lbf)
Air-lit	Vacuum: 516.2 kN (116.1 klbf)	Vacuum: 628.5 kN (141.3 klbf)	—
Isp			
Ground-lit	Sea level: 245.4 s Vacuum: 274.0 s	Sea level: 242 s Vacuum: 224 s	Vacuum: Fixed nozzle: 275.2 s TVC: 274.6 s
Air-lit	Vacuum: 283.4 s	Vacuum: 284 s	—
Chamber Pressure	56.3 bar (817 psi)	104.5 bar (1516 psi) peak	90.5 bar (1312 psi) peak
Nozzle Expansion Ratio	Ground-lit: 10.65:1 Air-lit: 16:1	Ground-lit: 14:1 Air-lit: 24.9:1	Ground-lit: 11.0:1
Attitude Control			
Pitch, Yaw, Roll	Stage 1 provides control. SRMs have fixed 10-deg nozzle cant.	Stage 1 provide control. SRMs have fixed 10-deg nozzle cant	The first two motors have hydraulic gimbaled nozzles with flex seals, the second two motors have a fixed 10-deg nozzle cant.
Staging			
Nominal Burn Time	63.3 s	75.2 s	Fixed nozzle: 89.7 s TVC: 90.9 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion
Stage Separation	Ordnance thruster	Ordnance thrusters	Ordnance thrusters

Vehicle Design

Stage 1

The first stage of the Delta II incorporates an engine section that contains the RS-27A main engine and two LR101-NA-11 vernier engines. The RS-27A engine burns LOX/RP-1 (kerosene) and generates approximately 890 kN (200 klbf) of thrust at sea level. The vernier engines use the same propellants, and provide roll control during first-stage flight and full attitude control between first-stage shutdown and second-stage separation. Propellants are stored in two cylindrical tanks with Delta's distinctive isogrid structure. This is produced by milling out a triangular grid pattern on the inside of the tank structure, which provides sufficient strength with less mass than a full-gauge monocoque structure. Between the two tanks is a center body section that houses first-stage avionics, including the telemetry system and ordnance sequencing equipment. The first stage also has an aft gyro for improved stability margins. At the base of the stage are attach hardware and separation systems for up to nine solid rocket motors.

The Delta IV first stage is called the Common Booster Core (CBC). The CBC is a large hydrogen-fueled stage. Its major subassemblies are a forward skirt, which attaches to the upper stage or payload fairing; an isogrid LOX tank in the forward position, an intertank which contains the first stage avionics, an isogrid hydrogen tank, and an engine compartment, which houses the RS-68 main engine. With an added nose cone and separation system, a CBC can serve as a strap-on booster for the Delta IV Heavy configuration. The aluminum isogrid tank barrel panels are welded together using friction stir welding techniques, which result in stronger bonds and more efficient production. Tank domes are produced in a single piece using a spun-formed process.

Hydrogen was selected as the first-stage fuel because it can provide high specific impulse even in an engine with low chamber pressure. However, hydrogen is a very low-density fuel, which means the fuel tank must be very large. While many launch systems, such as Ariane 5 and the H-II, have used hydrogen-fueled core stages with strap-on boosters, the Delta IV Medium is the first launch system to lift off using only hydrogen-fueled propulsion.

The CBC is powered by the Rocketdyne RS-68 engine. The RS-68 engine burn LOX/LH_2 and is the first large liquid fueled engine to be developed and certified in the United States since the Space Shuttle Main Engine was developed in the late 1970s. The design traces its technical heritage to the Space Transportation Main Engine program of the late 1980s, but unlike past engines the RS-68 was designed with a lower cost and higher reliability as the primary objectives. The need for low cost led to a low chamber pressure engine using a gas-generator cycle, a design approach that is relatively simple and low risk. Because much of the cost of past engine development programs was attributed to test failures that resulted in redesigning and retesting the engine, the RS-68 program emphasized low risk designs and component and subsystem verification to discover problems before ever firing the engine. Several techniques were used to reduce the time required to design and manufacture parts, including modern 3D computer design tools, stereolithography, and advanced casting techniques that reduce the number of separate parts that need to be welded together. The selection of an ablative nozzle eliminated complex cooling channels and decoupled the nozzle design from the combustion chamber design. The result is an engine that produces more thrust than the SSME, but has 80% fewer parts, requires only 8% as much labor to produce, and costs one-fourteenth as much per engine.

Vehicle Design

Stage 1 (cont.)

	Delta II	Delta IV
Dimensions		
Length	26.1 m (85.6 ft)	40.8 m (133.9 ft)
Diameter	2.44 m (8 ft)	5.1 m (16.7 ft)
Mass		
Propellant Mass	96.1 t (211.9 klbm)	199.6 t (440,000 lbm)
Inert Mass (w/interstage)	5680 kg (12,520 lbm)	26,760 kg (59,000 lbm)
Gross Mass	101.8 t (224.4 klbm)	226.4 t (499,000 lbm)
Propellant Mass Fraction	0.94	0.88
Structure		
Type	Tanks: isogrid Center body: skin-stringer	Tanks: isogrid Center body, interstage: skin-stringer
Material	Aluminum	Tanks: aluminum Interstage: graphite–epoxy Center body: graphite–epoxy
Propulsion		
Engine Designation	RS-27A (Rocketdyne)	RS-68 (Rocketdyne)
Number of Engines	1	1
Propellant	LOX/RP-1 (kerosene)	LOX/LH_2
Average Thrust	Sea level: 890 kN (200 klbf) Vacuum: 1085.8 kN (244.1 klbf)	Sea level: 2918 kN (656 klbf) Vacuum: 3341 kN (751 klbf)
Isp	Sea level: 254.2 s Vacuum: 301.7 s	Sea level: 357 s Vacuum: 409 s
Chamber Pressure	48.4 bar (700 psi)	97.2 bar (1410 psi)
Nozzle Expansion Ratio	12:1	21.5:1
Propellant Feed System	Gas generator turbopump	Gas generator turbopump
Mixture Ratio (O/F)	2.24:1	6.0:1
Throttling Capability	100% only	57% and 102%
Restart Capability	None	None
Tank Pressurization	High-pressure nitrogen gas	High-pressure helium gas
Attitude Control		
Pitch, Yaw	Hydraulic nozzle gimbal	Hydraulic nozzle gimbal
Roll	2 Rocketdyne LR101-NA-11 vernier engines	Vectoring of turbine exhaust gas
Staging		
Nominal Burn Time	261 s	249 s
Shutdown Process	Propellant depletion	Propellant depletion
Stage Separation	6 guided springs	Guided springs

Vehicle Design

Stage 2

The Delta II second stage, also known as the second-stage propulsion system (SSPS), is powered by the restartable Aerojet AJ10-118K engine developed by the U.S. Air Force. The AJ10-118K runs on N_2O_4 and Aerozine 50, a 50/50 mix of hydrazine and UDMH. A cold-gas nitrogen system provides attitude control during coast periods, and roll control during the main engine burn. A "miniskirt" and truss structure provide the interface between the second stage and the larger diameter shroud and interstage. The forward section of the second stage contains the vehicle guidance and control avionics.

Delta IV uses two similar second-stage designs. The stages are hydrogen-fueled and powered by an RL10B-2 engine. The RL10B-2 is similar to the RL10A-4 flown on the Atlas IIAS Centaur with the addition of a large carbon–carbon extendible nozzle cone produced by Snecma of France. The propellant tanks are separate, self-supporting structures for simplified production and operations. Second-stage avionics are mounted on an equipment shelf below the LOX tank. DIV-M uses a 4-m (13.1-ft) diameter version of the second stage. The DIV-H stage has a wider 5.1-m (16.7-ft) hydrogen tank, and lengthened LOX tank for a larger propellant capacity. Both stages can be fitted with a kit to improve thermal and pressurization systems allowing longer missions and an additional engine restart. This enables the upper stage to deliver payloads directly to into GEO. The DIV-M+ vehicles can use either upper stage depending on the configuration.

	Delta II	Delta IV-M	Delta IV-H
Dimensions			
Length	6 m (19.6 ft)	12.0 m (39.5 ft)	12.0 m (39.5 ft)
Diameter	2.4 m (8 ft)	4 m (13.1 ft)	5.1 m (16.7 ft)
Mass			
Propellant Mass	6004 kg (13,236 lbm)	20,400 kg (45,000 lbm)	27,200 kg (60,000 lbm)
Inert Mass	950 kg (2095 lbm)	2850 kg (6300 lbm)	3490 kg (7700 lbm)
Gross Mass	6954 kg (15,331 lbm)	24,170 kg (53,300 lbm)	30,710 kg (67,700 lbm)
Propellant Mass Fraction	0.86	0.84	0.89
Structure			
Type	Monocoque	Isogrid LH_2 tanks, monocoque LOX tanks	Isogrid LH_2 tanks, monocoque LOX tanks
Material	Tanks: stainless steel Structures: aluminum	Aluminum	Aluminum
Propulsion			
Engine Designation	AJ10-118K (Aerojet)	RL10B-2 (Pratt & Whitney)	RL10B-2 (Pratt & Whitney)
Number of Engines	1	1	1
Propellant	N_2O_4/Aerozine 50	LOX/LH_2	LOX/LH_2
Average Thrust	43,657 N (9815 lbf)	110 kN (24,750 lbf)	110 kN (24,750 lbf)
Isp	319.2 s	462.4 s	462.4 s
Chamber Pressure	57 bar (827 psi)	32.1 bar (465 psi)	32.1 bar (465 psi)
Nozzle Expansion Ratio	65:1	285:1	285:1
Propellant Feed System	Pressure-fed	Split-expander turbopump	Split-expander turbopump
Mixture Ratio (O/F)	1.8:1	5.5:1	5.5:1
Throttling Capability	100% only	100% only	100% only
Restart Capability	Multiple: 6 demonstrated	Nominally 1 restart or 2 with kit.	Nominally 1 restart or 2 with kit.
Tank Pressurization	Gaseous helium	Helium for LOX tank, warm hydrogen for LH_2 tank	Helium for LOX tank, warm hydrogen for LH_2 tank
Attitude Control			
Pitch, Yaw	Hydraulic gimbaling	Electromechanical gimbaling	Electromechanical gimbaling
Roll	Cold gas nitrogen	Hydrazine ACS	Hydrazine ACS
Staging			
Nominal Burn Time	431 s	850 s	1125 s
Shutdown Process	Guidance shutdown	Guidance shutdown or burn to depletion	Guidance shutdown or burn to depletion
Stage Separation	—	—	—

Vehicle Design

Stage 3

The optional Delta II third stage is powered by a Thiokol Star 48B solid motor. The motor is supported by a spin table that remains attached to the second-stage guidance section. Before separation, the motor and payload are spun up to the desired spin rate using small solid spin rockets, which are mounted on a spin bearing. Once the stage is spinning, an ordnance sequencing system separates the stage and ignites the motor. The payload attach fitting (PAF) on top of the motor supports the payload and contains the sequencer, an S-band telemetry transmitter, and batteries. It also includes a nutation control system (NCS) that uses a single hydrazine thruster to prevent wobbling. If necessary, the payload spin can be stopped before separation using a despin system in which weighted cables are deployed radially to reduce the spin rate. For the 7326 vehicle a smaller third stage is used, which is based on a Thiokol Star 37FM solid motor. The Star 37FM motor is similar to the Star 48B, but has only 1065 kg (2350 lbm) of propellant.

	Delta II 7925, 7425, 7325 Star 48	Delta II 7326 Star 37FM
Dimensions		
Length	2.0 m (6.7 ft)	1.7m (5.5ft)
Diameter	1.2 m (4.1 ft)	0.9m (3.1ft)
Mass		
Propellant Mass	2009 kg (4430 lbm)	1065 kg (2350 lbm)
Inert Mass	208 kg (456 lbm)	82kg (180 lbm)
Gross Mass	2217 kg (4887 lbm)	1147kg (2530 lbm)
Propellant Mass Fraction	0.91	0.93
Structure		
Type	Motor case: monocoque PAF: monocoque	Motor case: monocoque PAF: monocoque
Material	Motor case: titanium PAF and spin table: aluminum	Motor case: titanium PAF and spin table: aluminum
Propulsion		
Motor Designation	Star 48B (Thiokol)	Star 37FM (Thiokol)
Number of Motors	1	1
Propellant	HTPB	HTPB
Number of Segments	1	1
Average Thrust	66.4 kN (14.9K lbf)	45.8 kN (10.3 Klbf)
Isp	292.2 s	291.8 s
Chamber Pressure	39.7 bar (575 psi)	36.5 bar (529 psi)
Nozzle Expansion Ratio	54.8:1	43.7:1
Attitude Control		
Pitch, Yaw, Roll	Spin-stabilized, with hydrazine NCS	Spin-stabilized, with hydrazine NCS
Staging		
Nominal Burn Time	87.1 s	66.4 s
Shutdown Process	Burn to depletion	Burn to depletion
Stage Separation	Spring ejection	Spring ejection

Attitude Control System

Delta II first-stage attitude control is provided by gimbaling the RS-27A engine nozzle and using the verniers for roll control. The second-stage attitude control features include a nozzle gimbal on the main engine, as well as a cold-gas nitrogen ACS. The cold-gas system provides roll control during the main-engine burns. It also provides full attitude control during coast phases and enough forward acceleration for propellant settling before second-stage restart. The ACS can provide payload pointing before separation and then execute a collision avoidance maneuver after separation. The nitrogen is stored in a high-pressure spherical titanium tank at the bottom of the stage. For three stage missions, the third stage is spin stabilized on a spin table before separation to ensure stability. The third stage includes the NCS with a small hydrazine tank and a single thruster to prevent the spinning stage from coning after separation.

The Delta IV first stage controls pitch and yaw by hydraulically gimbaling the RS-86 engine. Roll is controlled by vectoring the turbine exhaust gases. In configurations that use solid rocket motors, two of the motors have nozzle gimbal systems to provide additional control authority. The second stage uses the RL10 engine nozzle gimbal system to control pitch and yaw, with an additional ACS system to control roll and provide attitude control during coast phases.

Vehicle Design

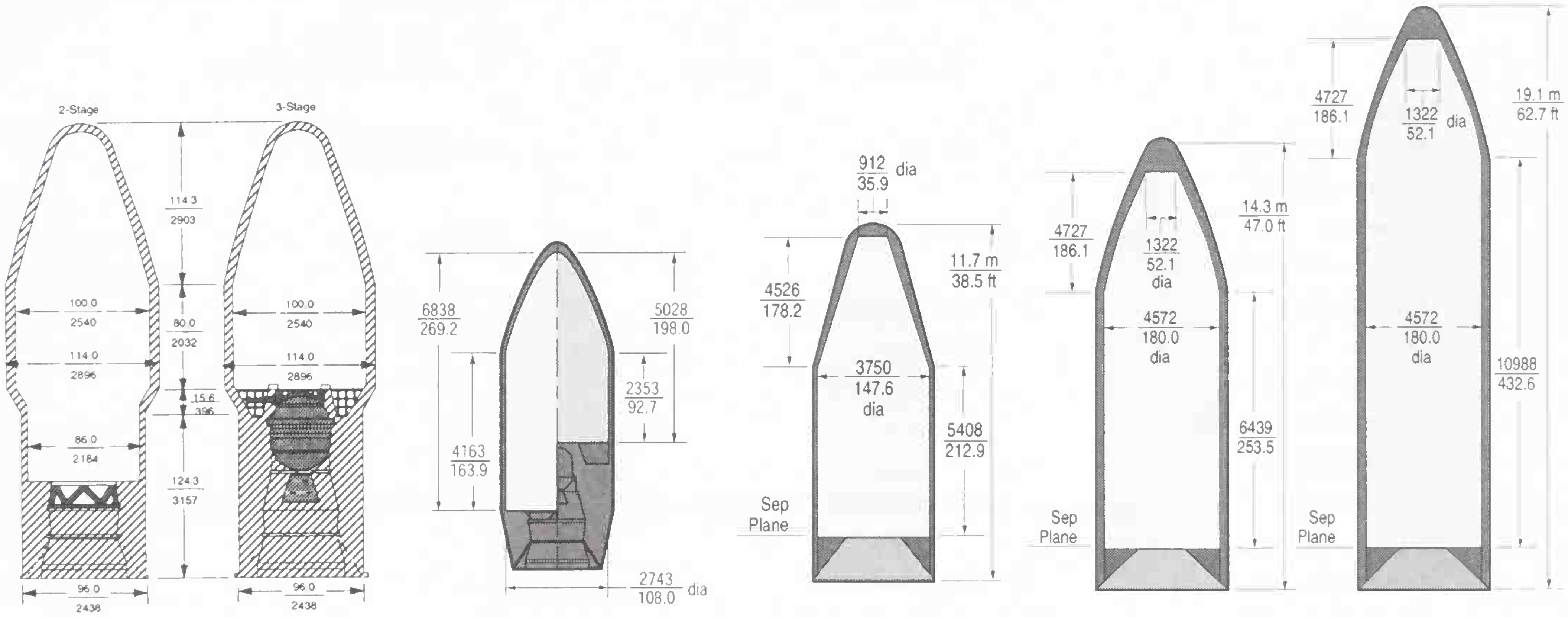
Avionics

The Delta inertial guidance system uses redundant inertial flight control assembly (RIFCA). The RIFCA is a modern inertial guidance system with six AlliedSignal RL20 ring laser gyros and six Sunstrand QA3000 accelerometers to provide redundancy in each axis. The RIFCA control logic uses the rate and acceleration data to determine the control commands for the nozzle control actuators and cold gas thrusters. Other vehicle commands are sequenced using timers. The control logic steers the vehicle into the wind after liftoff to reduce the angle of attack, and thus aerodynamic loads and required control forces. As the dynamic pressure decreases, the guidance system corrects for dispersions to ensure injection into the proper orbit. In addition to the RIFCA, the first- and second-stage avionics include a control box for the ordnance systems, an electronics package called the E-pack that interfaces with the RIFCA to distribute attitude control commands, and a telemetry system to relay vehicle performance data. Each of these systems is powered by the power and control box. Delta IV avionics are similar to Delta II. Unlike earlier vehicles, Delta IV uses a link to the TDRSS satellite system to transmit telemetry. Data rates are also increased from 320 KBPS on a Delta II to 1280 KBPS on a Delta IV M+ or 1920 KBPS on a Delta IV Heavy.

Payload Fairing

The fairings encapsulate the payload to provide a clean, controlled environment and protect the spacecraft from aerodynamic loads and heating during ascent. To prevent spacecraft contamination, the fairings are cleaned before integration, and use noncontaminating separation systems. Options such as acoustic or thermal blankets, access doors, or radio-transparent windows are available on each fairing. Delta fairings are referred to by the diameter. The Delta II uses both a 9.5-ft and a 10-ft diam fairing. The 9.5-ft metal fairing was developed for GPS satellites. It is primarily designed for the three-stage version of Delta II, with a larger diameter at the height of the three-stage configuration payload section, and a smaller diameter cylindrical section that encloses the third stage. The 10-ft composite fairing is a new lightweight replacement for an older, shorter metal fairing of the same diameter. The 10-ft fairing is also available in a 10L version, which has a 0.9-m (3-ft) longer cylinder section for taller payloads.

The Delta IV uses both a 4-m and a 5-m diam fairing. Two types of 5-m-diam payload fairings are available for use on the Delta Heavy. Boeing produces the aluminum isogrid trisector payload fairing for Titan IV, and offers a version of this fairing for military payloads transitioning to Delta IV. All other launches will use a composite bisector payload fairing that is based on the 4-m composite fairing. The 5-m diam composite fairing is available in two lengths. Future growth options include a dual-payload attach fitting (DPAF-5) available in two lengths. Its design is based on the SYLDA-5 payload encapsulation system used on Ariane 5. The fairings and DPAF-5 are designed for offline encapsulation of fueled payloads.



	9.5-ft Diameter (Delta II)	10-ft Diameter (Delta II)	4-m Diameter (Delta IV M and M+(4,2))	5-m Diameter (Delta IV M+(5,2), M+(5,4) and Heavy)
Length	8.49 m (27.8 ft)	8.9 m (29.2 ft)	11.7 m (38.5 ft)	14.3 m (47 ft) 19.1 m (63 ft)
Primary Diameter	2.9 m (9.5 ft)	3 m (10 ft)	4 m (13.1 ft)	5.1 m (16.7 ft)
Mass	841 kg (1850 lbm)	1040 kg (2300 lbm)	1677 kg (3697 lbm)	47 ft: 2935 kg (6470 lbm) 63 ft: 3520 kg (7760 lbm)
Sections	2	2	2	2
Structure	Isogrid base, skin-stringer center cylinder, monocoque cone	Composite sandwich	Composite sandwich	Composite sandwich
Material	Aluminum; nose cone is part fiberglass	Graphite–epoxy composite	Graphite–epoxy composite	Graphite–epoxy composite

Note: Fairings not drawn to scale.

Payload Accommodations

	Delta II	Delta IV
Payload Compartment		
Maximum Payload Diameter	9.5-ft PLF: 2540 mm (100.0 in.) 10-ft PLF: 2743 mm (108 in.)	4-m fairing: 2750 mm (147.6 in.) 5-m fairing Composite: 4572 mm (180 in.) Aluminum isogrid: 4572 mm (180 in.)
Maximum Cylinder Length	9.5-ft PLF: 2004 mm (78.9 in.) for 2 or 3 stage, plus additional 1810 mm (71.3 in.) length at 86.0-in. (2184-mm) diameter for 2 stage 10-ft PLF: 4163 mm (163.9 in.) for 2-stage and 2353 mm (92.7 in.) for 3-stage 10L PLF: 5117 mm (201.4 in.) for 2-stage and 3307 mm (130.2 in.) for 3-stage	4-m fairing: 5408 mm (212.9 in.) 5-m fairing Composite: 6439 mm (253.5 in.) or 10,988 mm (432.6 in.) Aluminum isogrid: 12,192 mm (480.0 in.)
Maximum Cone Length	9.5-ft PLF: 2676 mm (105.4 in.) 10-ft PLF: 2675 mm (105.3 in.)	4-m fairing: 4526 mm (178.2 in.) 5-m fairing Composite: 4292 mm (169.0 in.) or 4727 mm (186.1 in.) Aluminum isogrid: 4292 mm (169.0 in.)
Payload Adapter Interface Diameter	Model 937 PAF: 940 mm (37.0 in.) for 3-stage vehicle Model 6306 PAF: 1604.7 mm (63.2 in.) for 2-stage vehicle Additional options available	Model 6664 PAA: 1666 mm (66 in.) Model 4756 PAA: 1194 mm (47 in.) Model 6556 PAA: 1664 mm (65.5 in.) Model 6343 PAF: 1575 mm (62 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–24 months	T–24 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–4 m	T–5
On-Pad Storage Capability	Months	Extended storage in HIF
Last Access to Payload	T-17 h	Mission specific
Environment		
Maximum Axial Load	7925: 6.0 *g* 7320: 6.65 *g*	D-IV M, M+: 6.5 *g* D-IV H: 6.0 *g*
Maximum Lateral Load	2-stage: ±2.0 *g* 3-stage: ±2.5 *g*	D-IV M, M+: 2.0 *g* D-IV H: 2.5 *g*
Minimum Lateral/Longitudinal Payload Frequency	15 Hz/35 Hz	D-IV M, M+: 10/27 Hz D-IV H: 8/30 Hz
Maximum Acoustic Level	9.5-ft PLF, 7925: 130.0 dB 9.5-ft PLF, 7920: 140.5 dB 10-ft PLF, 7920 and 7925: 132.5 dB	D-IV M: 130 dB at 125–315 Hz D-IV H: 133 dB at 125 Hz
Overall Sound Pressure Level	9.5-ft PLF, 7925: 139.8 dB 9.5-ft PLF, 7920: 146.6 dB 10-ft PLF, 7920 and 7925: 140.6 dB	D-IV M: 140.0 dB D-IV H: 142.7 dB
Maximum Flight Shock	Determined by mission-specific separation system	4000 *g* at 10 kHz
Maximum Dynamic Pressure on Fairing	58.9 kPa (1230 lbf/ft^2)	42.6 kPa (890 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	4.14 kPa/s (0.6 psi/s)	4.14 kPa/s (0.6 psi/s)
Cleanliness Level in Fairing	Class 10,000+	Class 100,000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	LEO: ±9.3 km (5 nmi), ±0.05 deg inclination GTO: ±5.6 km (3 nmi) perigee, ±500–600 km (275-325 nmi) apogee, ±0.2–0.6 deg inclination	LEO: 500 km 90 deg orbit: ±8.6 km (4.6 nmi), ±0.06 deg inclination GTO: ±5 km (2.7 nmi) perigee, ±80 km (43 nmi) apogee, ±0.02 deg inclination
Attitude Accuracy (3 sigma)	±1–2.5 deg	±1–2.5 deg
Nominal Payload Separation Rate	0.3 m/s (1 ft/s) for 2 stage vehicle 0.6–2.4 m/s (2–8 ft/s) for 3 stage vehicle	0.3 m/s (1 ft/s)
Deployment Rotation Rate Available	0–4 rpm for 2-stage vehicle 30–110 rpm for 3-stage vehicle	0–5 rpm
Loiter Duration in Orbit	Mission dependent	Mission dependent
Maneuvers (Thermal/Collision Avoidance)	2-stage: yes 3-stage: no	Yes

Payload Accommodations

	Delta II	Delta IV
Multiple/Auxiliary Payloads		
Multiple Manifest	Delta II has been used for multiple launches of NASA (two spacecraft per launch), Iridium (five spacecraft per launch) and Globalstar (four spacecraft per launch) payloads. Delta does not generally provide dual launches for separate customers.	Future capability will include Delta IV-M and M+ versions that will carry single payloads or multiple spacecraft for a single customer. The Delta IV-H will have a comanifest capability for two spacecraft using DPAF5.
Auxiliary Payloads	Small auxiliary payloads can be attached to the sides of the second-stage avionics section, between the second stage and the payload fairing. The available space is approximately 330-mm (13-in) thick and 780-mm (30.7-in) tall. Several tethered and untethered payloads have made use of this capability.	Delta IV will have standardized auxiliary payload capabilities. The secondary attach module (SAM) is planned to be available in 2006.

Production and Launch Operations

Production

In the past Delta components were manufactured at Boeing's fabrication plant in Huntington Beach, CA, near Los Angeles. The components were then integrated into complete stage assemblies, interstages, and fairings at the final assembly facility in Pueblo, CO. However, during the development of the Delta IV, Boeing decided it would need a new, larger production plant to consolidate Delta IV production in one place. Boeing selected a location outside Decatur, Alabama, and built a 140,000 m^2 (1.5 million ft^2) state-of-the-art facility from the ground up. The factory was completed in December 1999. The facility performs complete production of the launch vehicle, from the arrival of aluminum stock to the shipment of completed stages. At full capacity it can produce 40 CBCs per year. Boeing has consolidated Delta II production at the Decatur factory as well, closing down its Pueblo facility, and reducing production in Huntington Beach.

Delta IV production uses a lean approach in which processes are continuously evaluated to improve efficiency. For example, analysis of the Delta II production process showed that tanks were moved more than 10,000 km (6200 mi) during production to multiple facilities as far away as Canada. The production flow for Delta IV tanks is only 3 km (2 mi) long, contained entirely within the Decatur plant. As a result of process improvements, Delta IV stages can be produced in only 6–7 months, compared to 25–30 months for Delta II stages.

Delta stages are so large that they must be transported by ship (although the 4-m upper stage can be trucked overland if necessary). So, Boeing commissioned a new vessel, the *Delta Mariner*, to transport stages. The *Delta Mariner* is capable of operating in the ocean or in rivers as shallow as 2.5 m (8 ft) and is large enough to carry all of the stages for a Delta IV Heavy in one trip. The trip along the Tennessee Tom Bigbee Waterway to the Gulf of Mexico and then around Florida to CCAFS takes 10 days. A trip to VAFB takes about 25 days.

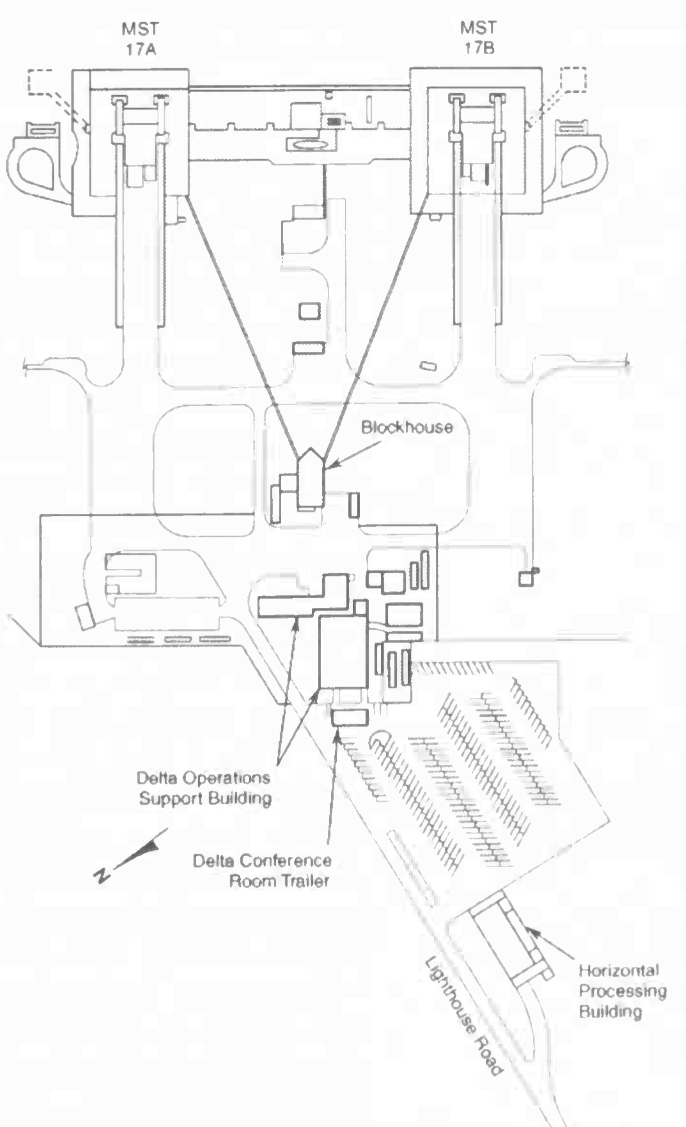
Subcontractors	**Responsibility**	
	Delta II	**Delta IV**
Boeing Expendable Launch Systems	Mission integration and first- and second-stage assembly	Mission integration, and first- and second-stage assembly
ATK Alliant Techsystems	GEM 40 and GEM 46 solid rocket motors and Star third-stage motors	GEM 60 solid rocket motors and composite structures
L3 Communications	RIFCA	RIFCA
Boeing Rocketdyne Propulsion and Power	RS-27A first-stage engine	RS-68 first-stage engine
Gencorp Aerojet	AJ-10-118K second-stage engine	—
Pratt & Whitney	—	RL10B-2 second-stage engine
Alenia Spazio	Second-stage fuel tanks	—
Mitsubishi Heavy Industry	—	Second-stage hydrogen tanks

Launch Operations—Cape Canaveral Air Force Station

Delta II and Delta III

At Cape Canaveral Air Force Station, Delta II and III are launched from Space Launch Complex 17 (SLC-17). The launch complex contains two active pads, 17A and 17B. Only pad B has been modified to accommodate the larger strap-on motors used by Delta II 7925H and the Delta III. The operations that take place on each of the pads include vehicle buildup and checkout, propellant servicing, spacecraft integration, and launch countdown. The two pads can be used for simultaneous buildup of two launch vehicles.

Launch Facilities



Courtesy Boeing.

Cape Canaveral Air Force Station, SLC-17

Delta II integration, checkout, and test activities are accomplished in two phases. The basic philosophy of vehicle integration is to complete checkout of the vehicle using separate test equipment, offline from pad operations, before committing the hardware to the online pad operations. The first phase includes the off-pad operations such as vehicle checkout and SRM processing. The second phase includes the on-pad operations that consist of first- and second-stage stacking, ground support equipment (GSE) validation, upper-stage and payload installation, battery trickle charging, fairing installation, and launch.

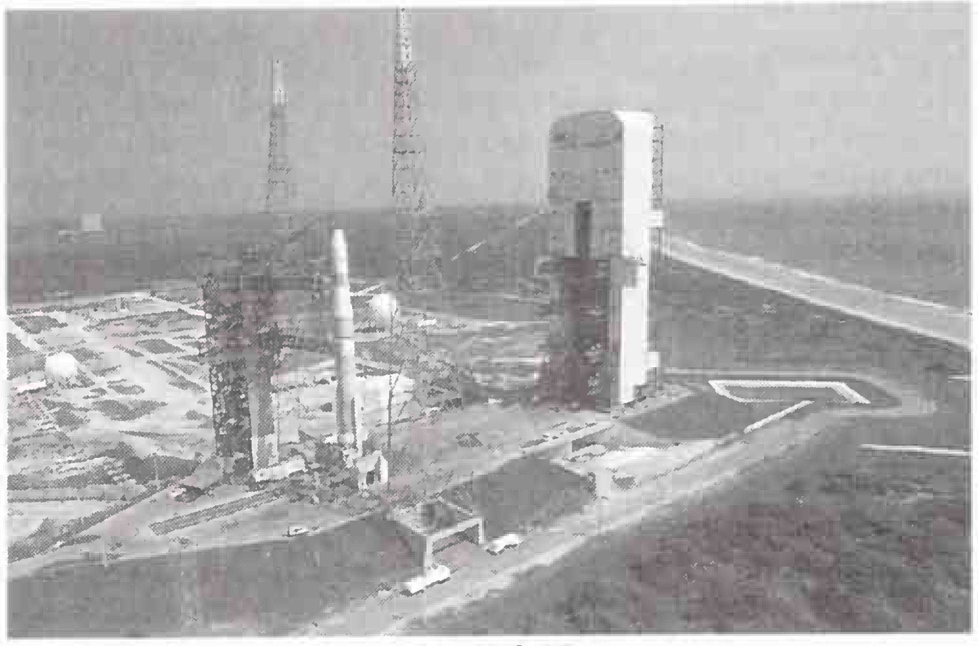
The Delta II first and second stages are received, unpacked, and inspected at Hangar M before Delta mission checkout (DMCO). The payload fairing and interstage are also received and stored until pad integration. DMCO operations on the first and second stages are a series of electrical checks on each booster subsystem before ordnance installation and integration at the pad. The hydraulic, propulsion, telemetry, destruct, and control subsystems are checked. The DMCO facility is located behind Hanger M in the CCAFS Industrial Area. After DMCO, the first and second stages are moved to the horizontal processing facility (HPF) and Area 55 for final pre-erection preparations and destruct charge installation. Besides destruct charge installation, ordnance to separate the SRMs is also installed on the first stage in the HPF. Second-stage processing at Area 55 also includes propulsion leak checks, remote leak evaluation, harness installation, and nozzle extension installation. The GEM SRMs are processed in Area 57. The SRMs are received, checked for leaks, and stored until required. At that time, destruct igniters are installed before pad integration.

Components to build up the upper stage arrive at the Navstar Processing Facility (NPF). The Star 48B solid motor is shipped from ATK Thiokol and stored at the launch site until needed. The PAFs and spacecraft-unique wire harnesses are built at Boeing facilities and shipped to the launch site. Upper-stage ground operations begin with the arrival of the Star-48B solid motor via commercial ground transportation. Following X-ray inspection of the motor, the PAF is mated to the forward end of the motor. Items to be installed on the PAF include the safe and arm mechanism, ETAs, yo-weight, yo-weight cutters,

Production and Launch Operations

test batteries, and wire harnesses. When the upper-stage buildup is complete, the upper stage is moved to the Delta Spin Test Facility (DSTF) where it is aligned. After balancing, the upper stage is moved to NPF for test to verify the deployment sequence circuitry, spin system, and flight sequence.

Pad activities include the erection of the Delta booster and mating of the payload before launch. Final preparations on the Delta first stage before erection are performed in the HPF after which the first stage is moved to SLC-17 for erection. After installing the interstage and SRMs, the second stage is then erected and mated. After pre-erection preparations on the fairing are completed at Hangar M, the fairing halves are erected and stowed on the MST. Once the Delta booster is stacked, system tests, including propellant loading, flight pressurization, hydraulic, pneumatic and electrical control, and guidance checks, are performed in preparation for mating the upper stage and payload.



The first Delta IV on the pad at SLC-37

Spacecraft processing for commercial Delta launches at CCAFS is typically performed at the Astrotech payload processing facilities near Titusville, Florida. Astrotech is a division of Spacehab, and provides spacecraft processing services and facilities for many payloads launched from Cape Canaveral and the Kennedy Space Center. Astrotech facilities include areas for controlled storage, fueling, ordnance installation, spin balancing, and testing of spacecraft. If necessary, additional NASA and USAF facilities, such as the spacecraft assembly and encapsulation facility (SAEF-2), can also be used for spacecraft processing. Delta II payloads are transported to the launch site inside a cylindrical handling canister. Inside the white room in the MST, the payload is removed from the canister, integrated onto the launch vehicle, and enclosed in the payload fairing.

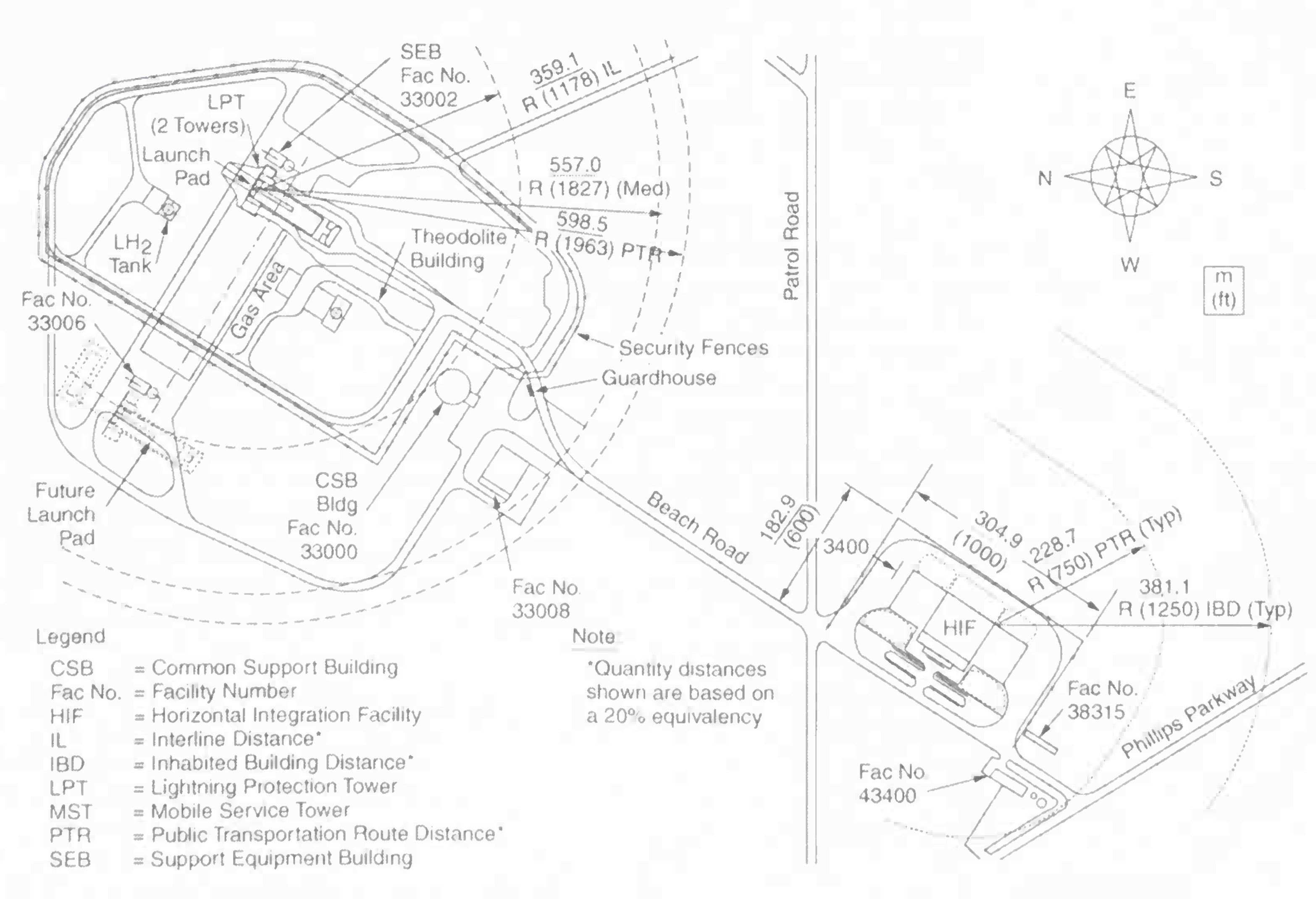
Integrated vehicle system testing starts with a flight sequence test to verify engine sequences, receipt of control, guidance, ordnance firing, and spacecraft discrete commands. Finally, a flight program verification test is performed through the complete mission to verify flight program events on internal power. Preflight finally gets underway as the fairing is installed, the test equipment connections are removed, and ordnance is installed and connected. The second-stage propellants are loaded. The guidance computer, flight beacon, and range safety systems are checked, fairing access doors are closed, and the MST is retracted. During the terminal countdown, the second-stage systems are pressurized, first-stage propellants are loaded, and electrical, hydraulic, and control systems are verified before liftoff.

The SLC-17 blockhouse is no longer used to control Delta launch operations. A 1995 decision to move the operations a safer distance away was validated in January 1997, when a Delta II exploded only 425 m (1400 ft) above the pad. Operations are now conducted from the 1st Space Launch Squadron Operations Building, located at the southern edge of CCAFS. On the day of launch, the mission management team monitors the status of the launch vehicle, payload, and downrange tracking stations in the mission director center in Hanger AE.

Delta IV

Delta IV vehicles are integrated horizontally in the new horizontal integration facility (HIF). The HIF has two bays, each 30.5-m (100-ft) wide and 76.2-m (250-ft) long, with a wide door on each end. The HIF can accommodate four single-core vehicles at a time, or one Delta IV-H vehicle in one bay, and two single-core Delta IV-M or M+ vehicles in the other. Each of the two bays has a 23 t (25 U.S. ton) bridge crane. In the HIF, the stages are tested and checked out, then assembled together. The fully integrated launch vehicle, without the payload and fairing, is transported horizontally to the launch pad. Commercial payloads will be processed and encapsulated in the Astrotech processing facilities and transported directly to the launch pad. U.S. government payloads may be processed in one of six other NASA or USAF facilities, including the Shuttle Payload Integration Facility for military payloads.

Delta IV launches at CCAFS are conducted from SLC-37. SLC-37 was originally built with two pads for launches of the Saturn I and IB in the 1960s and was then mostly dismantled. A new pad designed for all versions of Delta IV has been built at the site of Pad B. The Pad A location is available for future use if needed. Although fatalities that occur during spaceflight receive more attention, it is worth remembering two workers who lost their lives to accidents during construction of the facilities. SLC-37B includes a launch mount, a 73-m (240-ft) tall fixed umbilical tower (FUT), a mobile service tower (MST), two lightning protection towers, a propellant tank farm, and various support buildings. The pad does not have a sound suppression system, which is unusual for a launch vehicle of this size. The FUT has three large swing arms that connect to the first stage, second stage, and payload fairing. Once the launch vehicle is erected on the launch pad, the MST moves into place around it. The MST moves on rails using a hydraulic drive system. Using a 45 t (50 U.S. ton) bridge crane, the encapsulated payload assembly (payload, adapter, and fairing) is lifted into place on the launch vehicle. The MST includes work platforms for late access to the vehicle (particularly the interstage area) as well as the payload.



Courtesy Boeing

Cape Canaveral Air Force Station, SLC-37

Production and Launch Operations

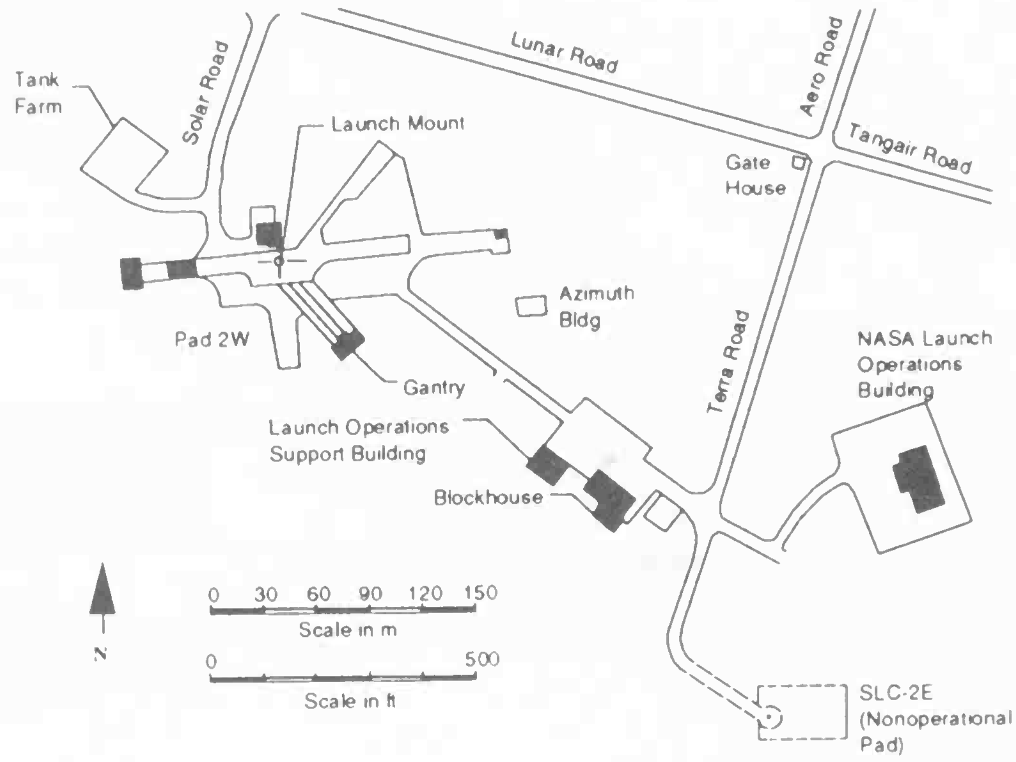
Launch Operations—Vandenberg Air Force Base

Delta II

At VAFB, Delta II vehicles are launched from Space Launch Complex 2 (SLC-2). The launch complex originally contained two pads, but SLC-2E was dismantled in 1972 and therefore only SLC-2W is still used for Delta launches from the West Coast.

Payload processing can be conducted in several facilities at VAFB, including NASA building 836, Astrotech commercial processing facilities in building 1032, or California Commercial Spaceport processing facilities at Building 375. The California Commercial Spaceport facilities are in the Integrated Processing Facility originally built to handle classified Space Shuttle payloads. Commercial payloads typically use Astrotech or California Commercial Spaceport facilities unless special requirements justify use of NASA or USAF facilities.

Vehicle integration and launch operations at VAFB are generally similar to those at Cape Canaveral. The pad consists of a blockhouse, fixed UT, and 54.3 m (178 ft) tall metal MST with a clean room for spacecraft encapsulation. Launch operations are conducted from the remote launch control center in Building 8510 and directed from the mission director center in Building 840.



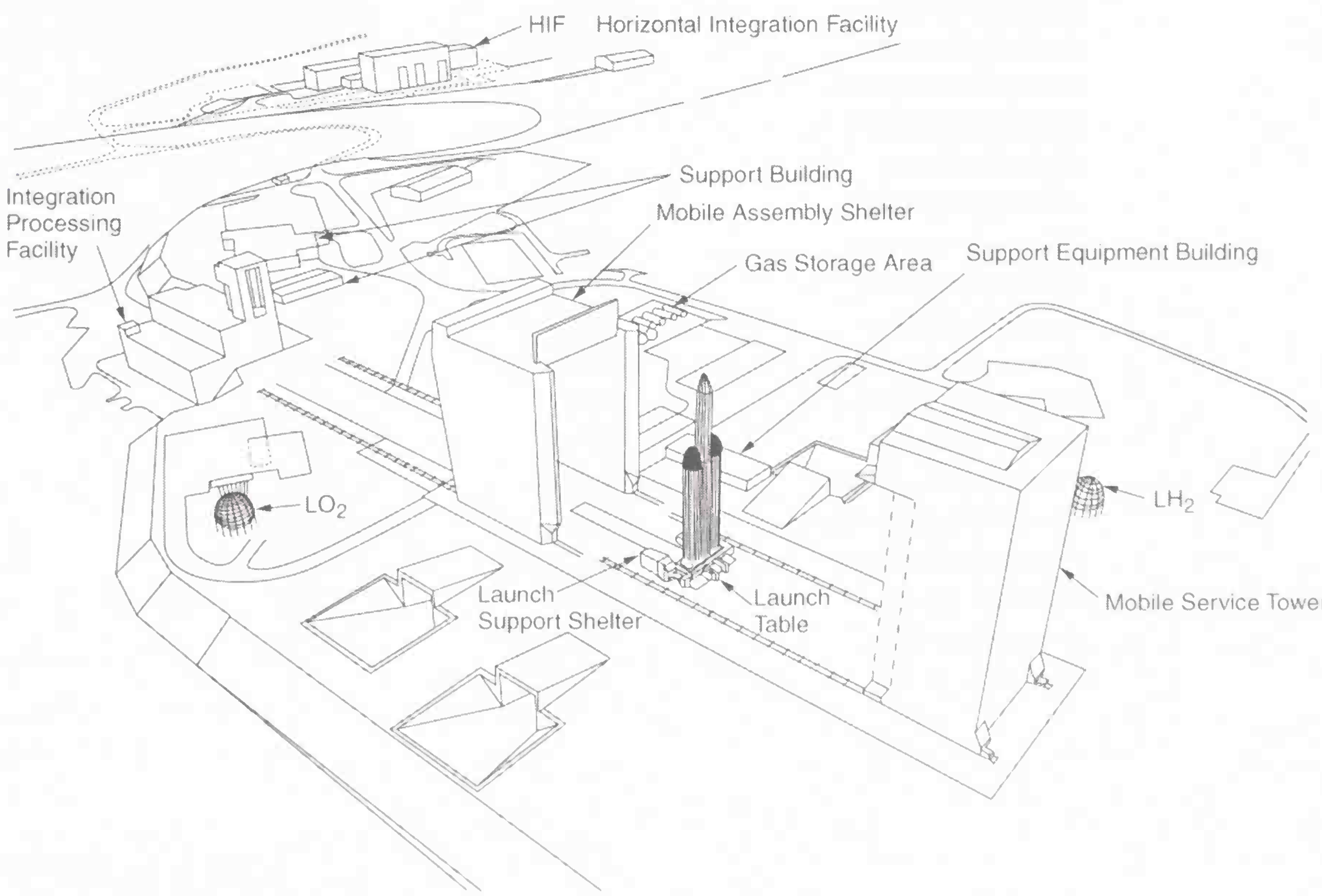
Courtesy Boeing.

Vandenberg Air Force Base, SLC-2

Delta IV

Delta IV launch operations will be conducted from SLC-6. SLC-6 was originally built in the 1960s to launch the planned Titan IIIM launch vehicle in support of USAF manned space missions. These programs were canceled, and SLC-6 was mothballed. SLC-6 was then rebuilt to launch USAF Space Shuttle missions, but again the military Space Shuttle program was canceled before any launches took place. From 1995 to 1999, part of SLC-6 was used as the West Coast launch site for the small, commercial Athena launch vehicle.

An existing Space Shuttle processing building has been modified into a Delta IV Horizontal Integration Facility with capabilities similar to the HIF at Cape Canaveral. The existing MST has been modified with a raised roof to service the vehicle once it is mounted in place on the launch table. As at Cape Canaveral, payloads will be encapsulated offline, and lifted into place on the launch vehicle using the MST crane. Currently only the VAFB Astrotech facilities are capable of encapsulating payloads into the 4-m fairing without modification. The former Space Shuttle Integrated Processing Facility, now run by Spaceport Systems International (SSI) as part of the California Commercial Spaceport, will be capable of processing both 4-m-fairing and 5-m-fairing payloads with some modifications. The SSI building is located immediately adjacent to SLC-6, and has three processing cells and a high bay and airlock for transfer of payloads.

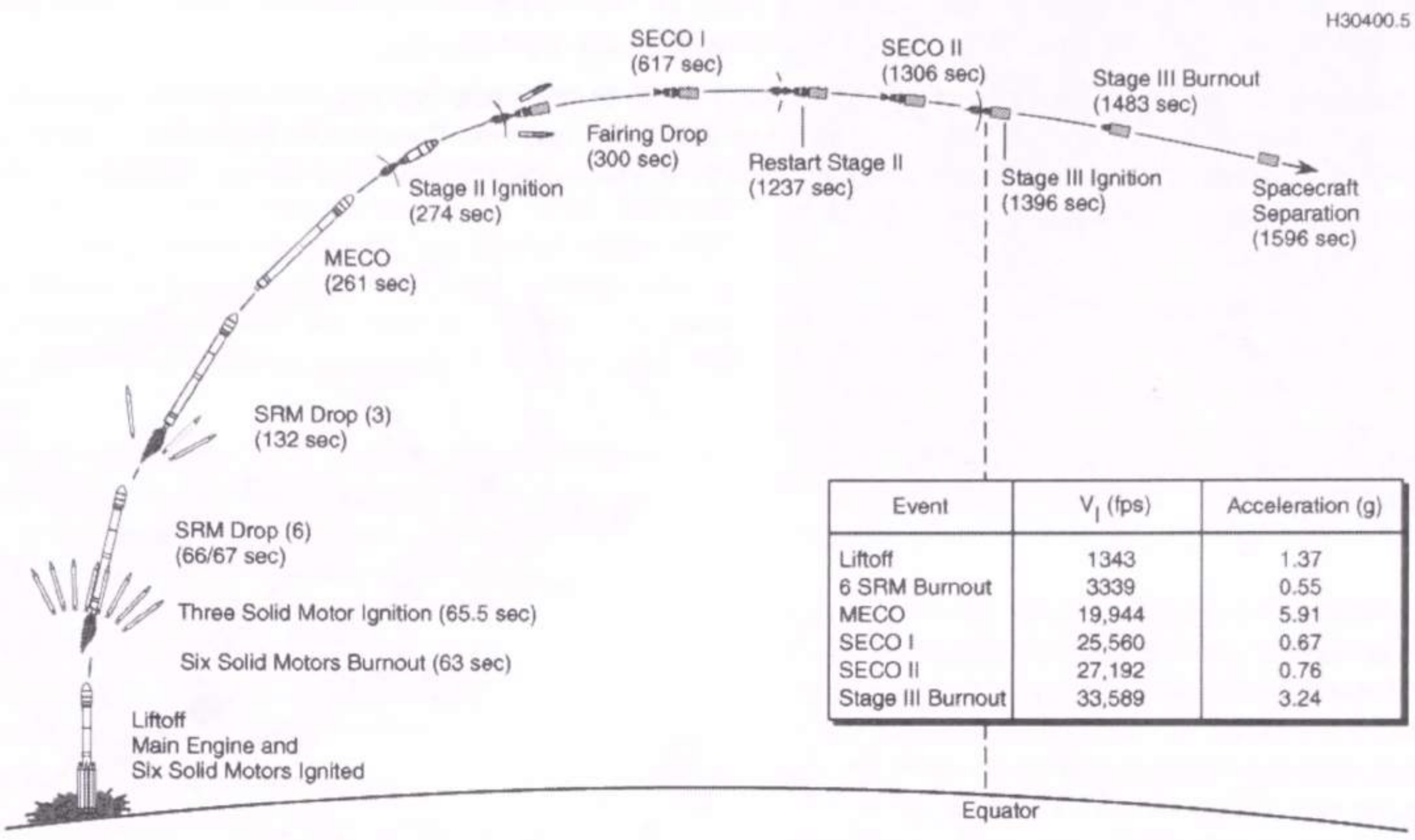


Courtesy Boeing.

Vandenberg Air Force Base, SLC-6

Production and Launch Operations

Flight Sequence



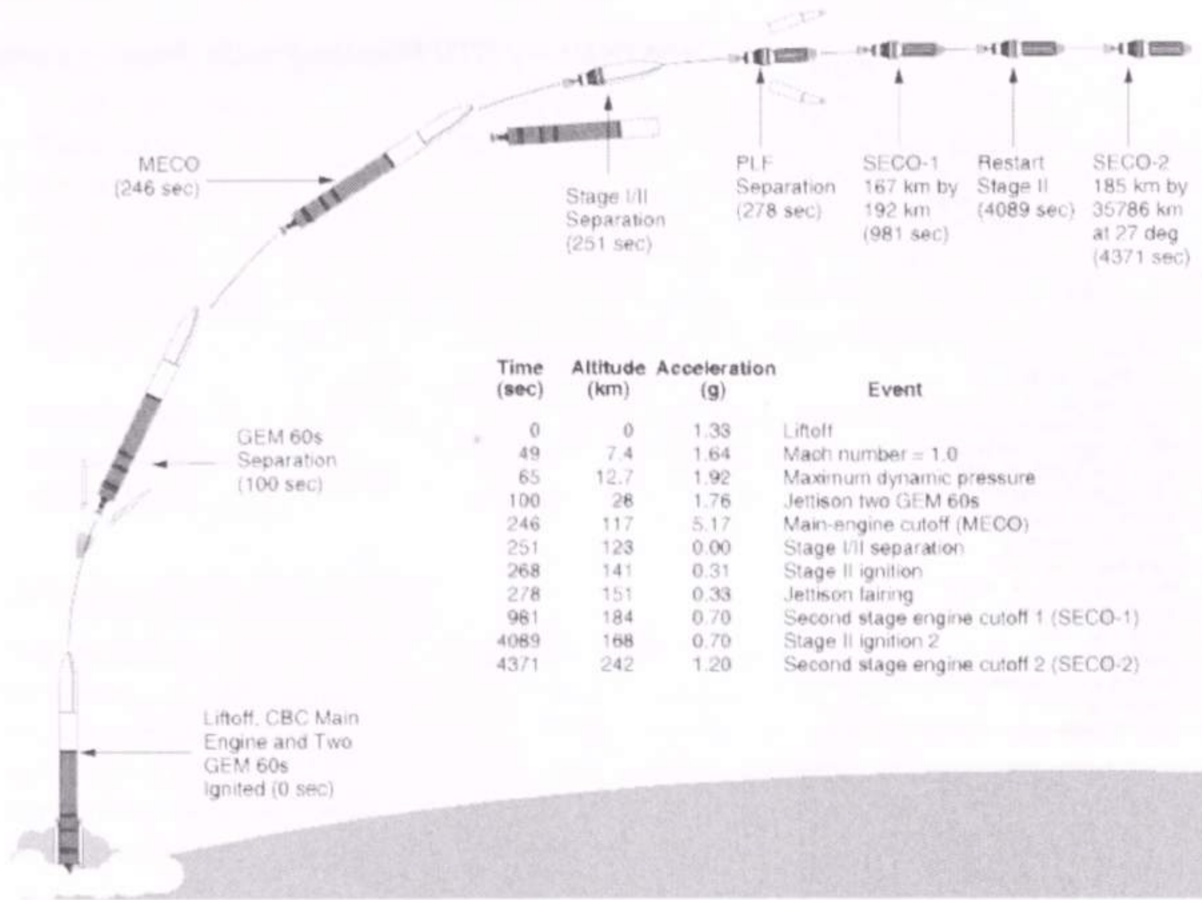
Event	V_I (fps)	Acceleration (g)
Liftoff	1343	1.37
6 SRM Burnout	3339	0.55
MECO	19,944	5.91
SECO I	25,560	0.67
SECO II	27,192	0.76
Stage III Burnout	33,589	3.24

Delta II GTO Mission Profile (Eastern Range)

Delta II event sequence for GTO missions

Event (Time, s)	Delta II 7325	Delta II 7925
Main-engine ignition	0.0	0.0
Solid-motor ignition	0.0	0.0
Solid-motor burnout	63.0	63.0
Air-lit solid motor ignition	—	65.5
Solid motor separation	66.0	66.0 and 67.0
Air-lit motor burnout	—	129.0
Air-lit motor separation	—	132.0
Main-engine shutdown	261.0	261.0
First-stage separation	269.0	269.0
Second-stage ignition	274.5	274.5
Fairing separation	300.0	300.0
Second-stage shutdown	676.0	617.0
Second-stage restart	871.0	881.0
Second-stage shutdown	892.0	950.0
Start third-stage spin rockets	942.0	1000.0
Second-stage separation	945.0	1003.0
Third-stage ignition	982.0	1040.0
Third-stage burnout	1069.0	1127.0
Spacecraft separation		

PRODUCTION AND LAUNCH OPERATIONS



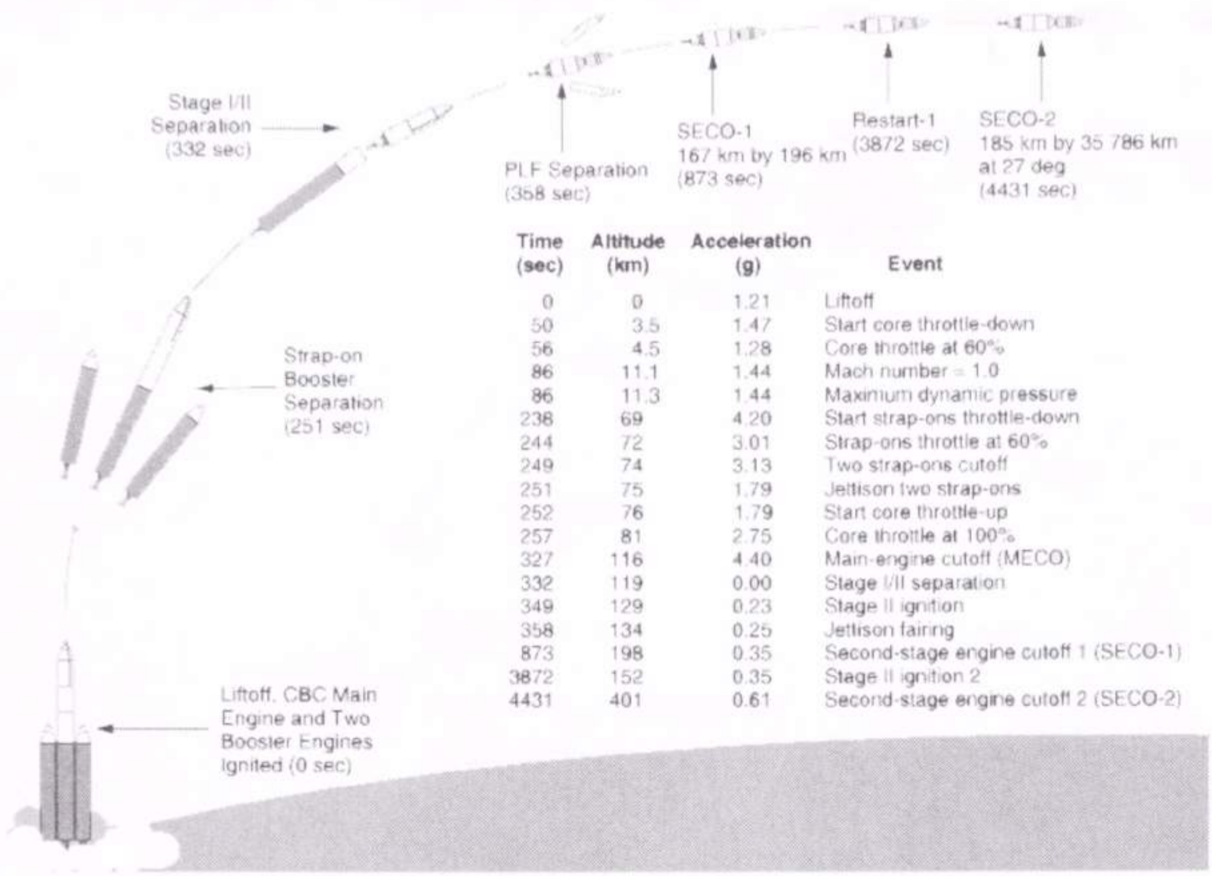
Delta IV Medium+(5,2) GTO Mission Profile (Eastern Range)

Delta IV Medium + (5,2) Event Sequence for GTO Missions (Eastern Range)

Event	Time, s	Altitude, km	Acceleration, g
Liftoff	0	0	1.33
Mach 1.0	49	7.4	1.64
Maximum dynamic pressure	65	12.7	1.92
Jettison of two GEM 60s	100	28	1.76
Main-engine cutoff (MECO)	246	117	5.17
First-stage separation	251	123	0.00
Second-stage ignition	268	141	0.31
Jettison of fairing	278	151	0.33
Second-stage engine cutoff (SECO-1)	981	184	0.70
Second-stage ignition	4089	168	0.70
Second-stage ignition cutoff (SECO-2)	4371	242	1.20

Production and Launch Operations

Delta IV Heavy GTO Mission Profile (Eastern Range)



Delta IV Heavy Event Sequence for GTO Missions (Eastern Range)

Event	Time, s	Altitude, km	Acceleration, *g*
Liftoff	0	0	1.21
Start core throttle down	50	3.5	1.47
Core throttle at 60%	56	4.5	1.47
Mach 1.0	86	11.1	1.44
Maximum dynamic pressure	86	11.3	1.44
Start strap-on throttle down	238	69	4.20
Strap-on throttle at 60%	244	74	3.13
Two strap-ons cutoff	244	72	3.01
Jettison of two strap-ons	251	75	1.79
Start core throttle up	252	76	1.79
Core throttle at 100%	257	81	2.75
Main-engine cutoff (MECO)	327	116	4.40
First-stage separation	332	119	0.00
Second-stage ignition	349	129	0.23
Jettison of fairing	358	134	0.25
Second-stage engine cutoff (SECO-1)	873	198	0.35
Second-stage ignition	3872	152	0.35
Second-stage ignition cutoff (SECO-2)	4431	401	0.61

Vehicle History

Vehicle Description

•Delta	Modified Thor first stage with MB-3 Block 1 engine, second-stage AJ10-118 propulsion system from Vanguard, third-stage Vanguard X-248 motor
•Delta A	First-stage engine replaced with MB-3 Block 2
•Delta B	Second-stage tanks lengthened, higher energy oxidizer used
•Delta C	Third stage replaced with Scout X-258 motor, new wider payload fairing
•Delta D	Three Castor I solid boosters
•Delta E	Three Castor II solid boosters, MB-3 Block III first-stage engine, wider second-stage tanks, FW-4 third-stage motor, 1.65-m (65-in) payload fairing
•Delta J	TE-364-3 third stage
•Delta L,M, N	Longer propellant tanks on first stage, RP-1 tank widened, vehicles used FW-4, TE-364-3, or no third stage respectively
•Delta M-6, N-6	Six Castor II solid boosters
•Delta 900	Nine Castor II solid boosters
•Delta 1604	Six Castor II solid boosters, second-stage AJ10-118F engine from Transtage
•Delta 1910, 1913, 1914	Nine Castor II solid boosters, optional TE-364-3 or 4 third-stage motor, new 2.44-m (96-in) payload fairing
•Delta 2310, 2313, 2314	Three Castor II solid boosters, RS-27 engine on first-stage TR-201 second-stage engine
•Delta 2910, 2913, 2914	Nine Castor II solid boosters

Vehicle History

•Delta 3910, 3913, 3914,	Nine Castor IV solid boosters
•Delta 3920, 3924	AJ10-118K second-stage engine
•Delta 4920	Castor IVA replaced Castor IV, MB-3 first-stage engine
•Delta 5920	RS-27 first-stage engine
•Delta 6925	First-stage tanks stretched 3.7 m (12 ft), Star 48B third-stage engine, new 2.9-m (9.5-ft) payload fairing
•Delta 7925	GEM strap-on motors, RS-27A main engine
•Delta 7326, 7425	Three or four GEM strap-on motors, new upper stage option based on Star 37FM motor
•Delta III	GEM-46 strap-on motors, shorter and wider first-stage RP-1 tank, new cryogenic second stage, 4-m (13.1-ft) composite fairing
•Delta IV-M	New common booster core cryogenic first stage with RS-68 main engine, larger second stage based on Delta III
•Delta IV-M+	Two or four GEM-60 solid motors, 4-m (13.1-ft) or 5.1-m (16.7-ft) second stage, optional 5.1-m (16.7-ft) fairing
•Delta IV-H	Two common booster core liquid strap-on boosters, 5.1-m (16.7-ft) second stage and fairing

Historical Summary

The Delta launch vehicle family originated in 1959 when NASA Goddard Space Flight Center awarded a contract to Douglas Aircraft Company, now part of Boeing, to produce and integrate 12 launch vehicles. The Delta, using components from the USAF Thor IRBM program and the U.S. Navy Vanguard launch vehicle program, was available 18 months after go-ahead. On 13 May 1960 the first Delta was launched from Cape Canaveral Air Force Station with a 81-kg (179-lbm) Echo I passive communications satellite. Although this first flight was a failure, the ensuing series of successful launches established Delta as one of the most reliable U.S. boosters. In the years since the first vehicles were produced, the Delta has evolved to meet the ever increasing demands of its payloads, including weather, scientific, and communications satellites.

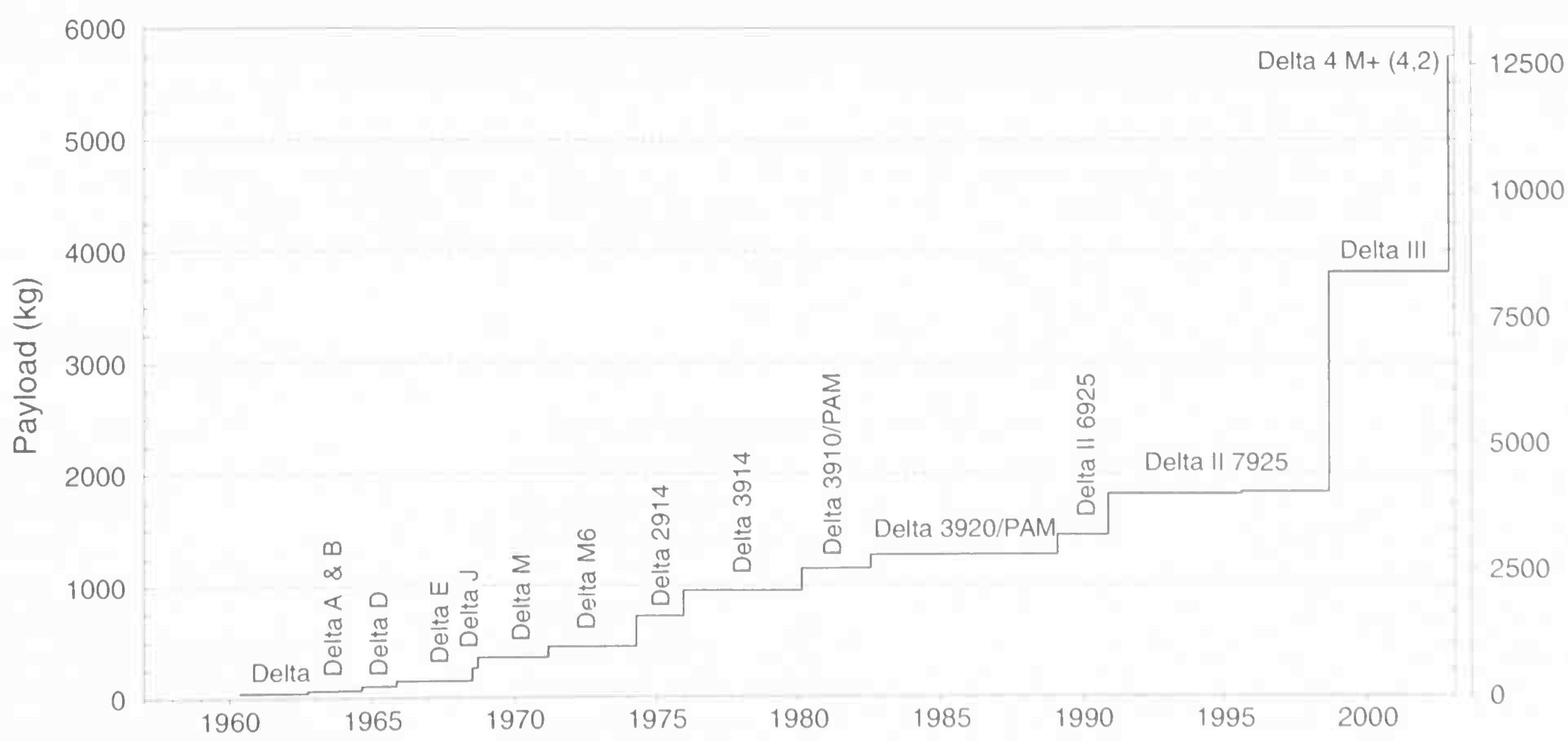
The 1960 Delta used a modified Thor booster with a Rocketdyne MB-3 engine as a first stage; the Vanguard second-stage engine, Aerojet's AJ10-118, on the second stage; and the Vanguard X-248 SRM as the third stage. In 1962, the Delta A configuration evolved with an improved MB-3 engine on the Thor first stage, and then the Delta B added a lengthened second stage with a higher energy propellant. The Delta C configuration had a bulbous fairing and replaced the Vanguard-developed third-stage motor with the Scout-developed X-258 motor. In 1964, three Thiokol Castor I solid strap-on motors were used on the Delta D. The Delta E of 1965 showed three major changes: larger payload fairing from the USAF Agena stage, larger diameter propellant tanks and restart capability on the second stage, and replacement of the X-258 third stage with the USAF-developed FW-4. In addition, more powerful Castor II strap-on motors were used.

In 1968, the third stage again was changed to the Thiokol TE-364-3 motor on the Delta J configuration. Modifications in 1968 also included lengthening the first-stage propellant tanks and increasing the RP-1 tank diameter on the Delta L, M, and N vehicles. These nearly identical configurations had different third stages—the FW-4, TE-36, and none, respectively. The M-6 and N-6 Deltas raised the number of solid strap-on boosters to six in the early 1970s, increasing performance by approximately 27%. In 1972, the number of solid strap-on boosters increased again from six to nine, and the Titan Transtage engine (Aerojet AJ-10-118F) replaced the original Vanguard second-stage propulsion system on the Delta 900.

In the early 1970s, eight variations of the 1000 series of Delta vehicles and the now familiar Delta nomenclature emerged. All eight configurations incorporated some first-stage changes but varied in the number of Castor II strap-on boosters and the type of third stage used (if any). The later 1900 series vehicles possessed a larger payload fairing—from 1.65 m to 2.44 m (from 65 in. to 96 in.). For example, the Delta 1914 had a larger payload fairing, the extended first-stage tank length, an isogrid structure in the first stage, and a longer third-stage motor.

McDonnell Douglas developed five types of Delta 2000 series vehicles. These vehicles, primarily flown between 1974 and 1978, again differed in the type of third stage and the number of solid Castor II SRMs. Two major evolutions occurred with the development of these vehicles. First, the 2000 series vehicles no longer employed the Aerojet AJ10-118F second-stage engine. Instead, McDonnell Douglas opted for the TRW TR-201 engine (a modification of the descent rocket of the lunar module). Second, the MB-3 main engine was replaced by the more powerful Rocketdyne RS-27 engine.

The Delta configurations of the late 1970s and early 1980s were designated the 3900 series, principally the 3914 and 3910/PAM. The 3900 series is very similar to the 2900 series except the 2900 employed Castor II solid strap-on boosters, whereas the 3900 uses nine larger and more powerful Castor IV



Evolution of Delta Performance to GTO

VEHICLE HISTORY

solid motors. Each Castor IV produced 378 kN (85,000 lbf) of thrust versus the 232 kN (52,200 lbf) of the Castor II. The new PAM was used as the third stage of the 3910/PAM vehicle. The PAM stage was based on the Star 48B motor, and was originally developed for the Space Shuttle to transfer satellites from a low Earth parking orbit to their final orbits.

In 1979, McDonnell Douglas realized that the transition to the Space Shuttle would take longer than anticipated. Not only did this mean that more Delta launches would be required, but the Delta configuration had to evolve to accommodate payloads designed for the higher capability of the Space Shuttle; therefore, the Delta 3920 was developed. The TR-201 second-stage engine was replaced by a new Aerojet engine designated the AJ10-118K. This engine was designed by Aerojet for the second stage of the Japanese N-II launch vehicle, using the Titan Transtage engine with larger tankage. The U.S. Air Force also had funded a program to increase the performance of the Transtage injector known as the Improved Transtage Injector Program (ITIP). McDonnell Douglas decided to increase the 3920 performance with minimum risk by using the ITIP engine and the existing N-II stage tanks for the second stage.

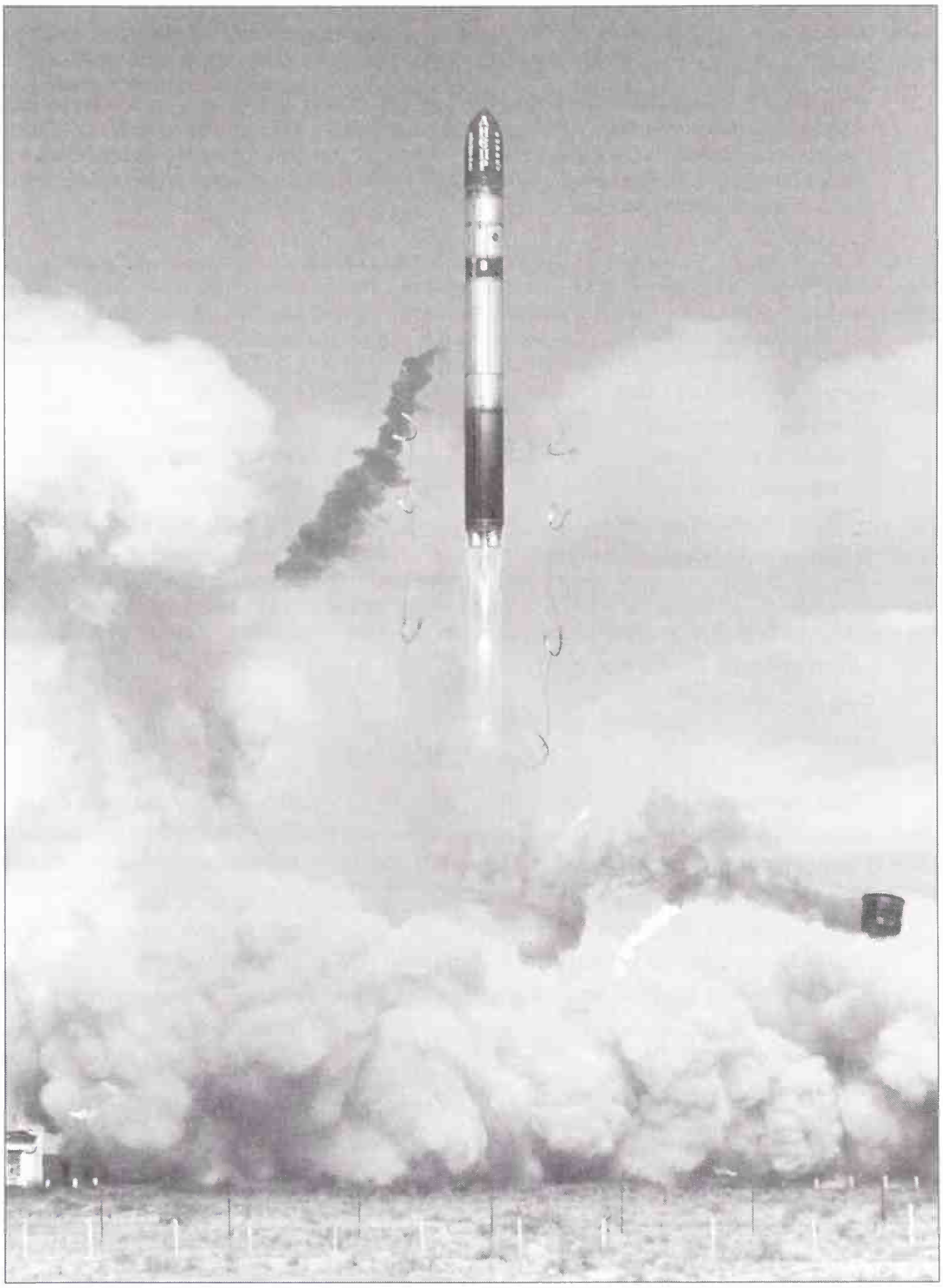
The unprecedented string of launch vehicle failures that occurred 1985 through 1987 seriously impeded U.S. space launch capability. In one of several steps to revitalize this, the U.S. Air Force held a competition for a medium launch vehicle that primarily would launch the GPS satellites. The contract was awarded to McDonnell Douglas for their Delta II series vehicles, designated the 6925 and 7925. These configurations were growth versions of the 3920/PAM-D vehicles. In 1992, the 6920 series vehicle was phased out.

Demand for Delta launches increased significantly in the mid and late 1990s. In addition to continuing GPS deployment missions, Delta was selected for a number of NASA payloads as a result of the agency's focus on smaller, less expensive spacecraft. These include a series of missions to Mars beginning with Mars Pathfinder, as well as missions to asteroids and comets, and satellites dedicated to Earth observation or astronomy. In 1994, NASA announced the Med-Lite program in which it planned to purchase smaller launch vehicles, described as "half a Delta for half the price." NASA selected the Delta 7300 and 7400 vehicles, which have fewer solid strap-on motors than the 7900 series. While somewhat larger and more expensive than the program goal, the new vehicles could be developed rapidly with minimal development cost and risk. The Delta II manifest continued to expand with numerous launches to deploy LEO communications constellations such as Iridium and Globalstar. However, McDonnell Douglas realized that commercial GEO satellites were rapidly growing too heavy for Delta II to lift, and that a new vehicle was needed. In response, the Delta III was developed. The strap-on motors were expanded, and an all new cryogenic upper stage was developed, powered by the proven RL10 engine. Commercial satellite builders responded positively, with Hughes purchasing 10 launches (later expanded to 13), and Loral purchasing five flights. The first two Delta III flights ended in failure in 1998 and 1999. After two failures for paying customers, Boeing was forced to perform a demonstration launch with a dummy payload in 2000. The launch succeeded, but by that time a combination of factors including Delta III's tarnished reputation, overcapacity in the launch market, and a growing number of spacecraft that were too heavy for Delta III to lift, made it difficult to sell any further launches. Marketing efforts were eventually abandoned in favor of the newer Delta IV. Delta III hardware, such as engines and solid rocket motors, was converted for use on Delta II and Delta IV.

During the early years of the 21st century, NASA's science payloads have continued to grow. In response Boeing developed a new heavy-lift Delta II configuration offering a 13 percent increase in capability by using larger strap-on graphite epoxy motors from Delta III, called the Delta II Heavy. The first three-stage Delta II Heavy successfully deployed the Mars Exploration Rover *Opportunity* on 7 July 2003. The first two-stage Delta II Heavy successfully launched the Space Infrared Telescope Facility on 25 August 2003.

The Delta IV was designed as part of the U.S. Air Force Evolved Expendable Launch Vehicle (EELV) program, which began in 1995. In November 1996 the U.S. Air Force awarded four $30-million phase one study contracts to Alliant Techsystems, The Boeing Company, Lockheed Martin Corporation, and McDonnell Douglas Aerospace. During the second phase, preengineering and manufacturing development, two $60-million, 17-month contracts were awarded to Boeing (which had just merged with McDonnell Douglas and continued the McDonnell Douglas proposal) and Lockheed Martin to continue refining their system concepts and complete a detailed system design. The Lockheed Martin approach is based on an evolution of their existing Atlas II/III product line, with the McDonnell Douglas design being an evolution of their Delta II/III products. Originally the U.S. Air Force planned a winner take all competition, in which one contractor would provide EELV launch services to the government. At that time the U.S. Air Force believed that the total market was too small to sustain two competing launch systems. However, in reviewing the market with the industry-led Commercial Space Transportation Advisory Committee (COMSTAC), the U.S. Air Force was persuaded that the commercial market was large enough to support competition between the EELV winners. In November 1997 the U.S. Air Force announced that it intended to introduce competition across the lifespan of the EELV program, to encourage greater contractor investment and competition in the U.S. space launch industry, and decrease the Air Force's overall development cost. The U.S. Air Force selected both the Delta IV and Atlas V configurations, with competitive bidding expected to split the government missions, while providing increased competition within the commercial market. The inaugural Delta IV flight of a Medium Plus (4,2) configuration launched a commercial satellite for Eutelsat on 20 November 2002, and has been followed by two successful EELV flights for the U. S. Air Force (two Delta IV Mediums). The Delta IV manifest currently stands at 15 launches, with 11 for the U.S. Air Force and 4 for NASA. Upcoming milestones include the first flight of the Delta IV Heavy and the first EELV launch from SLC-6 at Vandenberg AFB.

DNEPR



Courtesy ISC Kosmotras.

The Dnepr is an R-36M2 (SS-18 Mod 4) heavy ICBM that has been converted into a space launch system. Commercialization began early in the 1990s and the first flight took place in 1999. Dnepr is capable of launching either single or multiple medium and small spacecraft to LEO. The second Dnepr-1 launch on 26 September 2000 is shown.

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General Description

Dnepr

Dnepr-1

Summary

The Dnepr is a converted R-36M2 ICBM (RS-20, SS-18 Mod 4) developed by a team of companies headed by SDB Yuzhnoye and manufactured by PA Yuzhniy Machine Building Plant of Ukraine. It is silo launched from Baikonur and is capable of deploying multiple small to medium spacecraft into LEO. Dnepr launch services are provided by International Space Company (ISC) Kosmotras, a partnership of the Russian and Ukrainian organizations that are responsible for Dnepr production and launch operations.

Status

Operational. First launch in 1999.

Origin

Ukraine and Russia

Key Organizations

Marketing Organizations	ISC Kosmotras
Launch Service Provider	ISC Kosmotras
Prime Design Company	SDB Yuzhnoye
Primary Production Company	PA Yuzhniy Machine Building Plant

Primary Missions

Small and medium payloads to LEO

Estimated Launch Price

$8–11 million (ISC Kosmotras, 2002)

Spaceport

Launch Site	Baikonur, silos 103, 104, 106, 109
Location	45.6° N, 63.4° E
Available Inclinations	50.5, 64.8, 87.1, and 97.8 deg, or others with yaw steering maneuvers

Performance Summary

300 km (162 nmi), 50.6 deg	3700 kg (8150 lbm)
300 km (162 nmi), 87.3 deg	2650 kg (5840 lbm)
Space Station Orbit: 407 km (220 nmi), 50.5 deg	3400 kg (7490 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	300 kg (661 lbm)
GTO	No capability
Geostationary Orbit	500 kg (1100 lbm) with optional upper stages

Flight Record (through 31 December 2003)

Total Orbital Flights	3
Launch Vehicle Successes	3
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–1 per year demonstrated, 25–27 possible

Nomenclature

The Dnepr is a decommissioned and modified R-36M2 ICBM. The ICBM version is also known by its treaty designation as the RS-20, and designated by the U.S. and NATO as SS-18 Mod 4 "Satan." The name "Dnepr" comes from the Dnepr River, which runs through both Russia and Ukraine, including the city of Dnepropetrovsk, Ukraine, where the missiles were originally designed and built. In early proposals, Dnepr was also referred to as Ikar, or as the SS-18K, where K stands for Konversiya (Conversion).

Cost

In 2002, Kosmotras reported that typical prices for a dedicated Dnepr launch were about $8–11 million. The lower end of the price range is typical of smaller satellites without electrical interfaces to the launch vehicle, while the higher end of the range covers larger satellites with more complex interfaces to the launch system. For cluster launches of multiple small satellites, Kosmotras charges each customer $10,000–12,000 per kilogram. These low prices are possible because only minor modifications are required to the refurbished missiles and launch facilities.

Availability

Dnepr launch services are provided by International Space Company Kosmotras, created in Moscow in 1997. Dnepr launch services are available both for domestic Ukrainian and Russian launches and for international commercial launches. The first launch was conducted in 1999, carrying a commercial spacecraft. Dnepr launch vehicles are modified from existing R-36M2 ICBM assets. At least 308 R-36M, M1, and M2 missiles were deployed in Russia and Kazakhstan. All remaining missiles are now located in Russia. As of January 2002, START I treaty documents indicated that Russia had 144 deployed R-36M2 missiles and four non-deployed missiles. Dnepr could perform as many as 25 to 27 launches per year should the market demand it.

Performance

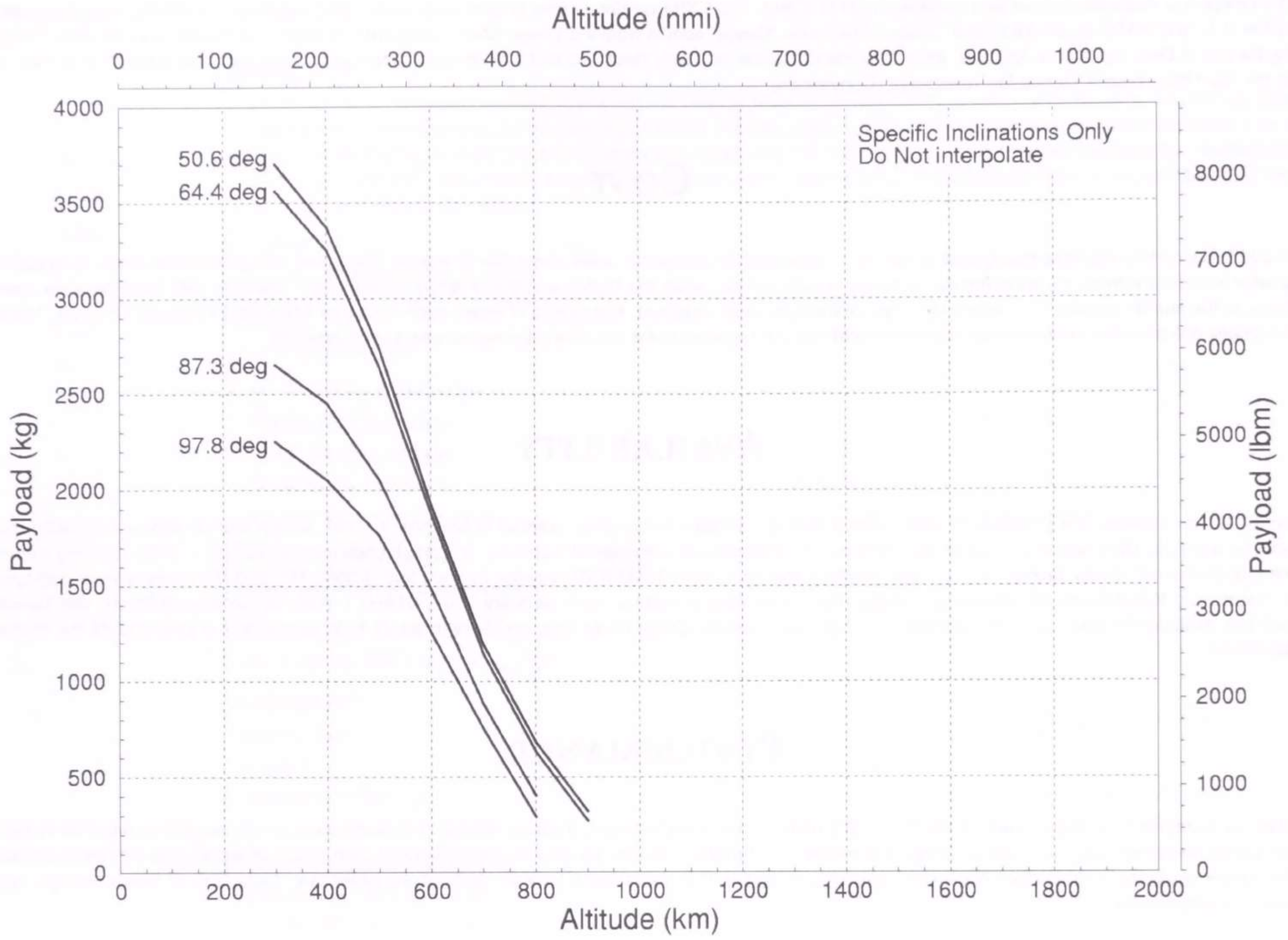
Dnepr is launched from missile silos at the Baikonur Cosmodrome in Kazakhstan. Because Baikonur is landlocked, jettisoned stages must fall in specific zones away from populated areas. Dnepr is therefore constrained to fly along a small number of launch azimuths that are aligned with these zones. The Dnepr guidance system does have yaw steering capabilities that can allow it to reach inclinations other than those shown below, though with reduced performance.

Launch Azimuth, deg	Orbit Inclination, deg
35.1	64.8
64.3	50.5
177.6	87.1
192.9	97.8

The R-36M2 is the largest ICBM ever deployed, giving the Dnepr higher performance than other launch systems converted from ICBMs. However, like most converted ICBMs the basic Dnepr-1 can only reach low altitude orbits, because it lacks sufficient upper-stage propulsion to circularize orbits at higher altitudes.

The Dnepr launch vehicle does not have the capability to deploy payloads directly into GTO. However, Kosmotras has studied a technique to deliver small spacecraft to GEO using the gravity of the Moon to provide the plane change and perigee raising. In this scenario, the spacecraft is attached to Star 48A and Star 27 solid motors, supplied separately by ATK Thiokol. The Star 48A would send the spacecraft to the Moon, where a gravity slingshot maneuver would lower the transfer orbit inclination from 50.5 deg to 0 deg, and raise the orbit perigee to geostationary altitude. When the spacecraft reaches perigee of the new transfer orbit, the Star 27 motor would fire to circularize the orbit at GEO. Using this method, a 500 kg (1100 lbm) spacecraft could be delivered to GEO.

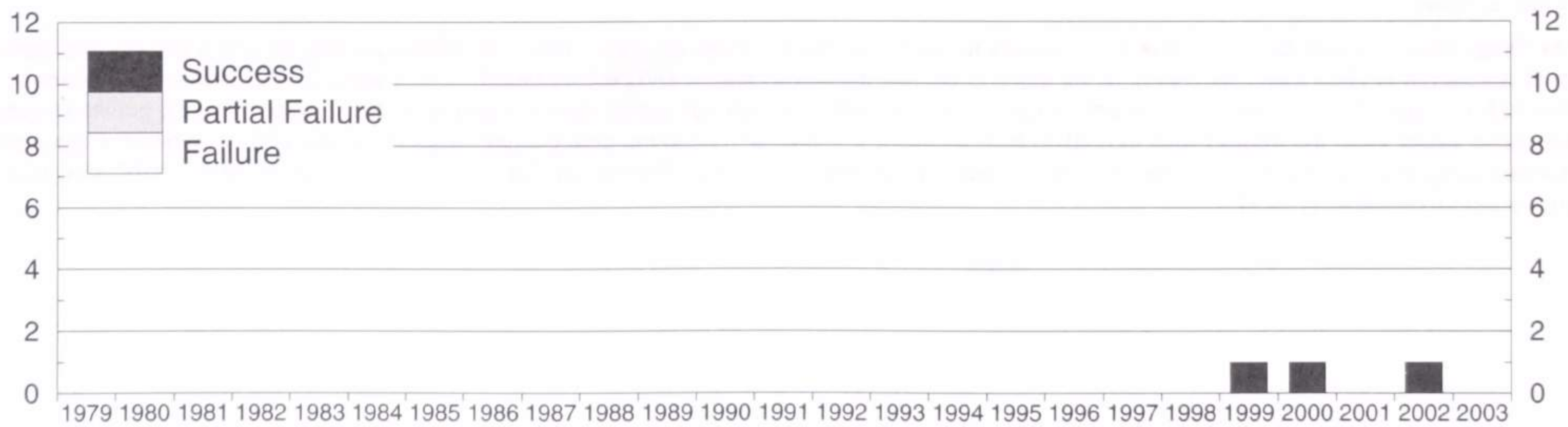
Performance



Dnepr-1: LEO Capability

Flight History

Orbital Flights Per Year



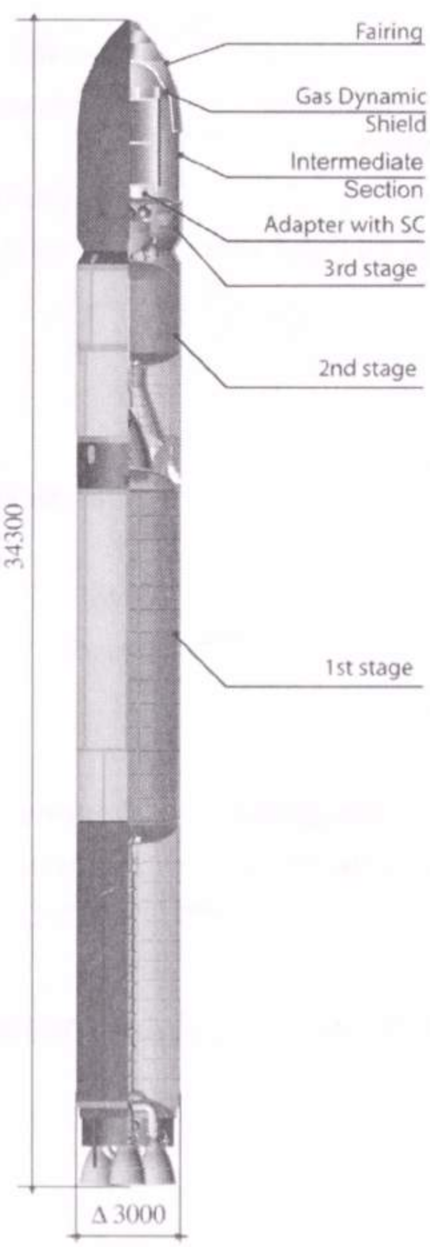
Flight Record (through 31 December 2003)	
Total Orbital Flights	3
Launch Vehicle Successes	3
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight History

In addition to the orbital launches of Dnepr shown below, there have been 157 suborbital launches of the nearly identical R-36M2 missile. According to Kosmotras, four of these flights suffered malfunctions or anomalies. Kosmotras estimates the reliability of the Dnepr to be 97%.

		Date (UTC)	Interval (days)	Vehicle	Pad	Payload Desig.		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
1	1999	Apr 21	-	6703542509	LC 109/95	1999 021A		UoSat 12	550	LEO (65)	NGO	UK
2	2000	Sep 26	524		LC 109/95	2000 057A	C	Tiungsat 1	50	LEO (65)	CIV	Malaysia
					LC 109	2000 057B	C	MegSat 1	50	LEO (65)	CML	Italy
					LC 109	2000 057C	A	UniSat 1	10	LEO (65)	NGO	Italy
					LC 109	2000 057D	A	SaudiSat 1A	10	LEO (65)	CIV	Saudi Arabia
					LC 109	2000 057F	A	SaudiSat 1B	10	LEO (65)	CIV	Saudi Arabia
3	2002	Dec 20	815		LC 109/95	2002 058A	C	UniSat 2	14	LEO (64.57)	NGO	Italy
						2002 058B	C	LatinSat 1	12	LEO (64.57)	CML	Argentina
						2002 058C	C	LatinSat 2	12	LEO (64.57)	CML	Argentina
						2002 058D	C	SaudiSat 1S	15	LEO (64.57)	CIV	Saudi Arabia
						2002 058E	A	Trailblazer mockup	14	LEO (64.57)	CML	USA
						2002 058F	A	Rubin 2	14	LEO (64.57)	CIV	Germany

Vehicle Design



Courtesy ISC Kosmotras.

Overall Vehicle

	Dnepr
Height	34.3 m (112.5 ft)
Gross Liftoff Mass	208.9 t (356 klbm)
Thrust at Liftoff	Vacuum: 4525 kN (1017 klbf)

Vehicle Design

Stages

The Dnepr space launch vehicle is a decommissioned R-36M2 (SS-18 Mod 4) missile, the largest ICBM ever deployed. The R-36M2 has two primary stages plus a small post-boost stage, all of which are powered by storable N_2O_4 and UDMH propellants. Only minor hardware and software modifications to the upper stage (the R-36M2 post-boost stage) and the payload adapter are required for the Dnepr-1. Because of security considerations, relatively little technical information is available regarding the vehicle design. The first stage has a propulsion system with four thrust chambers, fed by two tanks with a common bulkhead. The oxidizer main runs through the middle of the lower fuel tank. The second stage has a single main engine and four separate steering verniers. This RD-0255 propulsion system is a descendant of the RD-0210 engine used on the Proton third stage. The Dnepr-1 third stage uses the R-36M2 missile post-boost propulsion system for precision orbit injection. The thrusters of this system point forward instead of aft. To be used for orbit injection, the stage must flip around to point its thrusters (and the payload) backward. To protect the payload from possible contamination by the thrusters, and to ensure the required cleanliness around the spacecraft at all stages of preparation for launch, the third stage includes a protective cover and a gas-dynamic shield (or Encapsulated Payload Module) that remain in place around the spacecraft after the aerodynamic nose fairing separates.

	Stage 1	**Stage 2**	**Stage 3**
Dimensions			
Length	22.3 m (73.3 ft)	5.7 m (18.7 ft)	1 m (3.3 ft)
Diameter	3.0 m (9.8 ft)	3.0 m (9.8 ft)	3.0 m (9.8 ft)
Mass			
Propellant Mass	147.9 t (326.0 kblm)	36,740 kg (80,975 lbm)	1910 kg (4210 lbm)
Inert Mass	13.6 t (30.0 kblm)	4374 kg (9640 lbm)	2356 kg (5193 lbm)
Gross Mass	161.5 t (356.0 klbm)	41,114 kg (90,615 lbm)	4266 kg (9402 lbm)
Propellant Mass Fraction	0.92	0.89	0.45
Structure			
Type	Skin stringer?	?	?
Material	Aluminum?	Aluminum?	?
Propulsion			
Engine Designation	RD-264	RD-0255	?
Number of Engines	1 propulsion system with 4 thrust chambers	1 main engine + 4 verniers	1 propulsion system with 4 chambers
Propellant	N_2O_4 /UDMH	N_2O_4 /UDMH	N_2O_4 /UDMH
Average Thrust	Vacuum: 4525 kN (1017 klbf)	760 kN (170 klbf)	Main mode: 18.6 kN (4200 lbf) Low thrust mode: 7.8 kN (1800 lbf)
Isp	?	?	?
Chamber Pressure	?	?	?
Nozzle Expansion Ratio	?	?	?
Propellant Feed System	Turbopump	Gas generator turbopump?	?
Mixture Ratio (O/F)	?	?	?
Throttling Capability	No	No	Yes
Restart Capability	None	None	None
Tank Pressurization	?	?	?
Attitude Control			
Pitch, Yaw	Main engine nozzle gimbal	Verniers	Nozzle gimbal
Roll	Main engine nozzle gimbal	Verniers	Nozzle gimbal
Staging			
Nominal Burn Time	?	?	?
Shutdown Process	Command shutdown	Command shutdown	Command shutdown
Stage Separation	?	?	?

Vehicle Design

Attitude Control System

The first stage is controlled by independently gimbaling the four thrust chambers of the propulsion system. The second stage has a single, fixed main engine with four vernier engines for steering. The third stage also uses verniers for steering.

Avionics

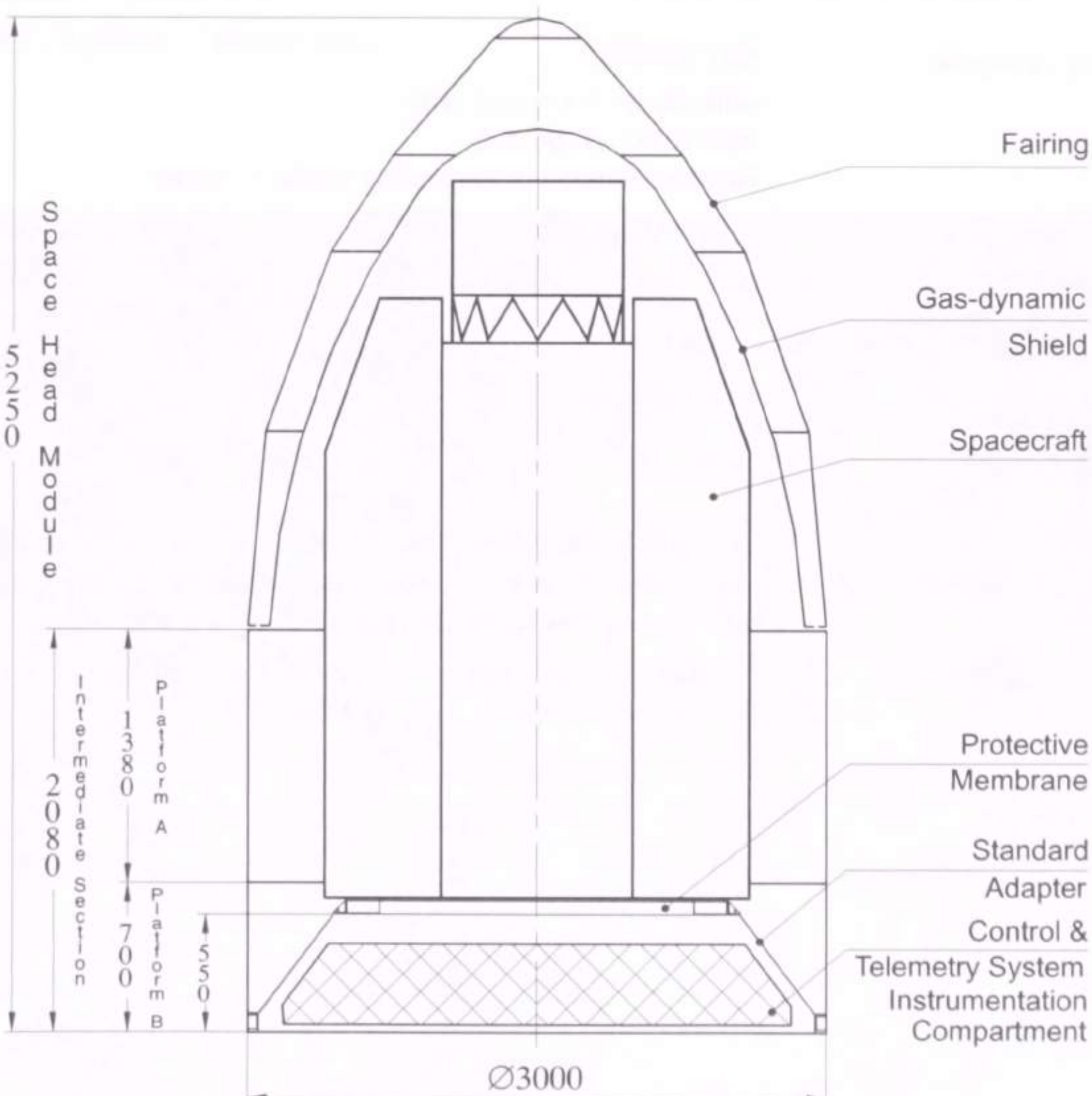
Dnepr uses the same avionics systems developed for the R-36M2. The avionics are mounted on the Dnepr third stage. Flight control is provided by a three channel digital computer with inertial navigation. Sequencing commands (such as ignition and staging) generated by the computer are sent when two of three signal relays are triggered, thus providing redundancy for a relay failing in either the open or closed state. The computer can also send commands or separation timing information to the payload. Launch vehicle telemetry at high and low sample frequencies is transmitted at 512 Kbps to receiving stations in Kazakhstan and Russia.

Payload Fairing/Space Head Module

Dnepr payloads are contained in a Space Head Module, which comprises the payload fairing, a nonseparating intermediate section, a payload adapter with a protective membrane, a Gas Dynamic Shield (GDS) or Encapsulated Payload Module (EPM), and the spacecraft.

The Dnepr payload fairing is the same design used on the R-36M2 ICBM. It is shaped in four conical sections, with a longitudinal joint held together by 28 pyrotechnic devices that can split the fairing into two sections. The sections are mounted to a fixed cylindrical skirt by 8 more pyrotechnic devices. When the devices are activated, the halves are pushed apart by 4 springs on each half, and rotate before being released at a predetermined angle. The intermediate section is a fixed cylindrical spacer that provides additional payload volume. It is typically 2080 mm (81.9 in.) long, but larger units up to 4080 mm (160.6 in) long can be used for larger payloads.

The payload adapter mounts to the bottom ring frame of the intermediate section, and interfaces directly to one or more spacecraft. A protective membrane at the top of the adapter isolates the spacecraft from the third stage avionics compartment. To protect the spacecraft from ground contamination and the exhaust of the forward-pointing third stage thrusters, either a GDS or an EPM is used. The GDS is a thin sheet mounted to the forward end of the intermediate skirt. The EPM is a two-piece shell that is mounted inside the Space Head Module. The GDS or the forward section of the EPM is separated in orbit before spacecraft deployment.



Courtesy ISC Kosmotras.

	Dnepr-1 Fairing
Length	5.25 m (17.25 ft)
Primary Diameter	3.0 m (9.8 ft)
Mass	?
Sections	3 (1 fixed, 2 separating)
Structure	?
Material	?

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	2700 mm (106.3 in.)
Maximum Cylinder Length	1030 mm (40.6 in.) standard, can be extended to 3030 mm (119.3 in.)
Maximum Cone Length	3170 mm (124.8 in.)
Payload Adapter Interface Diameter	2640 mm (104.0 in.)
Payload Integration	
Nominal Mission Schedule Begins	12 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T – 3 min
On-Pad Storage Capability	Limited only by spacecraft
Last Access to Payload	T – 7 days
Environment	
Maximum Axial Load	8.3 *g* (7.8 *g* quasi-static + 0.5 *g* dynamic) in compression 0.5 *g* in tension during stage 3 flight
Maximum Lateral Load	0.5 *g* quasi-static + 0.5 *g* dynamic
Minimum Lateral/Longitudinal Payload Frequency	20 Hz/10 Hz
Maximum Acoustic Level	135 dB at 100–200 Hz
Overall Sound Pressure Level	140 dB
Maximum Flight Shock	1000 *g* at 1–5 kHz
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	1000 W/m^2 (0.09 BTU/ft^2/s^2)
Maximum Pressure Change in Fairing	3.4 kPa/s
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	300 km orbit Altitude: ±4.0 km (2.2 nmi) Inclination: ±0.04 deg Right Ascension of Ascending Node: 0.05 deg
Attitude Accuracy (3 sigma)	±1.5 deg, ±0.5 deg/s
Nominal Payload Separation Rate	?
Deployment Rotation Rate Available	?
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	?
Multiple/Auxiliary Payloads	
Comanifest or Multiple Manifest	Dnepr has flown comanifested missions of up to six satellites for different customers. The large diameter of the payload adapter makes it easy to mount multiple spacecraft on the adapter plane without complex deployment structures.
Auxiliary Payloads	Kosmotras offers flight opportunities for small spacecraft, and several sizes of adapters for payloads as small as 10 kg (22 lbm).

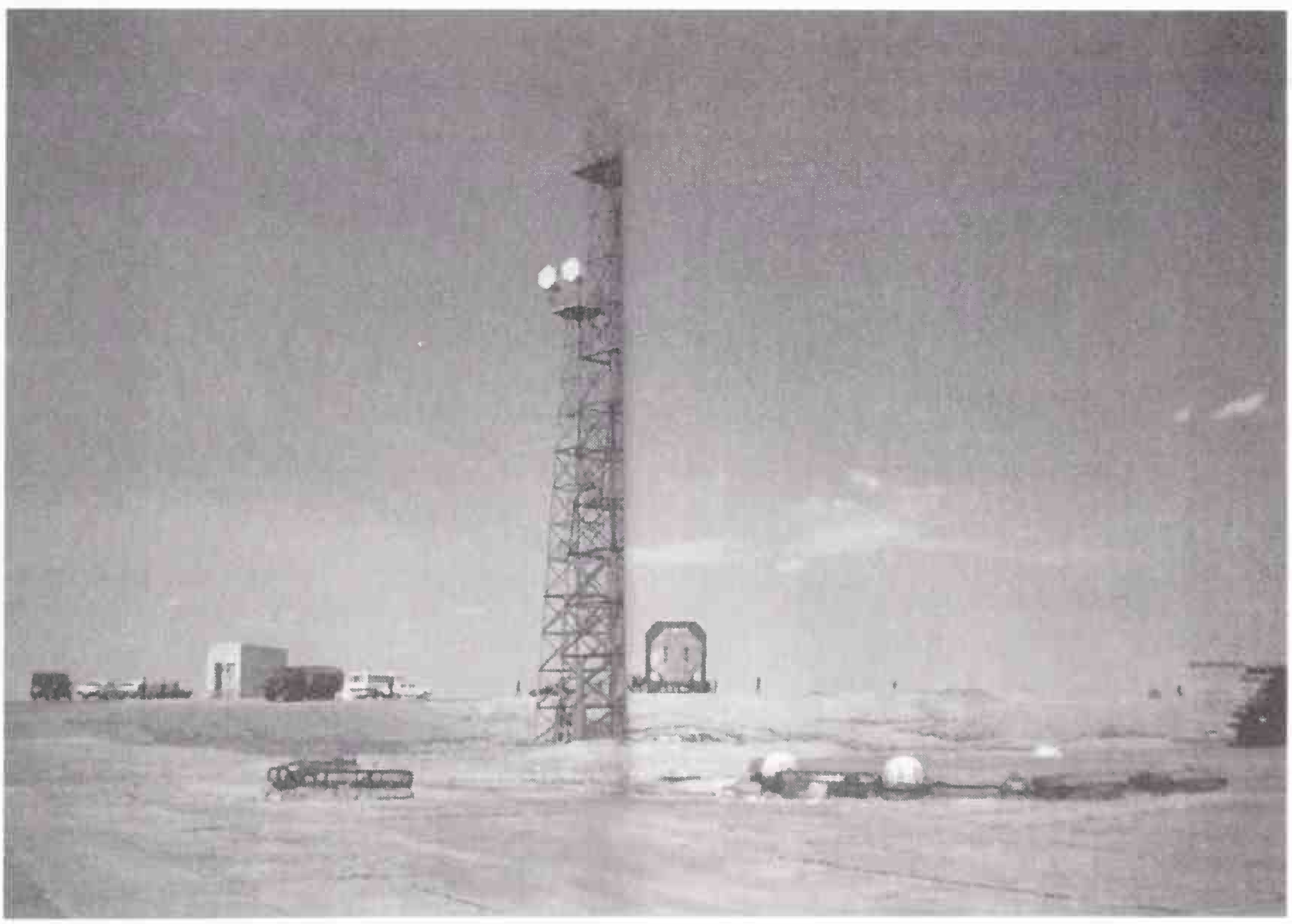
Production and Launch Operations

Production

Production of the R-36M2 ICBM ceased in March 1992. The vehicles were designed by a team of companies headed by SDB Yuzhnoye and built by its factory, PA Yuzhniy Machine Building Plant, both based in Dnepropetrovsk, Ukraine. ISC Kosmotras is the joint stock company responsible for vehicle modifications, marketing, and launch services. The following organizations are members of the Kosmotras team:

Organization	Responsibility
Russian Aviation and Space Agency	State support and supervision of program, and provides facilities and services at Baikonur Cosmodrome
Russian Ministry of Defense	Allocates R-36M2 missiles to be converted into Dnepr launch vehicles, and missile storage and Dnepr standard launch operations
ASKOND joint stock company (Russia)	Manages modification program to turn R-36M2 into Dnepr and handles foreign relations
Rosobschemash joint stock company (Russia)	Coordinates with Russia's Defense Ministry to deactivate R-36M2 for conversion and supports launch operations at Baikonur
Special Machine Building Design Bureau (Russia)	Responsible for development and modification of processing and launch facilities
Central Research Institute of Machine-Building, TSNIIMASH (Russia)	Rocketry research institute responsible for performing studies and experiments pertaining to Dnepr
National Space Agency of Ukraine	State support and supervision of program
SDB Yuzhnoye (Ukraine)	Engineering design organization for R-36M2, responsible for project integration and engineering of Dnepr
PA Yuzhniy Machine Building Plant (Ukraine)	Manufacturing organization
Khartron joint stock company (Ukraine)	Develops and produces control systems for space launch systems
Aerospace Committee of the Ministry of Energy and Mineral Resources (Republic of Kazakhstan)	State support and supervision of program
State Enterprise "INFRAKOS" (Republic of Kazakhstan)	Dnepr program activities at Baikonur Cosmodrome
State Enterprise "INFRAKOS-EKOS" (Republic of Kazakhstan)	Ecological support

Launch Facilities



Courtesy Kosmotras

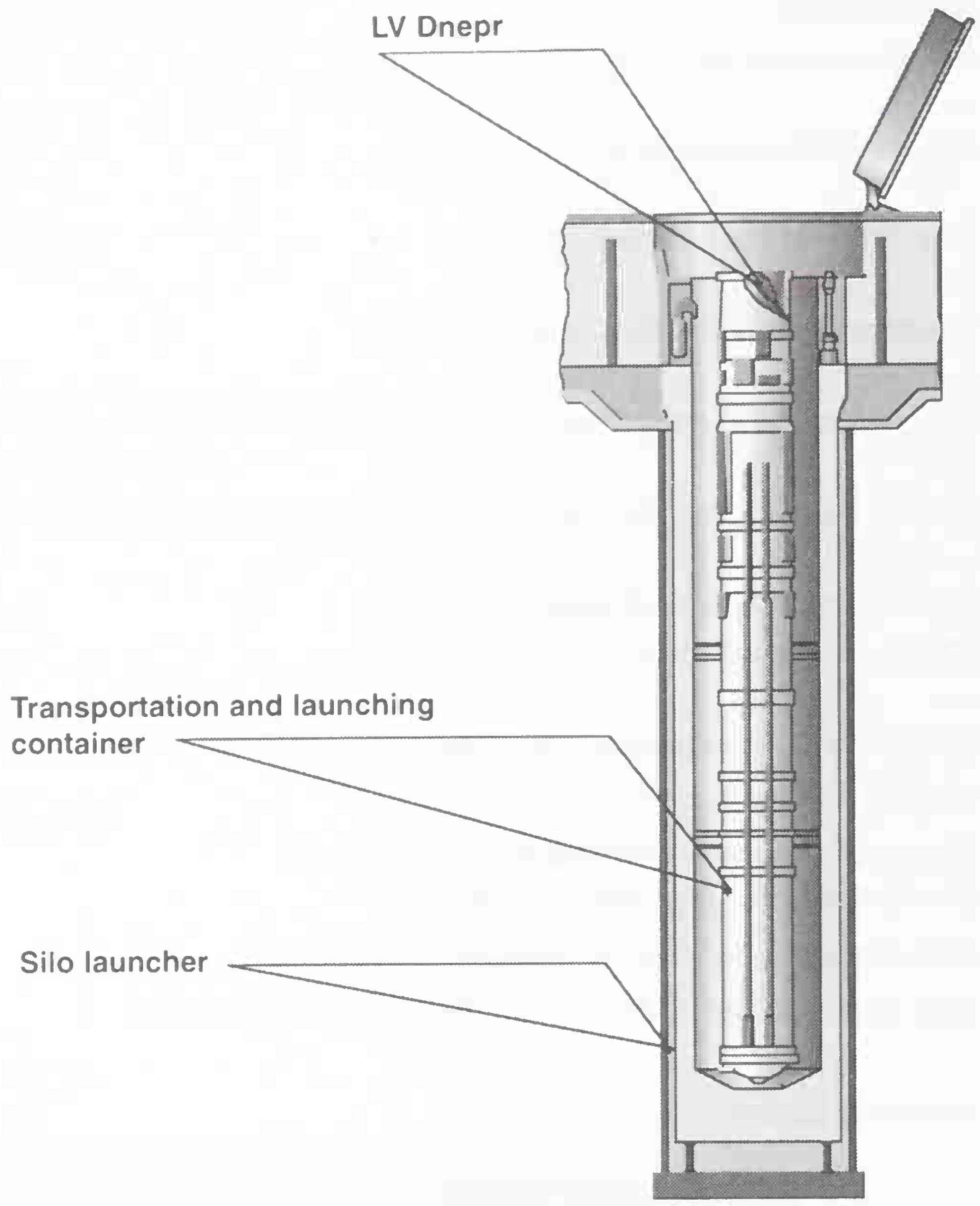
Dnepr launch site at Baikonur. The square object to the right of the communications tower is the open cover of the launch silo.

Production and Launch Operations

Launch Operations

Dnepr is launched from the Baikonur Cosmodrome. (For general information on Baikonur, please refer to the Spaceports chapter.) The Dnepr launch area is located on the eastern branch of the cosmodrome, between Soyuz Launch Complex 1 at the center of Baikonur, and the Soyuz and Zenit facilities on the eastern side. Spacecraft processing and encapsulation into the Space Head Module can be performed at the Assembly and Test Buldings (ATBs) at Area 31 and Area 42, which were built to service Soyuz, Zenit, and Proton launch vehicles and payloads. Four R-36M2 missile launch silos, designated sites 103, 104, 106, and 109, are present at Baikonur. Three of these are available for Dnepr launches. Dnepr launches are controlled from a launch control center at Site 111/2.

The Dnepr vehicle is shipped by rail to the cosmodrome inside its launch container. The launch vehicle and its launch container are loaded into transport containers for transfer from the receiving area to the silo. The vehicle and launch container are then inserted into the silo. The guidance and control unit is installed on the third stage, which is fueled and delivered separately, and then mated to the launch vehicle in the silo. Ground servicing lines and telemetry equipment are installed, and electrical checks are performed from the command post. The vehicle is then fueled and placed in standby mode, awaiting arrival of the payload. When the encapsulated payload arrives, it is integrated onto the launch vehicle in the silo. Telemetry systems and electrical connections between the vehicle and spacecraft are checked. When preparations are completed, the silo door is closed to protect launch equipment from precipitation and to ensure required temperature and humidity conditions. At this point the fueled launch vehicle can remain in a ready state for a long period of time, limited only by the spacecraft. Preparation of the launch vehicle and integration of the encapsulated spacecraft requires about 120 hours (15 single shift workdays) to reach this state. On the launch day, prelaunch operations are performed in less than 3 h. At T−20 min activities are completed and all personnel evacuate the launch complex. The launch command is given by an operator at a preset time, and all final prelaunch and launch operations are conducted by the onboard vehicle computer. The last countdown hold, in case of problems with the spacecraft, is possible at T–3 min.The vehicle is ejected from its silo by a hot-gas generator. Once the vehicle clears the silo, the main engines ignite.

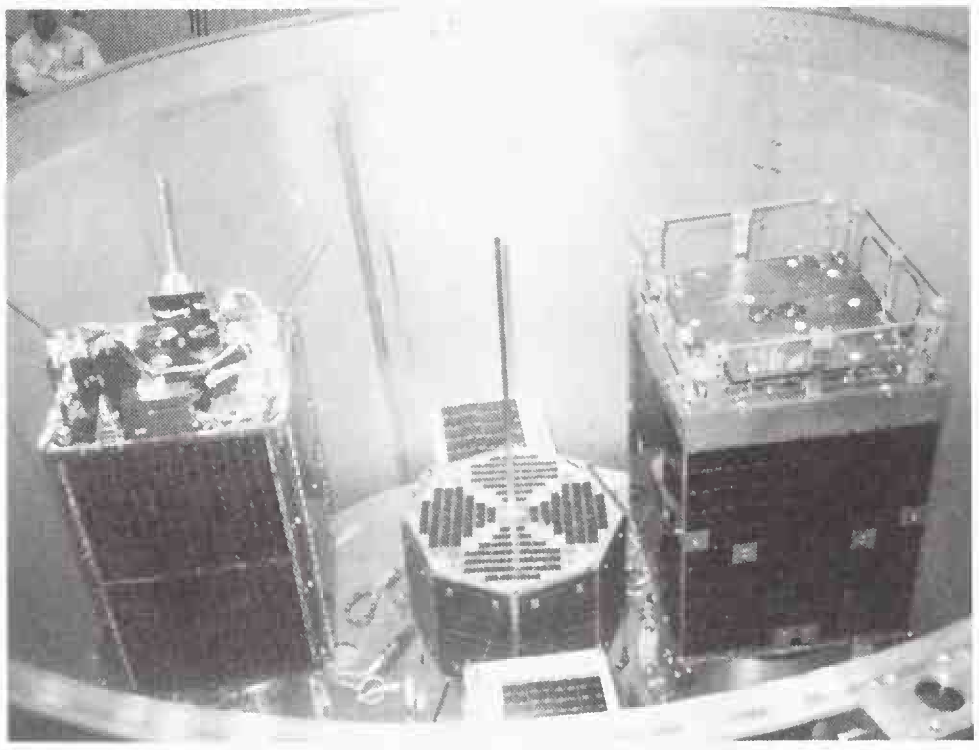


Courtesy ISC Kosmotras.

Dnepr Launch Silo

Courtesy ISC Kosmotras.

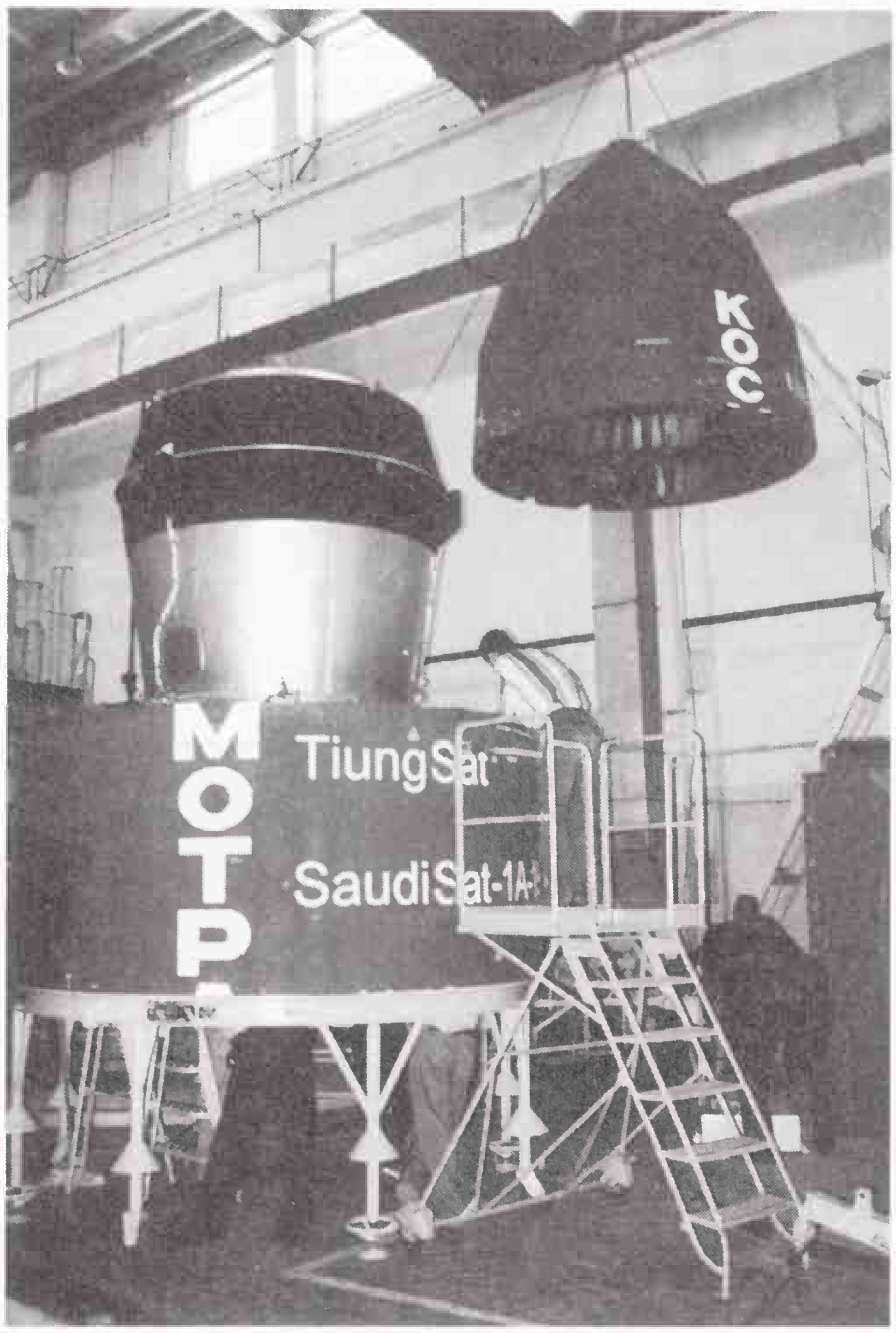
The lower two stages of the launch vehicle, in their protective Transport and Launch Canister, are towed to the launch complex and loaded into the silo.



Courtesy ISC Kosmotras.

Spacecraft are encapsulated in the Encapsulated Payload Module, which protects them from the exhaust of the third stage thrusters.

Production and Launch Operations

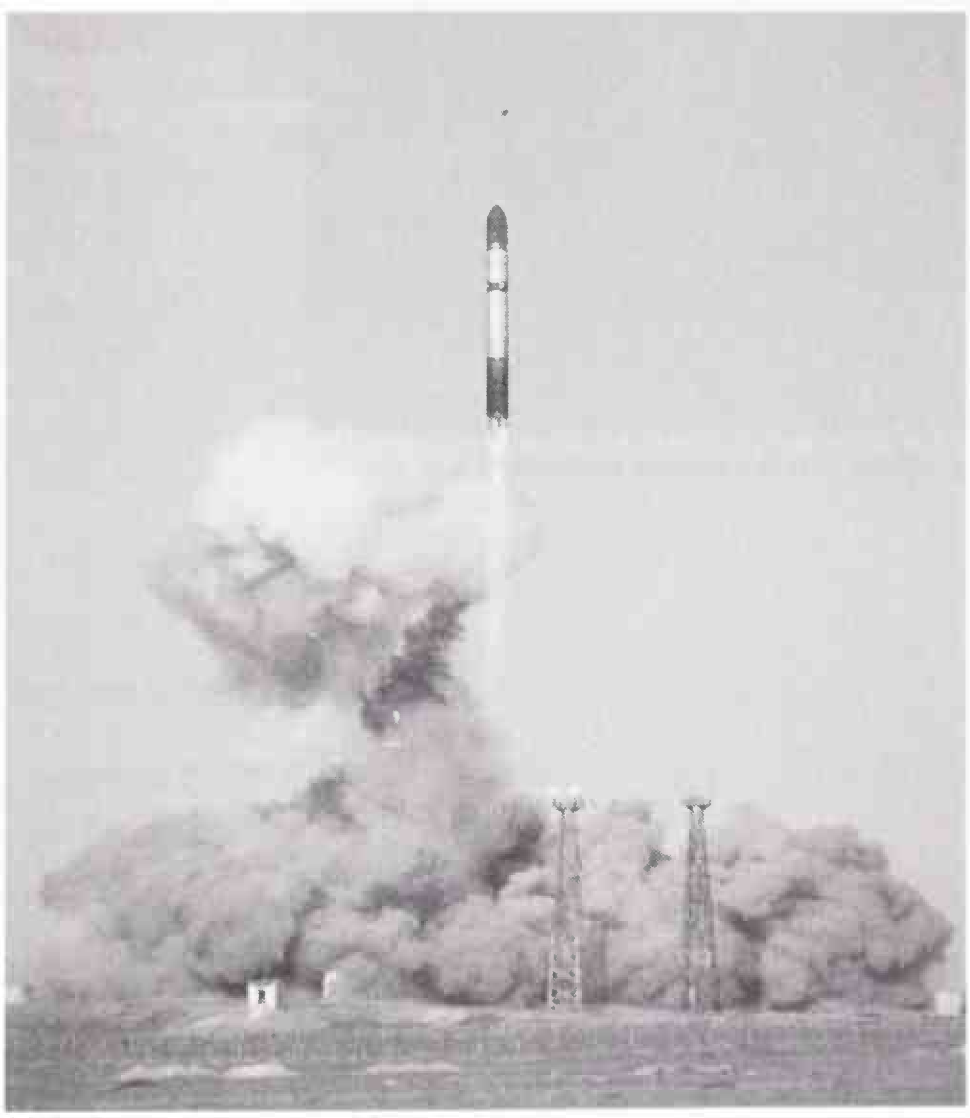


Courtesy ISC Kosmotras.

The EPM is loaded into the payload fairing. The lower cylindrical section is fixed, while the upper conical section is separable.

Courtesy ISC Kosmotras.

The Space Head Module, consisting of the third stage and payload fairing assembly, is loaded into a climate-controlled transporter-erector truck for the trip to the launch complex.



Courtesy ISC Kosmotras.

At launch, Dnepr is ejected from the silo by a black powder gas generator. The main engine ignites approximately 20 m (60 ft) above the ground.

Courtesy ISC Kosmotras

The transporter lowers the Space Head Module into place on the launch vehicle.

Vehicle Upgrade Plans

In the past, various upper stages have been proposed for Dnepr to improve the capability for launches of higher altitude LEO satellite constellations. However, the collapse of this market segment has ended pursuit of upgraded configurations such as the Dnepr-M. Kosmotras is studying missions in which the spacecraft is equipped with a kick motor to enable 950 kg (2100 lbs) to be sent to the Moon or 700 kg (1550 lbs) to Mars.

ISC Kosmotras has been in discussions with the government of the Orenburg oblast regarding the use of the Dombarovskiy strategic missile base for Dnepr launches. The base is one of three in Russia that still operates R-36M2 missiles. Launching Dnepr from the facility would simplify export issues and might be less expensive than using Baikonur in Kazakhstan.

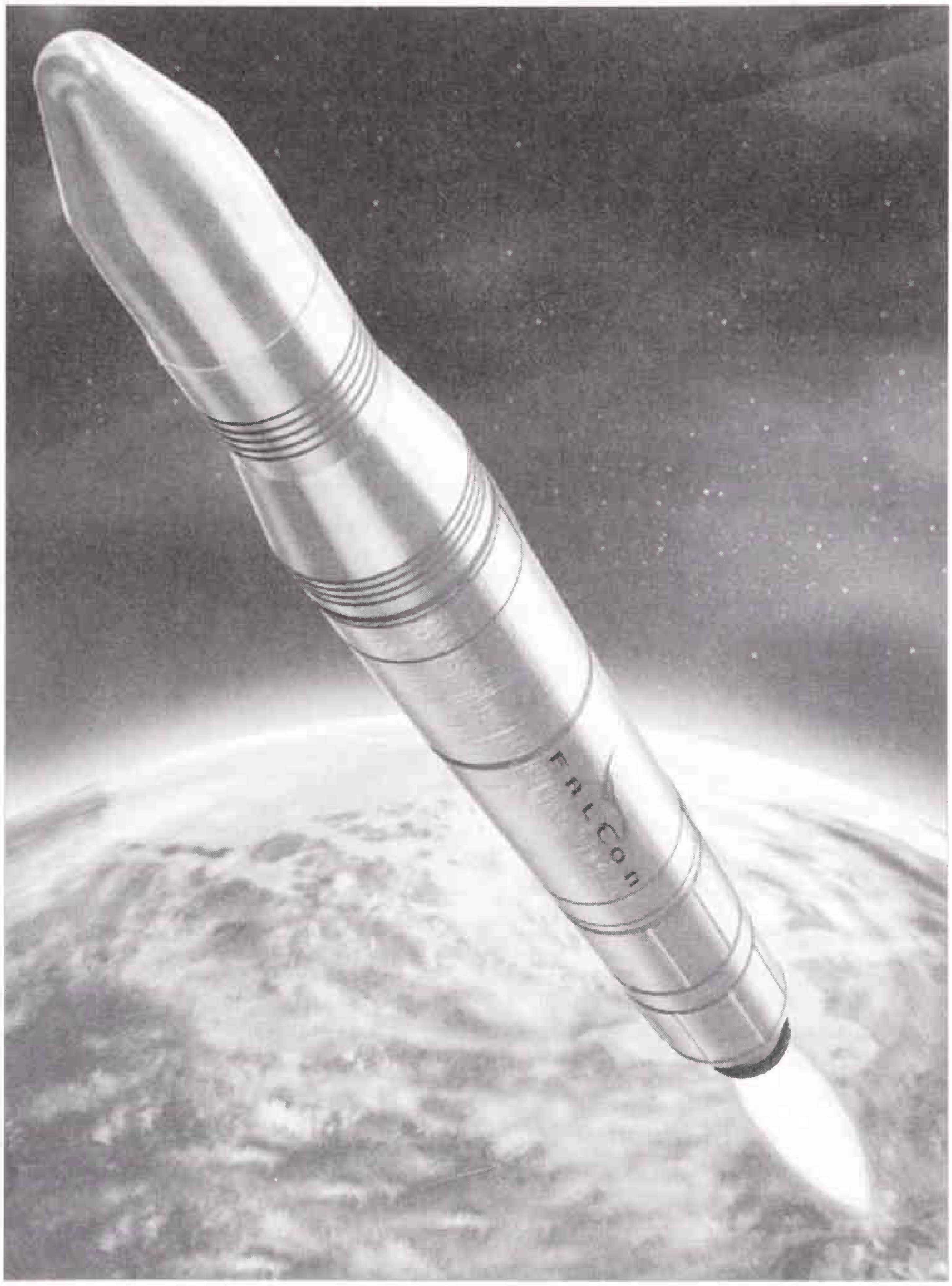
Vehicle History

Historical Summary

The R-36M (article number 15A14, NATO designation SS-18 Satan) was developed beginning in the late 1960s as a third-generation replacement for the R-36 (SS-9 Scarp) second-generation ICBM. The original R-36 was the predecessor to the Tsiklon launch vehicle. Like the R-36, the R-36M was fueled with storable N_2O_4/UDMH propellants and was designed and built by NPO Yuzhnoye. (In 1991, NPO Yuzhnoye split into two organizations; SDB Yuzhnoye is the design bureau, while PA Yuzhniy is the manufacturing organization.) The R-36M began test flights in the early 1970s and had entered service by 1975. It went through several modifications, until it reached its current configuration, the R-36M2 (15A18M, SS-18 Mod 4), which began test flights in 1986.

Commercialization of the R-36M2 was first discussed in 1990 and a variety of unsuccessful proposals were announced to fly various Russian micro-gravity and remote sensing payloads throughout the 1990s. No space launches of the RS-36M2 took place until after the formation of Moscow-based International Space Corporation Kosmotras in 1997. Kosmotras was successful in signing up its first Western customer, Surrey Satellite Technology Ltd., a builder of small satellites based in the United Kingdom. Surrey purchased launch services for the UoSat 12 minisatellite, which was orbited on the first Dnepr launch in April 1999.

FALCON



Falcon is a new small launch vehicle under development by Space Exploration Technologies.

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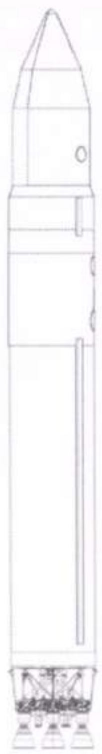
FALCON

Falcon I



Falcon V

GENERAL DESCRIPTION

Summary

The Falcon is a small liquid-fueled launch vehicle under development by Space Exploration Technologies (SpaceX). SpaceX is a start-up company founded in 2002 by Elon Musk, a successful Internet entrepreneur. Falcon is designed to be a highly reliable launch vehicle with a much lower cost than comparably sized vehicles. It is intended to be the first in a family of low-cost launch vehicles. The vehicle has two stages, both fueled by LOX and kerosene. The first stage can be recovered by parachute and reused.

Status

In development. First launch of Falcon I early 2004. First launch of Falcon V planned for mid-2005.

National Origin

United States

Key Organizations

Marketing Organizations	Space Exploration Technologies
Launch Service Provider	Space Exploration Technologies
Prime Contractor	Space Exploration Technologies

Primary Missions

Small spacecraft to LEO

Estimated Launch Price

Falcon I: $5.9 million plus range and payload specific costs (SpaceX, 2004)

Falcon V: $12 million plus range and payload specific costs (SpaceX, 2004)

Spaceports

Launch Site	Vandenberg AFB SLC 3W
Location	34.7°N, 120.6°W
Available Inclinations	60°–120°
Launch Site	Cape Canaveral AFS SLC 46
Location	28.5°N, 81.0°W
Available Inclinations	28.5°–50°

Performance Summary

	Falcon I	Falcon V
200 km (108 nmi) at 28°	668 kg (1472 lbm)	5040 kg (11,088 lbm)
200 km (108 nmi) at 90°	494 kg (1090 lbm)	3800 kg (8360 lbm)
Space Station orbit: 407 km (220 nmi), 51.6°	583 kg (1286 lbm)	4284 kg (9425 lbm)
Sun-synchronous orbit: 800 km (432 nmi), 98.6°	408 kg (898 lbm)	3173 kg (6980 lbm)
GTO: 108 × 35,786 km (100 nmi × 19,323), 28.5°	No capablility	1500 kg (3300 lbm)
Geostationary orbit	No capability	500 kg (1100 lbm)

Flight Record (through 31 December 2003)

Total Flights	0

Flight Rate

Planned at 2–3 per year initially, increasing to 4–6 per year.

Nomenclature

The name Falcon is a reference to the *Millennium Falcon* from the popular *Star Wars* movie franchise. Falcon is offered in two configurations, Falcon I and Falcon V. Falcon V uses the Falcon I architecture with a larger tank diameter and five Merlin engines for first-stage boost. Individual vehicles will be assigned a build number based on manufacturing sequence and a flight number based on manifest sequence.

Cost

The development of the Falcon launch vehicle is fully funded to first launch by private investment, primarily by founder Elon Musk, who made his fortune in Internet start-up firms.

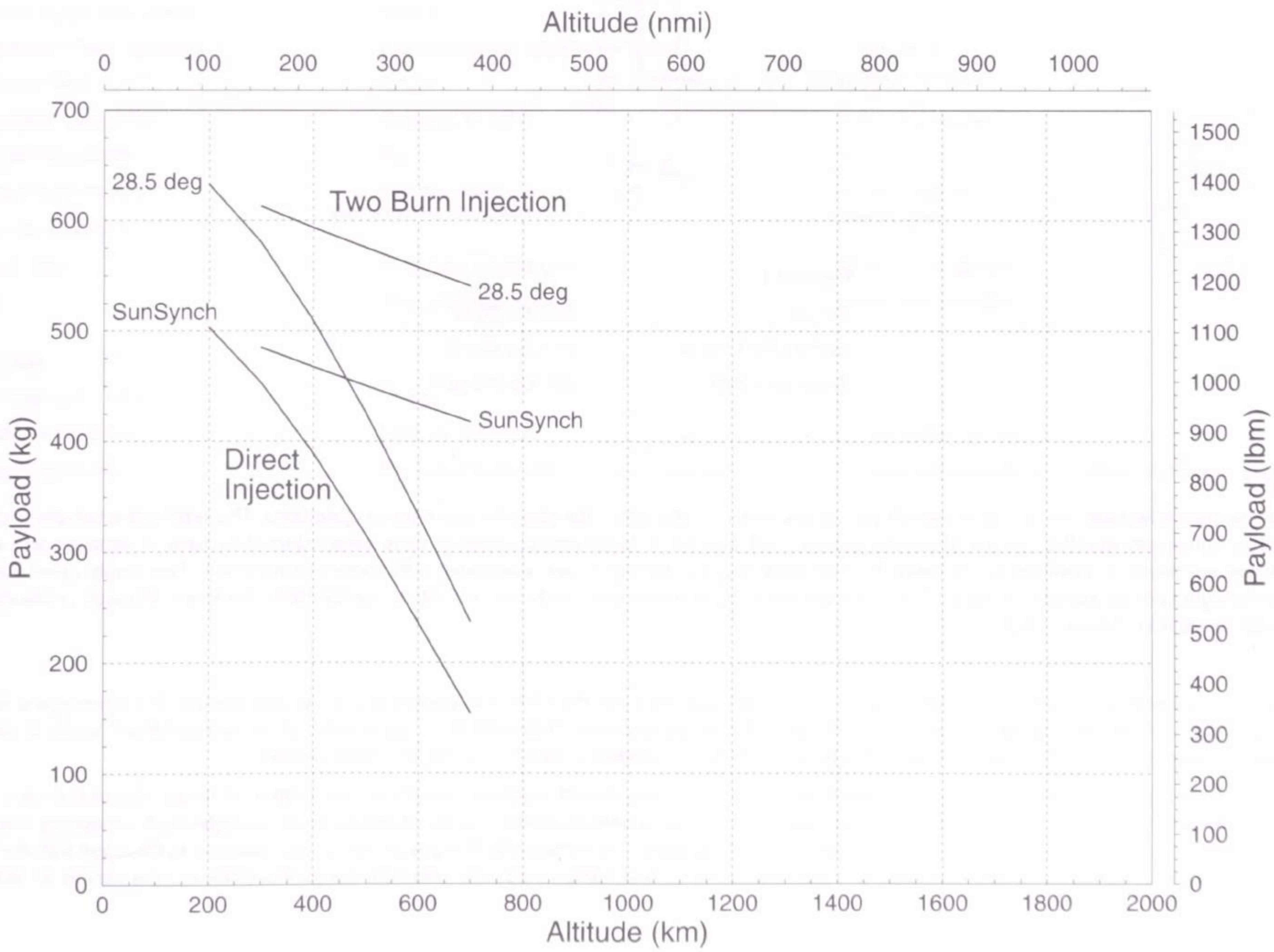
The cost of the development program has not been formally disclosed by SpaceX, but estimates of up to $50 million have been published in press reports. Falcon I launch services are priced at $5.9 million and Falcon V is offered for $12 million plus range costs and payload-specific costs.

Availability

Development of the Falcon is scheduled to be complete in late 2004. First launch will occur in late 2004. An initial flight rate of 2–3 flights per year is planned, increasing to 4–6 per year if there is sufficient market demand.

Performance

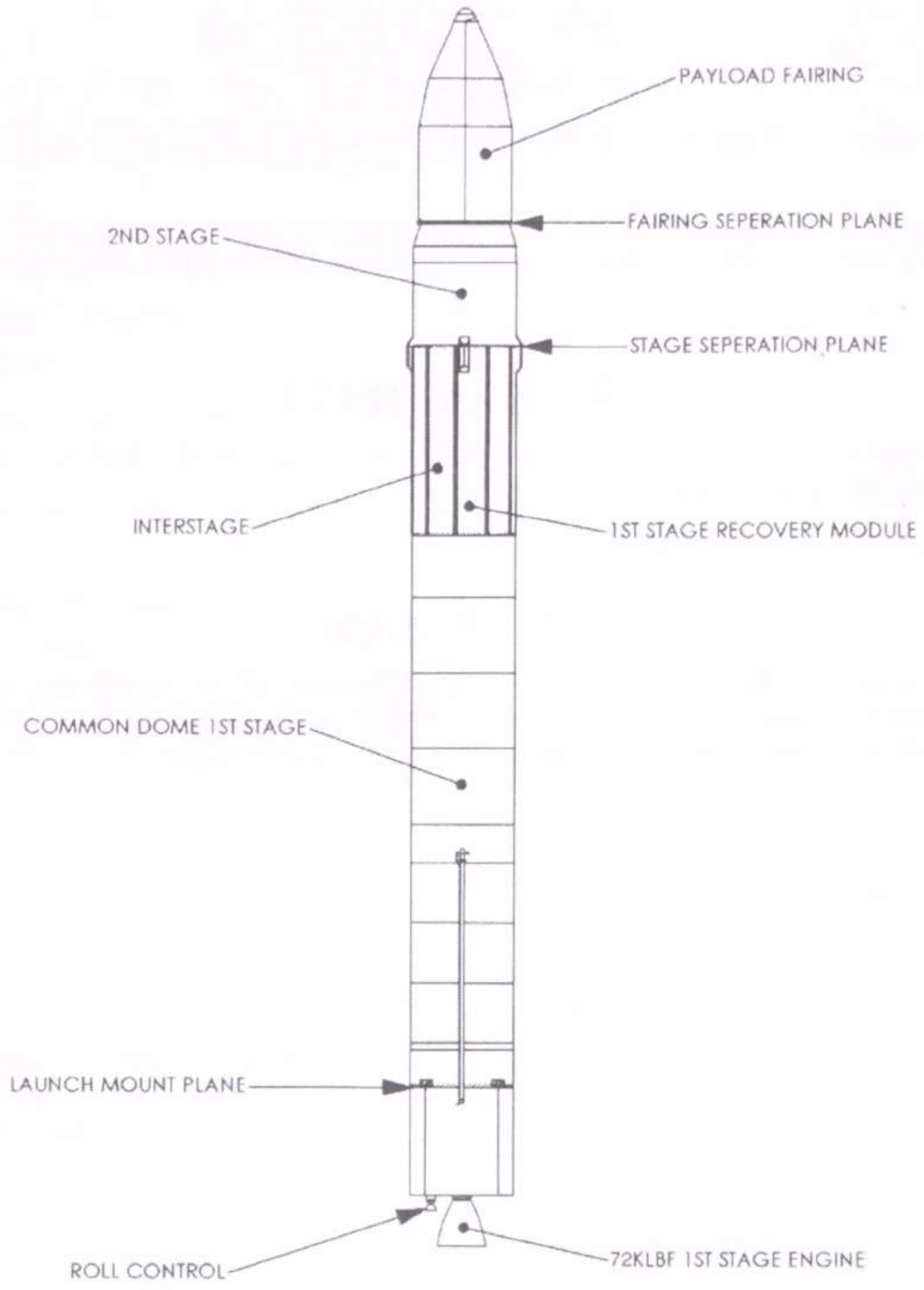
The Falcon launch vehicle accommodates low inclination orbits from its Cape Canaveral AFS launch site and high-inclination orbits from Vandenberg AFB. Two-impulse maneuvers are baselined, using injection into a transfer orbit followed by a second burn to circularize at apogee. Direct insertion performance is also shown because of its increased performance capability for very low altitude orbits.



Falcon I Performance to LEO

Vehicle Design

Overall Vehicle



Falcon I

Height	20.7 m (68 ft)
Gross Liftoff Mass	24 t (53 klbm)
Thrust at Liftoff	300 kN (72 klbf)

Stages

Many of the design features of the Falcon launch vehicle are driven by the desire for reliability and ease of operations. The vehicle is a two-stage design to minimize complexity and the number of staging events. LOX and RP-1 (kerosene) propellants were selected for their ease of handling and safety. Both stages are made of aluminum to eliminate the manufacturing and testing issues associated with composite materials. The empty vehicle is relatively lightweight and can be moved easily. The tanks are structurally stable and can be moved without pressurizing the tanks although pressurization is required to withstand loads in flight.

Stage 1

The first stage consists of a forward interstage, two common-bulkhead tanks with the LOX tank forward, and an engine section. The interstage is mounted on top of the LOX tank and accommodates a parachute for first stage recovery. Propellant tanks are lightly pressurized by helium, which is warmed by a heat exchanger in the turbine exhaust duct. The tanks are made of welded aluminum with spun-formed domes.

The Falcon Main Engine, Merlin, is a new turbopump-fed LOX/kerosene engine that develops 300 kN (72 klbf) sea level thrust. The injector is a pintle design and the thrust chamber is a silica–phenolic ablative chamber with a carbon–phenolic outer structural shell. A single-shaft turbopump driven by a fuel-rich gas generator delivers propellant to the chamber at high pressure. The turbopump and gas generator are mounted to the stage with the thrust chamber assembly gimbaled by hydraulic actuators. A four leg titanium thrust frame carries the load from the gimbal block into the vehicle aft skirt and the fuel tank.

Stage 2

The second stage is powered by a pressure-fed LOX/kerosene engine, Kestrel, using high-pressure heated helium as the tank pressurant. The engine develops 33 kN (7500 lbf) vacuum thrust and is gimbaled by electrical actuators. This engine has an ablative chamber similar to the first stage engine, except that it uses a radiation-cooled nozzle extension with an area ratio of 10:1–60:1.

Vehicle Design

Falcon I	Stage 1	Stage 2
Dimensions		
Length	?	?
Diameter	1.7 m (5.5 ft)	1.5 m (5 ft)
Mass		
Propellant Mass	6642 kg fuel +14,445 kg oxidizer (14,643 + 31,857 lbm)	1147 kg fuel +2707 kg oxidizer (2530 + 5970 lbm)
Inert Mass	1296 kg (2858 lbm)	360 kg (800 lbm)
Gross Mass	22,388 kg (49,358 lbm)	3745 kg (8260 lbm)
Propellant Mass Fraction	0.94	0.91
Structure		
Type	Monocoque	Monocoque
Material	Aluminum	Aluminum–Lithium
Propulsion		
Motor Designation	Merlin	Kestrel
Number of Motors	1	1
Propellant	LOX/Kerosene	LOX/Kerosene
Thrust	Sea Level: 320 kN (72 klbf) Vacuum: 352 kN (85 klbf)	Vacuum: 33kN (7500 lbf)
Isp	Sea Level: 261 s Vacuum: 306 s	Vacuum: 324 s
Chamber Pressure	47 bar (766 psia)	10.3 bar (150 psia)
Nozzle Expansion Ratio	14.5:1	60:1
Propellant Feed System	Gas-generator turbopump	Pressure-fed
Mixture Ratio (O/F)	2.17:1	2.35:1
Throttling Capability	Optional to 60%	Partial blowdown
Restart Capability	No	Yes
Tank Pressurization	Heated helium	Heated helium
Attitude Control		
Pitch, Yaw	Hydraulic nozzle gimbal	EMA nozzle gimbal
Roll	Gas-generator output	Warm-gas thrusters
Staging		
Nominal Burn Time	164 s	372 s
Shutdown Process	Burn to depletion	Command shutdown
Stage Separation	Explosive bolts with hydraulic pushers	Payload separation via lightband system

VEHICLE DESIGN

Attitude Control System

First-stage attitude control is provided by a nozzle gimbal system driven by a hydraulic actuators using the high-pressure kerosene from the turbopump, which is then recycled into the low-pressure inlet of the turbopump. This eliminates the need for a separate hydraulic system. The turbine exhaust gas nozzle is gimbaled to provide roll control torque. Second-stage attitude control is provided by gimbaling the main engine using electromechanical actuators. Roll control is provided by separate warm-gas thrusters.

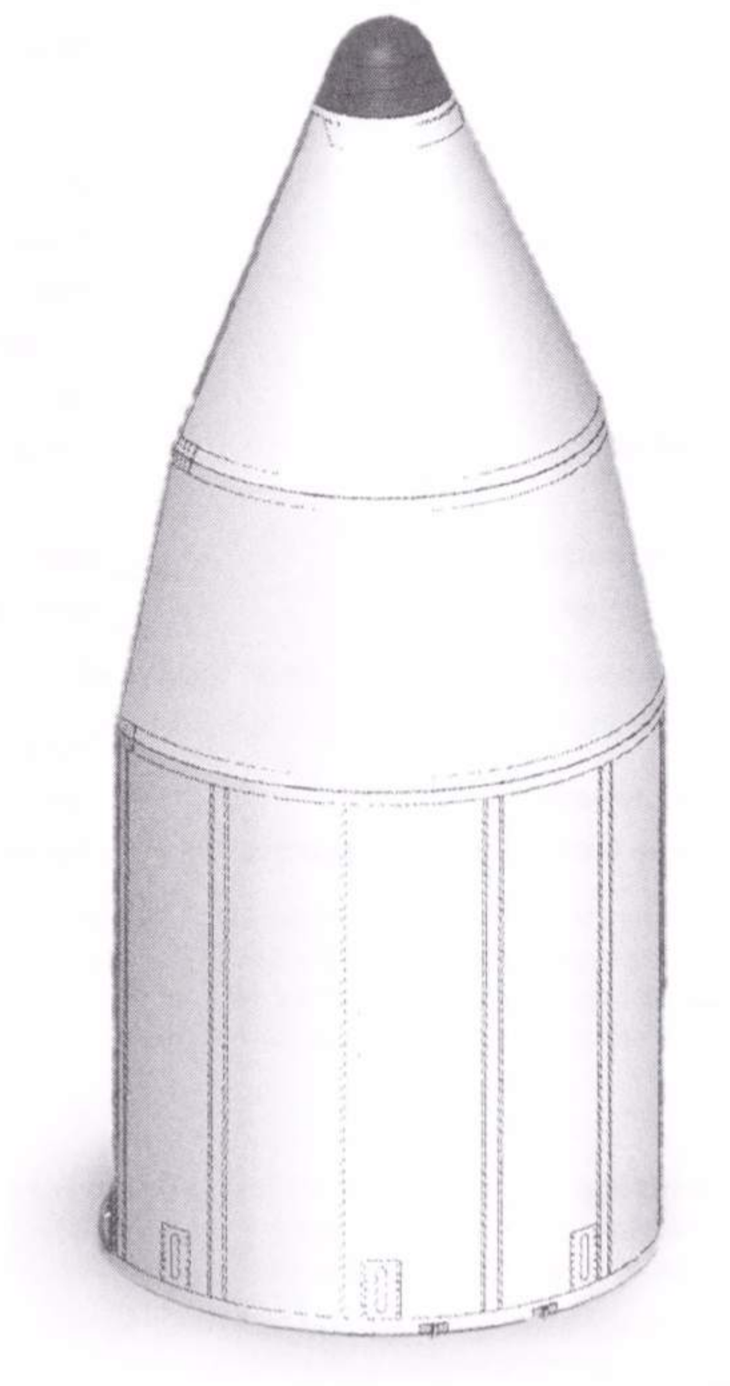
Avionics

The first-stage engine computer controls the engine startup and monitors the gas generator, turbopump and main engine parameters. It also drives and controls the two servovalves of the gimbal system and collects telemetry in the engine bay. The engine computer is connected to the flight computer in the second stage avionics bay via ethernet.

The second-stage avionics include the redundant flight computers and IMUs. The flight computers have analog and digital IO and provide an interface to the payload via ethernet. Ethernet was selected to eliminate the workmanship issues associated with large cable bundles and connections. A GPS receiver provides navigational information and can compensate for wind and IMU drift error. The avionics system also includes an S-band telemetry system, a video downlink, and a C-band transponder. The flight termination system (FTS) makes use of all range approved heritage hardware and includes redundant batteries, command receivers, and safe and arm systems.

Payload telemetry will be provided while the spacecraft is integrated on the launcher at the launch site and is optional during flight. One channel of payload flight telemetry with bit rates up to 38K is planned using an RS-232 or RS-422 connection. Power will be provided to the payload while integrated with the launcher at the launch site, and is available as an option during flight. A separation signal will be provided.

Payload Fairing



Length	3.5 m (11.3 ft)
Primary Diameter	1.5 m (5 ft)
Mass	?
Sections	2
Structure	Skin-stringer
Material	Aluminum

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	1370 mm (54 in.)
Maximum Cylinder Length	1270 mm (50 in.)
Maximum Cone Length	2794 mm (60 in.)
Payload Adapter Interface Diameter	985 mm (38.8 in.)
Payload Integration	
Nominal Mission Schedule Begins	T–7 days
Environment	
Maximum Axial Load	6.4 *g*
Maximum Lateral Load	2 *g*
Maximum Acoustic Level	122 dB
Overall Sound Pressure Level	131 dB
Maximum Flight Shock	1400 ft
Maximum Dynamic Pressure on Fairing	33.5 kPa (700 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Cleanliness Level in Fairing	Class 100,000 purge
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±10 km (18 nmi) at perigee ±20 km (36 nmi) at apogee
Attitude Accuracy (3 sigma)	0.5°
Nominal Payload Separation Rate	0.1–0.9 m/s (1–3 ft/s)
Nominal Payload Tip Off Rate	< 1 deg/s
Deployment Rotation Rate Available	1 rpm
Loiter Duration in Orbit	1 h
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Falcon will accommodate comanifest payloads
Auxiliary Payloads	Falcon will accommodate secondary and piggyback payloads

Production and Launch Operations

Production

The Falcon vehicle stages will be produced at SpaceX facilities in El Segundo, California. Unlike most launch vehicle manufacturers, SpaceX is designing and producing its own engines. The vehicle will be integrated as three primary assemblies: the first-stage assembly, the second-stage assembly, and the payload accommodations. The first- and second-stage assemblies will be processed horizontally at the SpaceX facility.

Launch Operations

Two launch sites have been selected for Falcon. At Vandenberg AFB, SpaceX will use the former Atlas pad at SLC-3W. Launches at Cape Canaveral AFS will be conducted from SLC-46, the commercial pad operated by the Florida Space Authority. The Falcon launch vehicle will be capable of launching from multiple launch sites because of the low level of infrastructure required for launch. SpaceX is investigating the feasibility of launching from Kwajalein for very low inclination launches. Required launch site infrastructure includes portable processing facilities, a frame or stool over a flame bucket, a blockhouse or mobile communications system, a payload processing facility, and office space.

Falcon launch operations are designed to be simple—achieving operations as close to "ship and shoot" as possible. SpaceX plans to be able to launch within one week of payload arrival at the launch site—with an ultimate goal of launching within a few hours. This requires limiting payload processing time and efficient on-pad operations.

The two stage assemblies will be shipped separately to the launch site. Once at the site, the two assemblies will be processed horizontally. Flight termination system connections will be made and system tests performed. The first stage will be stacked onto the launch frame. A dress rehearsal will verify all major first stage functions. The second stage will then be stacked, and mechanical and electrical connections will be made, including final FTS integration. Post inspection, a second rehearsal will verify first and second stage functionality.

In parallel with these operations, the payload will be integrated to the payload adapter and encapsulated in the fairing in a vertical operation. This assembly will be moved to the launch stack and lifted onto the integrated first and second stage. Mechanical and electrical connections will be made and interface testing performed. After fueling the vehicle, a final systems check will verify full vehicle functionality.

SpaceX plans to recover the first stage after launch. This will be accomplished with the deployment of a hypersonic drogue with subsequent parachute deployment at 3 km (10,000 ft). Stage 1 is expected to land within 800 km (430 nmi) downrange. A salvage ship will be deployed and waiting near the intended impact zone. The stage is expected to float for less than 24 h while the ship locates it using a beacon mounted on the stage. The ship will then tow it to a harbor for pick up and ultimate inspection and refurbishment.

Vehicle Upgrade Plans

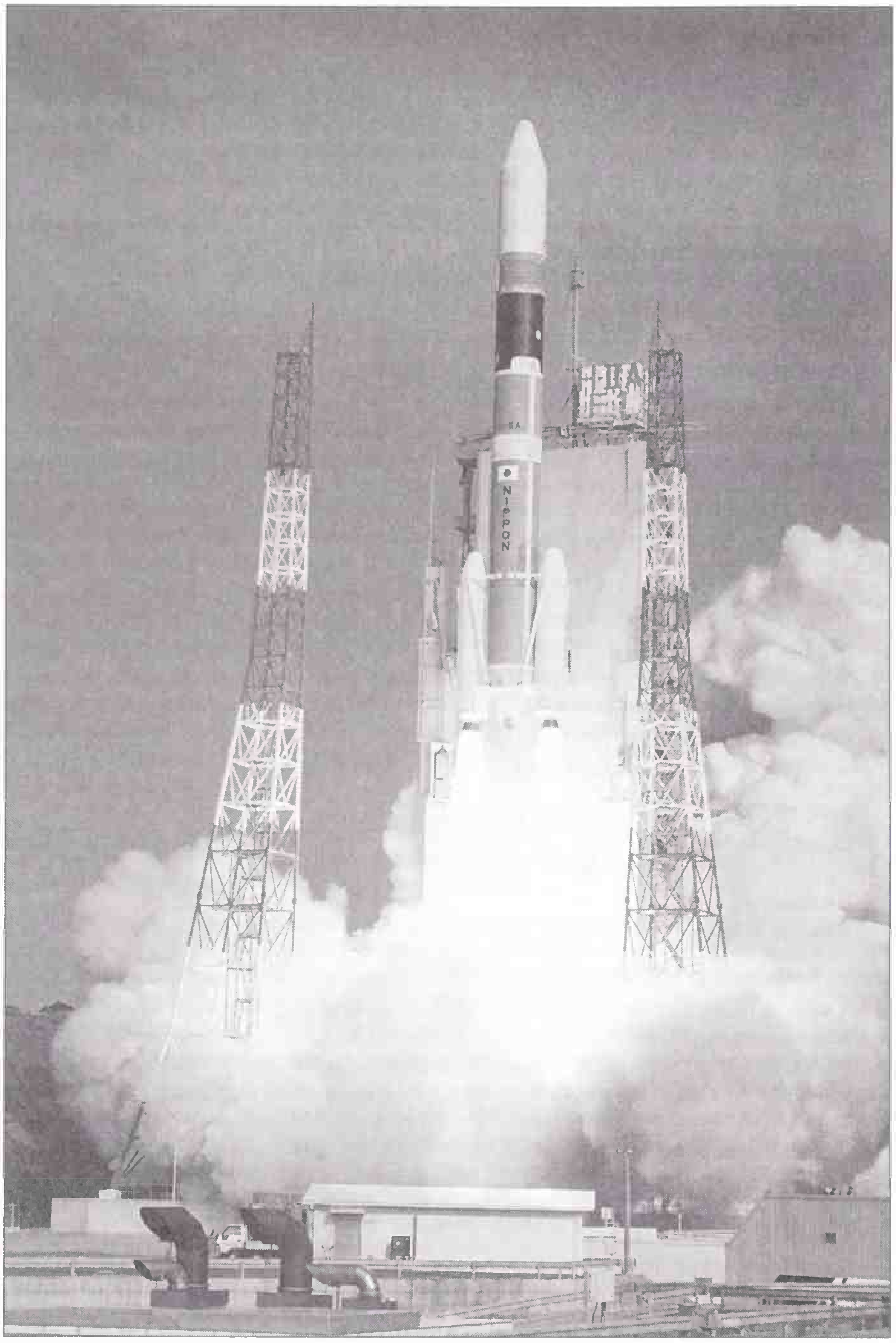
SpaceX plans to use the first and second stages of Falcon as the second and third stages of a much larger vehicle. This vehicle, in its standard configuration, will deliver on the order of 24.5 t (40 klbm) to LEO. Development should support a launch in 2008.

For missions requiring low inclination orbits, SpaceX is considering an equatorial launch site.

Vehicle History

SpaceX was founded by Elon Musk, who became wealthy as an Internet entrepreneur. A native of South Africa, Musk dropped out of Stanford in 1995 to found Zip2, which was sold in 1999 for $300 million. His second company, PayPal, was sold to eBay for $1.5 billion in 2002. Early in 2002, Musk assembled a team of people experienced with the space launch industry to study whether it was possible to build a low-cost launch vehicle in a developed country with high labor costs. The results of the study were positive, and so Musk founded Space Exploration Technologies (SpaceX) in June 2002. He hired a few dozen managers and engineers from established aerospace companies to build the Falcon launch vehicle. Development has progressed rapidly, with first launch planned in 2004.

H-IIA



Courtesy JAXA.

The H-IIA, shown here in its first launch, is Japan's primary launch system. While it looks similar to the H-II from which it is derived, it shares few parts in common, having been completely redesigned to reduce costs.

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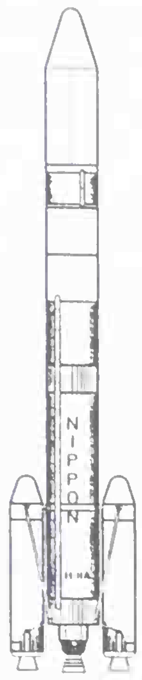
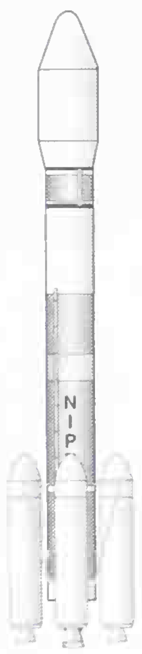
H-IIA

General Description



H-IIA 202



H-IIA 204

Summary

Japan's H-IIA is derived from the earlier H-II, with modifications to reduce launch costs and increase performance. The cost has been reduced by modifying the vehicle for easier manufacturing and launch operations, increasing the launch rate, and purchasing some components from lower-cost foreign suppliers. The H-IIA family includes several variants, each consisting of two cryogenic core stages with different combinations of strap-on boosters.

Status

Operational: First launch in 2001 for 202 configuration. First launch in 2002 for 2024 configuration. H-IIA 204 in development. First launch planned for 2006.

Origin

Japan

Key Organizations

Marketing Organization	Rocket System Corporation/Mitsubishi Heavy Industries
Launch Service Provider	JAXA/Mitsubishi Heavy Industries
Prime Contractor	Rocket System Corporation/Mitsubishi Heavy Industries

Primary Missions

Intermediate GTO and LEO payloads

Estimated Launch Price

The following costs were published by NASDA, and represent launch costs for government missions, not necessarily prices that would be charged for commercial launches. They are based on an assumed exchange rate of 120 yen to the dollar.

H-IIA 202: 8.5 billion yen ($70 million)

H-IIA 2022: 9.0 billion yen ($75 million)

H-IIA 2024: 9.5 billion yen ($80 million)

H-IIA 204: 9.9 billion yen ($83 million)

Spaceport

Launch Sites	Yoshinobu Launch Complex at Tanegashima Space Center
Location	30.4°N, 131.0°E
Available Inclinations	28.5–100 deg

Performance Summary

A single-payload 4-m fairing is assumed for the performance given below, unless otherwise noted. A standard 100-kg adapter is accounted for. Performance for polar orbits is higher for launches in winter than in the summer. Winter values are shown. Performance shown reflects full-length first-stage nozzle. Performance on initial launches with short nozzle is slightly lower.

300 km (162 nmi), 30.4 deg	H2A202: 9940 kg (21,900 lbm) H2A2022: 10,740 kg (23,675 lbm) H2A2024: 11,730 kg (25,860 lbm)
185 km (100 nmi), 90 deg	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	H2A202: 8400 kg (18,500 lbm) H2A2022: 9350 kg (20,600 lbm) H2A2024: 10,240 kg (22,590 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	5-m payload fairings H2A202: 4350 kg (9590 lbm) H2A2022: 4940 kg (10,890 lbm) H2A2024: 5270 kg (11,600 lbm)
GTO: 250×36,226 km (135×19,560 nmi), 28.5 deg	H2A202: 4100 kg (9040 lbm) H2A2022: 4500 kg (9920 lbm) H2A2024: 5000 kg (11,025 lbm) H2A204: 5800 kg (12,800 lbm)
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

	H-IIA	N and H Family Total
Total Orbital Flights	6	37
Launch Vehicle Successes	5	33
Launch Vehicle Partial Failures	0	1
Launch Vehicle Failures	1	3

Flight Rate

0–3 per year demonstrated, 6–8 per year capability planned

Nomenclature

H-IIA vehicle discrimination codes are used to describe the configuration of strap-on boosters.

Example **H2A2024-4S**

- Number of fairing compartments (S = single payload, D = dual payload)
- Fairing diameter in meters (4 or 5). A 4/5D fairing is also available with one 4-m and one 5-m compartment
- Number of SSB supplemental boosters (0, 2, or 4). Digit is omitted if no SSBs are used.
- Number of SRB-A boosters (2 or 4)
- Number of LRBs (0 or 1)
- Number of core stages (1 or 2). All launches will use both the first and second stages
- Designates the basic H-IIA vehicle type, as distinguished from the original H-II

Cost

A primary goal of the H-IIA program is to develop the original H-II vehicle into a lower-cost launch system. The development cost of the original H-II was approximately 270 billion yen (roughly $2.2 billion at 2002 exchange rates). Of this, roughly 90 billion yen ($730 million) was spent on development of the LE-7 engine. According to NASDA*, it typically cost about 19 billion yen ($155 million) to produce and launch each H-II. The most expensive mission, flight 3F, cost 19.5 billion yen ($158 million) because of an additional pair of small solid boosters. The least expensive mission was 8F, costing only 14 billion yen ($114 million) because it used the less expensive version of the second stage developed for the H-IIA. Japan's Transport Ministry and Meteorological Agency paid NASDA 10 billion yen ($81 million) to launch the first MT-Sat on this flight in November 1999.

The H-IIA development program was initially budgeted at 90 billion yen ($730 million), but is estimated to have cost closer to 120 billion yen ($976 million) because of a variety of delays and technical problems. JAXA's goal is to reduce per flight costs of the basic H2A202 vehicle to less than 8.5 billion yen (about $69 million). The first H-IIA flight actually cost around 9 billion yen ($73 million), but this was due in part to the additional costs associated with a first test launch. The second and third launches, each using the more powerful 2024 configuration, cost 10.6 billion yen ($86 million) and 10.2 billion yen ($83 million) respectively. In 2001 Japan's Ministry of Land, Infrastructure, and Transport signed a $72 million contract to launch a replacement MT-Sat on an H-IIA. In 1996 Hughes ordered 10 launches with 10 options on H-IIA vehicles. When the contract was cancelled in 2000, Japanese newspapers reported that the contract had been worth 103 billion yen ($836 million).

Availability

The first H-IIA test launch, using the 202 configuration, was performed in August 2001. The 2024 configuration was first launched in a second test launch in February 2002. After two successful test flights, H-IIA is consider to be operational.

The H-IIA is used primarily by the Japanese government, which plans a flight rate of three government missions per year. It is also marketed internationally to commercial customers by Rocket System Corporation (RSC), a consortium of the Japanese companies involved in the production of the H-IIA. RSC has been responsible for building vehicles and for marketing and operating them for commercial users since the creation of the H-II. However, in 2002 the Space Activities Commission signed an agreement with Mitsubishi Heavy Industries giving it control of the launch and marketing operations of H-IIA as part of a privatization process. Mitsubishi will take over in 2006 with the ninth flight.

Historically, launches at Tanegashima have been limited to two periods of roughly six weeks each per year, starting in January and August, because of concerns by local fishermen that launches reduce their catch. In the past this limited launches to two per year and imposed strict schedule constraints. To achieve higher flight rates and reduce launch window restrictions for the H-IIA and smaller rockets such as the J-I and M-V, the Japanese government negotiated with local governments and the fishing unions to double the duration of the launch windows. The new periods cover 1 January through 28 February, and 22 July through 30 September. In addition, 60 days from June through July and November through December are available as launch periods for high-priority launches such as planetary or international missions. JAXA also expanded its vehicle assembly building and built a second launch pad at the Yoshinobu Launch Complex. The combination of longer launch periods, enhanced facilities, and improvements in the H-IIA design for faster launch processing make it possible to launch up to 6–8 vehicles per year.

Performance

The H-IIA is a flexible vehicle that can be used effectively for both GTO and LEO missions. The performance of the H-IIA can be tailored to the mission requirements by adding either two or four small solid strap-on boosters (SSBs). Each pair of SSBs increases GTO performance approximately 400–500 kg (900–1100 lbm). The typical GTO inclination for H-IIA launches is 28.5 deg, comparable to launches from Cape Canaveral. For LEO missions the H-IIA can deliver payloads to inclinations as high as sun-synchronous orbit. However, polar inclinations cannot be reached directly because of range safety concerns near Tanegashima. Instead, the launch vehicle must launch southeast, and then use a dogleg maneuver toward the southwest to reach the correct flight azimuth. During most of the year, the prevailing winds blow out to sea, but this is not the case in the summer. To prevent debris being blown back toward the launch site in the event of a failure, the H-IIA must make a wider turn, thus reducing performance for summer launches more than in winter. Launches to very low altitudes at 30.4 deg cannot be monitored continuously by tracking stations. Performance for this inclination is shown in the following charts assuming that this is acceptable to both the customer and JAXA.

The 4S fairing is assumed for the performance shown unless otherwise noted. Changing to a 5-m fairing (5S) reduces GTO performance roughly 300 kg (650 lbm). In the case of dual payload missions, the mass of the lower fairing assembly must be subtracted from the capability shown in the performance charts. A 100 kg (220 lbm) standard adapter is accounted for. If a different adapter is required, performance should be adjusted accordingly. Adapter masses range from 40 to 300 kg (90 to 650 lbm) depending on spacecraft size, mass, and interface requirements. The probability of command shutdown (PCS), which corresponds to the probability having sufficient performance to meet the specified orbit tolerance, is typically 99.7% or greater.

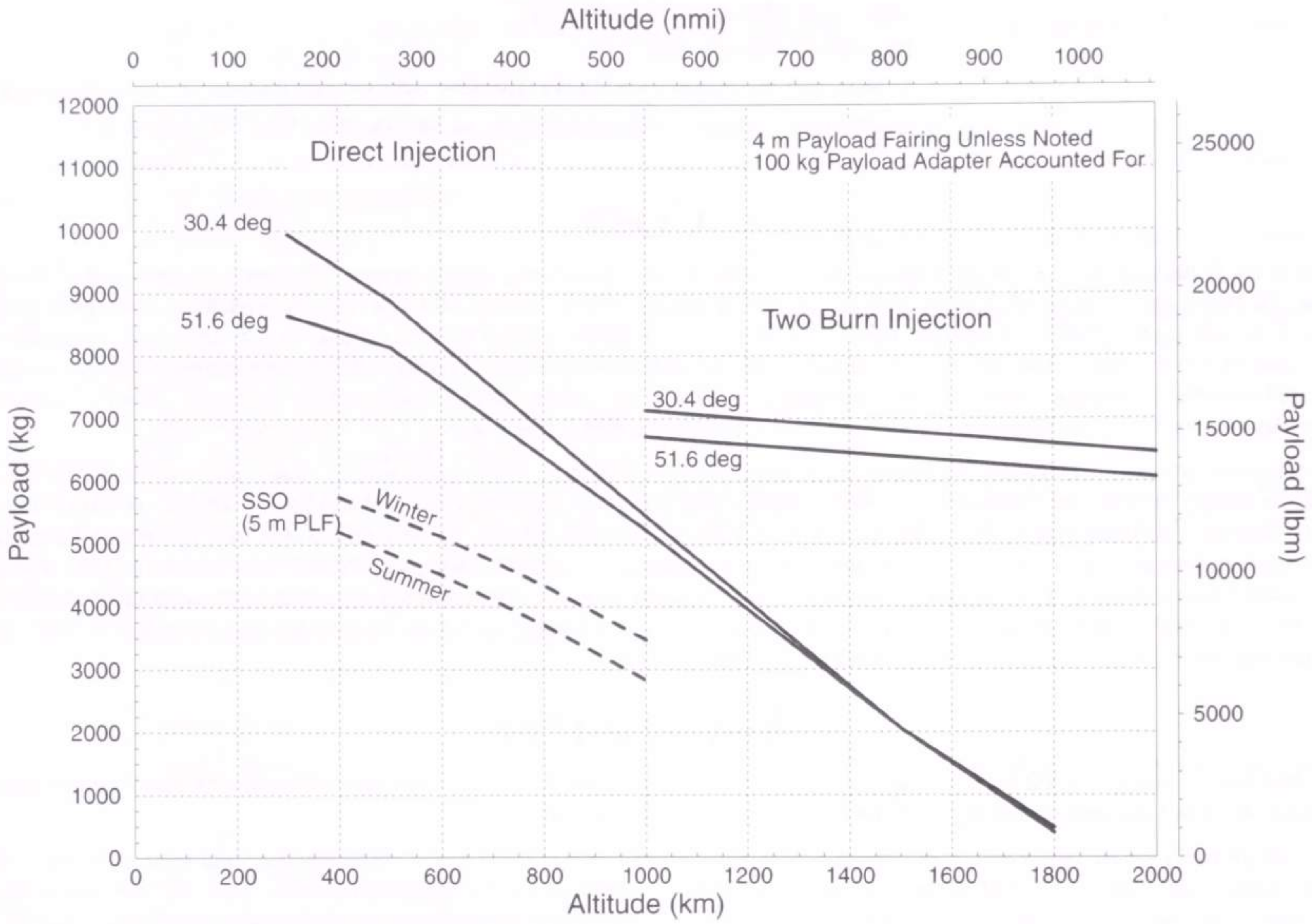
Initial flights of H-IIA have not been able to use a planned nozzle extension on the first stage engine and therefore performance has been lower than planned. (See Vehicle Design, Stage 1, for more information). The following performance graphs reflect capability with the long nozzle, which should be available by 2004. Performance to GTO with and without the nozzle extension is shown below for comparison purposes. Performance to the 800 km (432 nmi) sun-synchronous reference orbit is typically 100–200 kg (220–440 lbm) lower with the short nozzle.

*JAXA, the Japanese Aerospace Exploration Agency, was formed 1 October 2003 by the merger of the Institute of Space and Astronautical Science (ISAS), National Aerospace Laboratory (NAL), and National Space Development Agency of Japan (NASDA). References to historical events will preserve the original organization's name. All ongoing and new references will be to JAXA.

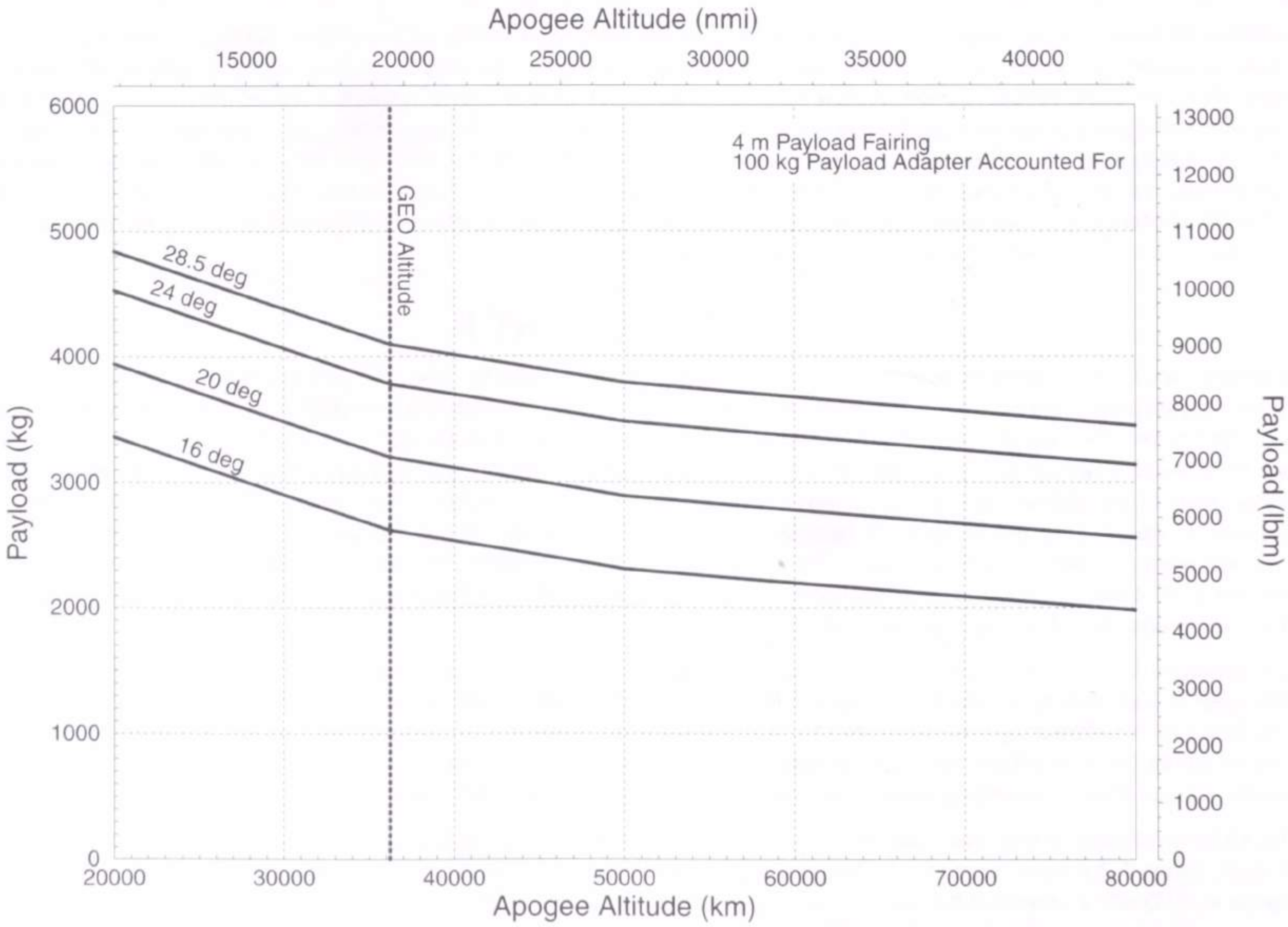
Performance

Performance to 28.5 deg GTO

	Short Nozzle	Long Nozzle
H-IIA 202	3700 kg (8157 lbm)	4100 kg (9039 lbm)
H-IIA 2022	4100 kg (9039 lbm)	4500 kg (9921 lbm)
H-IIA 2024	4600 kg (10,141 lbm)	5000 kg (11,023 lbm)
H-IIA 204		5800 kg (12,787 lbm)

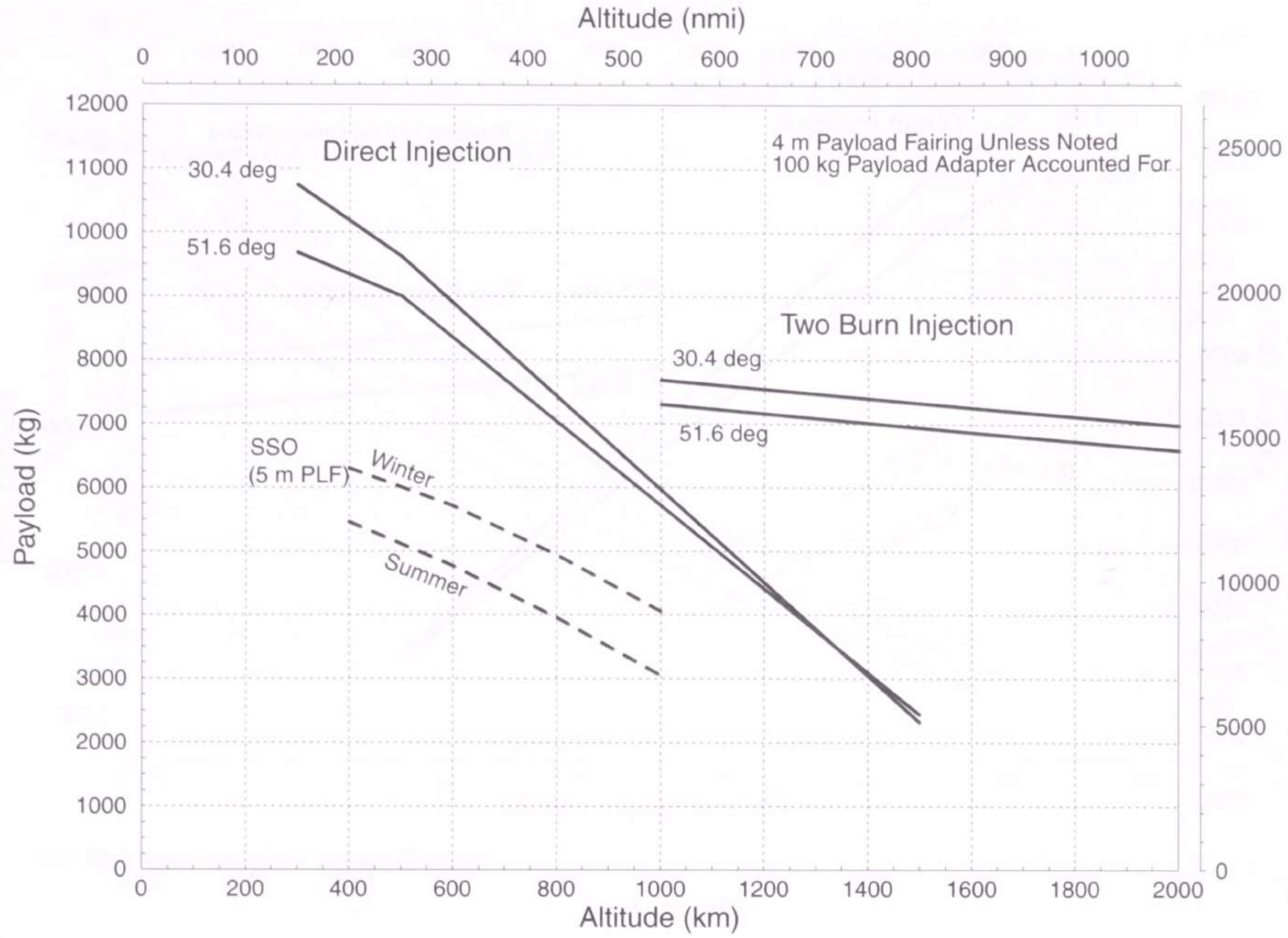


H2A202: LEO Performance

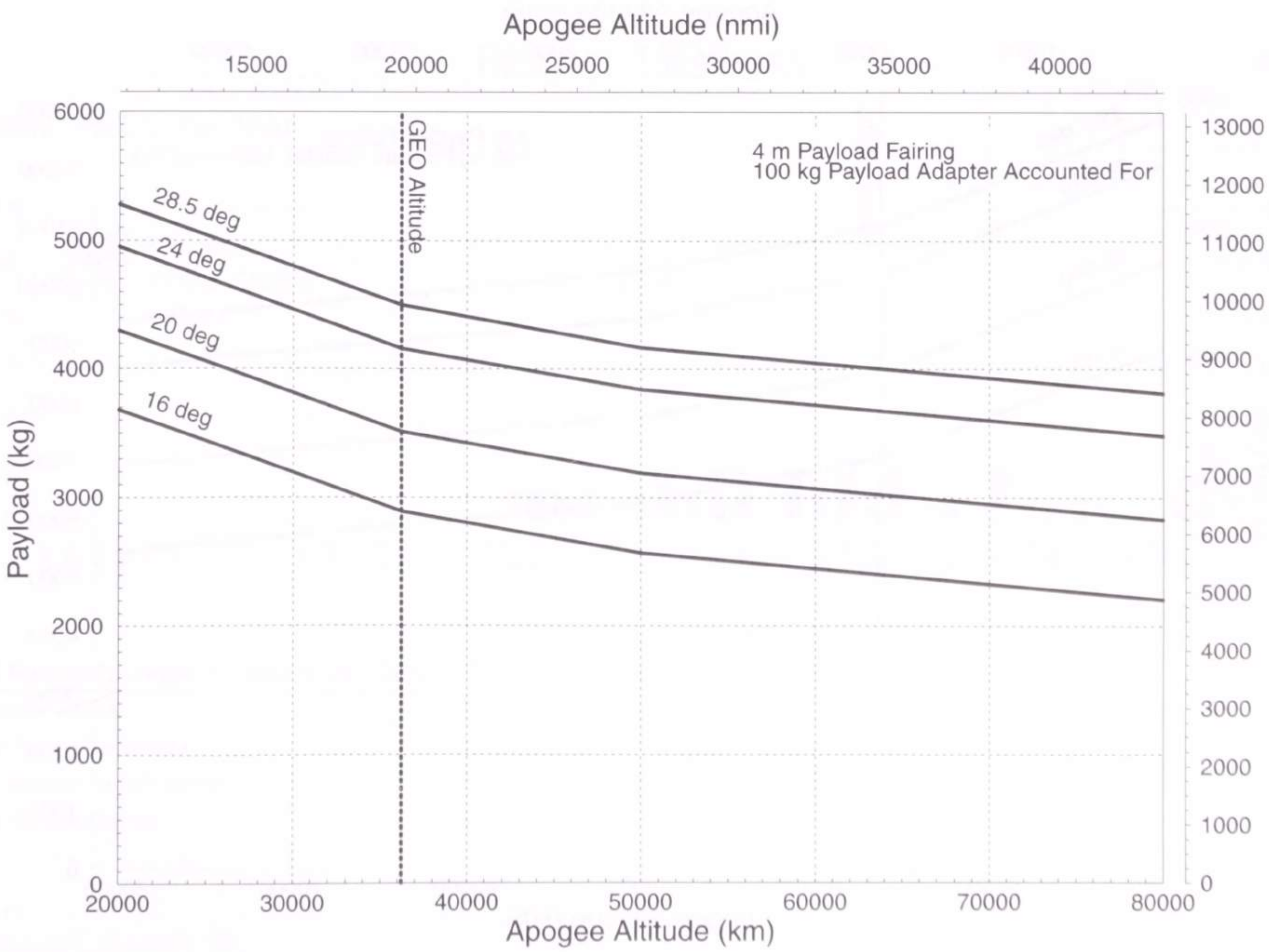


H2A202: GTO Performance

Performance

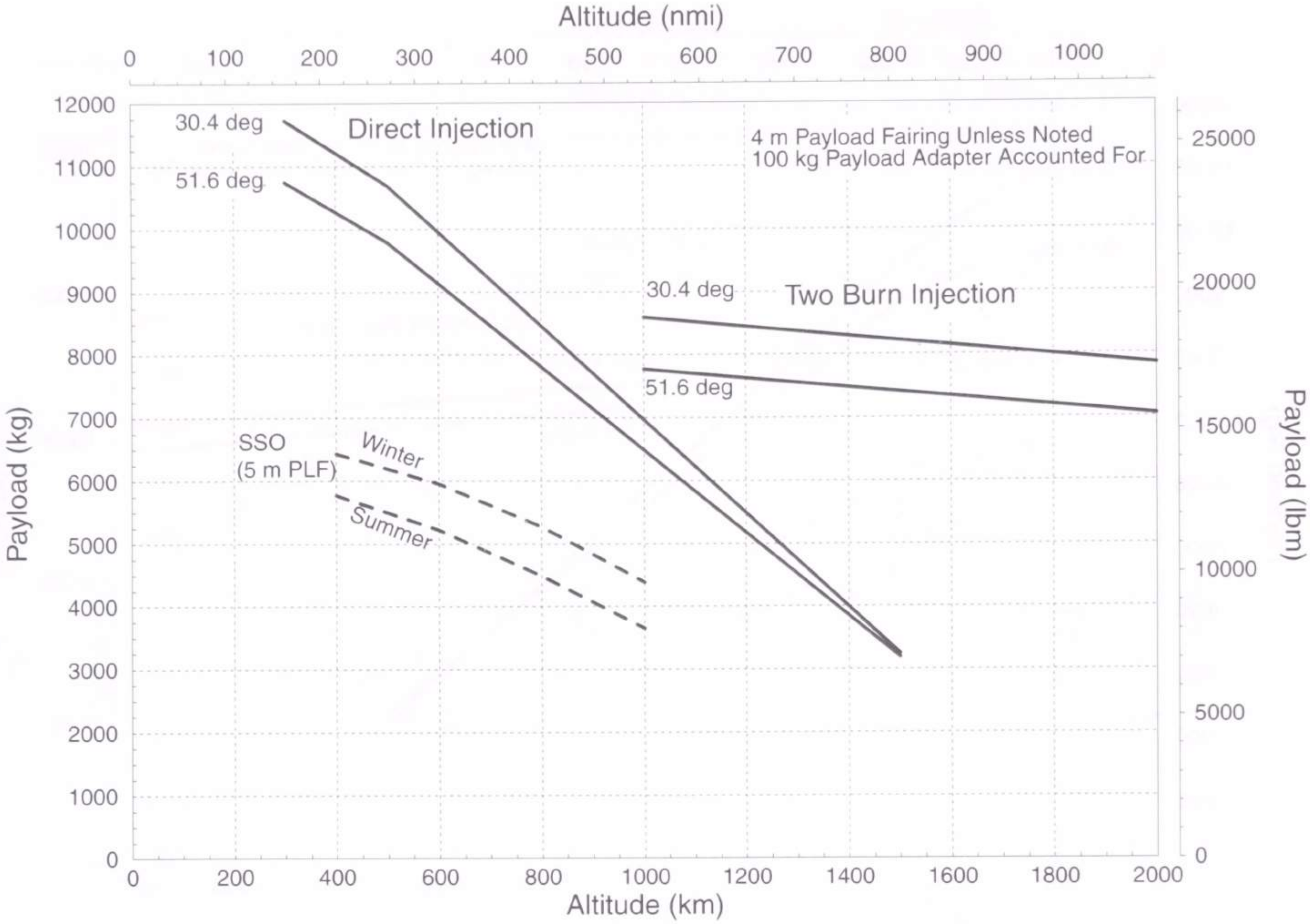


H2A2022: LEO Performance

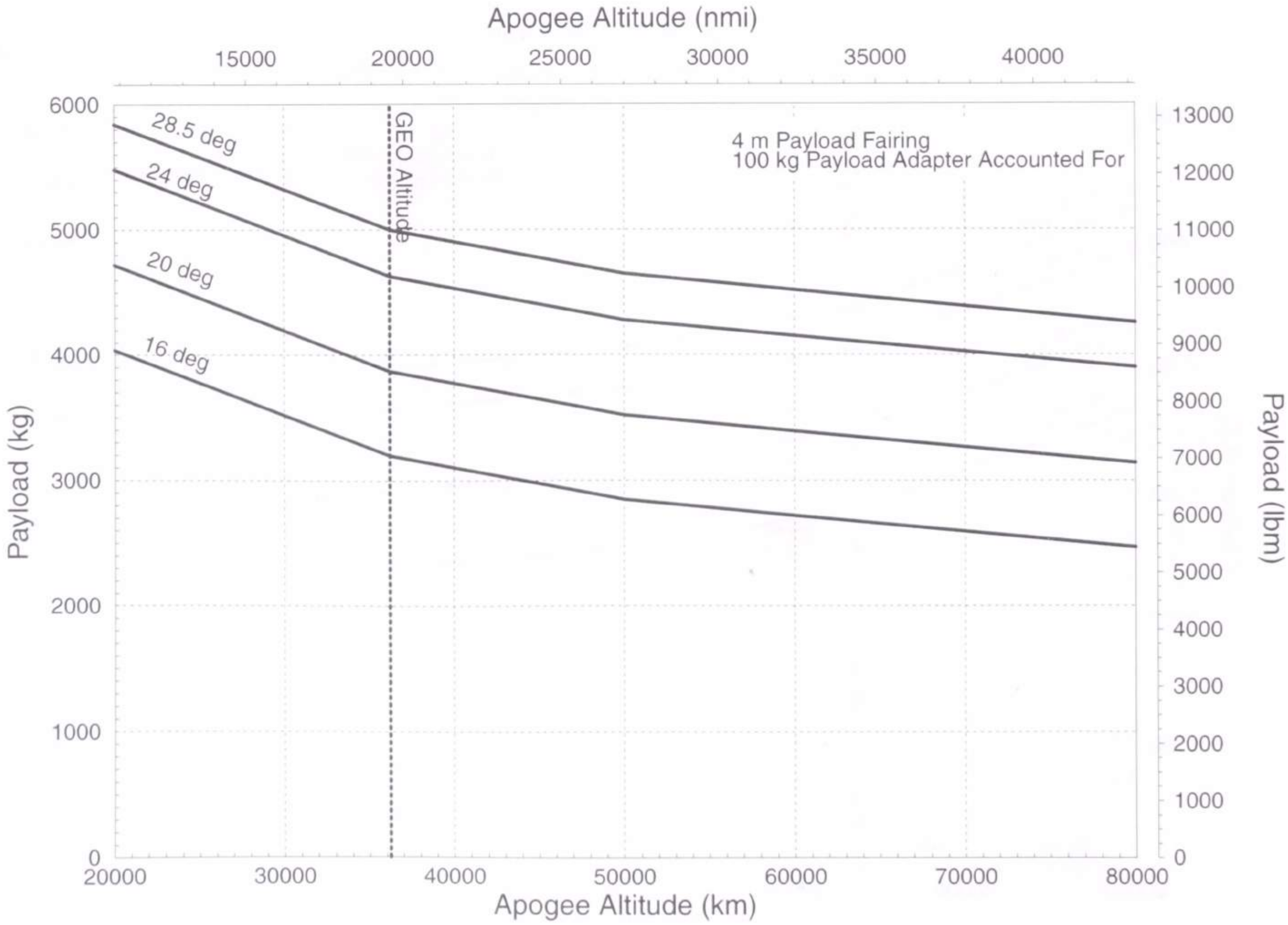


H2A2022: GTO Performance

Performance

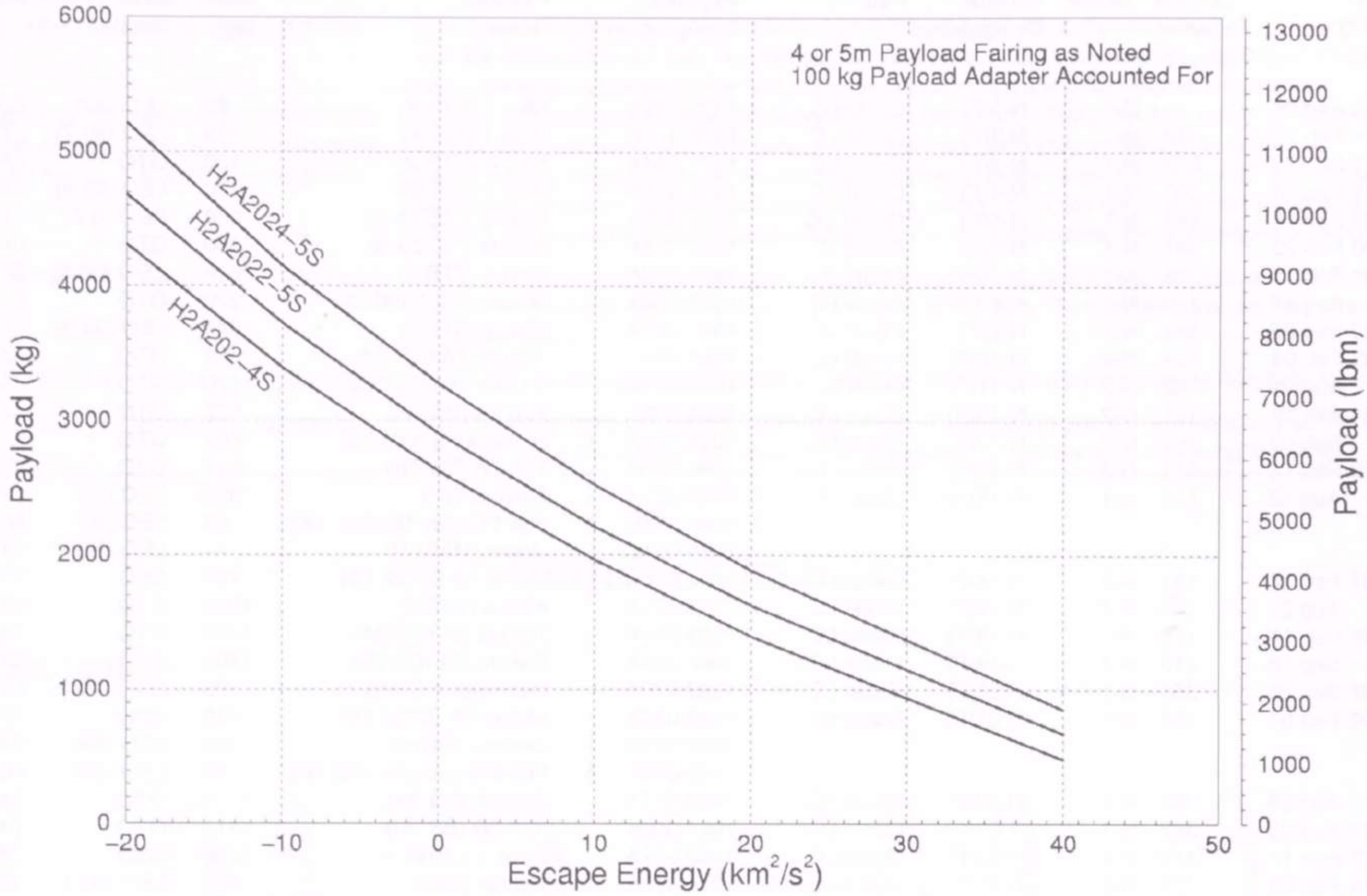


H2A2024: LEO Performance



H2A2024: GTO Performance

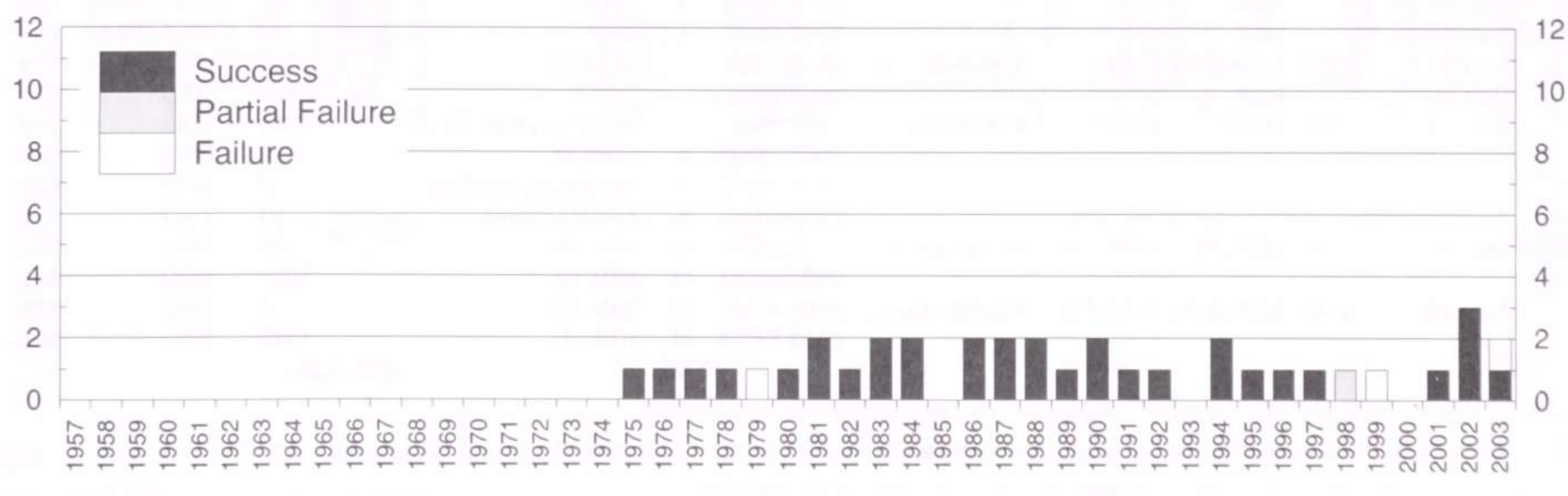
Performance



H-IIA Performance for Earth Escape Missions

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	H-IIA	H-II and H-IIA	N and H Family Total
Total Orbital Flights	6	13	37
Launch Vehicle Successes	5	10	33
Launch Vehicle Partial Failures	0	1	1
Launch Vehicle Failures	1	2	3

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1	1975 Sep 09	—	N-1	N-1(F)	Osaki LC	1975 082A		Kiku 1 (ETS 1)	85	LEO (47)	CIV	Japan
	2	1976 Feb 29	173	N-1	N-2(F)	Osaki LC	1976 019A		Ume 1 (ISS A)	139	LEO (69.7)	CIV	Japan
	3	1977 Feb 23	360	N-1	N-3(F)	Osaki LC	1977 014A		Kiku 2 (ETS 2)	130	GTO	CIV	Japan
	4	1978 Feb 16	358	N-1	N-4(F)	Osaki LC	1978 018A		Ume 2 (ISS B)	140	LEO (69.4)	CIV	Japan
F	**5**	**1979 Feb 06**	**355**	**N-1**	**N-5(F)**	**Osaki LC**	1979 009A		Ayame 1 (ECS A)	260	EEO (0.7)	CIV	Japan
S	6	1980 Feb 22	381	N-1	N-6(F)	Osaki LC	1980 018A		Ayame 2 (ECS B)	260	GTO	CIV	Japan
	7	1981 Feb 11	355	N-2	N-7(F)	Osaki LC	1981 012A		Kiku 3 (ETS 4)	640	EEO (28.6)	CIV	Japan
	8	Aug 10	180	N-2	N-8(F)	Osaki LC	1981 076A		Himawari 2 (GMS 2)	670	GTO	CIV	Japan
	9	1982 Sep 03	389	N-1	N-9(F)	Osaki LC	1982 087A		Kiku 4 (ETS 3)	385	LEO (44.6)	CIV	Japan
	10	1983 Feb 04	154	N-2	N-10(F)	Osaki LC	1983 006A		Sakura 2A (CS 2A)	772	GTO	CML	Japan
	11	Aug 05	182	N-2	N-11(F)	Osaki LC	1983 081A		Sakura 2B (CS 2B)	670	GTO	CML	Japan
	12	1984 Jan 23	171	N-2	N-12(F)	Osaki LC	1984 005A		Yuri 2A (BS 2A)	670	GTO	CML	Japan
	13	Aug 02	192	N-2	N-13(F)	Osaki LC	1984 080A		Himawari 3 (GMS 3)	682	GTO	CIV	Japan
	14	1986 Feb 12	559	N-2	N-14(F)	Osaki LC	1986 016A		Yuri 2B (BS 2B)	677	GTO	CML	Japan
T	15	Aug 12	181	H-1	H-15(F)	Osaki LC	1986 061A		Ajisai (EGP)	685	LEO (50)	CIV	Japan
							1986 061B	A	Fuji 1/Oscar 12 (JAS 1A)	50	LEO (50)	NGO	Japan
							1986 061C	A	Jindai (MABES)	—	LEO (50)	CIV	Japan
	16	1987 Feb 19	191	N-2	N-16(F)	Osaki LC	1987 018A		Momo 1A (MOS 1A)	745	SSO	CIV	Japan
T	17	Aug 27	189	H-1	H-17(F)	Osaki LC	1987 070A		Kiku 5 (ETS 5)	1070	GTO	CIV	Japan
	18	1988 Feb 19	176	H-1	H-18(F)	Osaki LC	1988 012A		Sakura 3A (CS 3A)	1100	GTO	CML	Japan
	19	Sep 16	210	H-1	H-19(F)	Osaki LC	1988 086A		Sakura 3B (CS 3B)	1100	GTO	CML	Japan
	20	1989 Sep 05	354	H-1	H-20(F)	Osaki LC	1989 070A		Himawari 4 (GMS 4)	725	GTO	CIV	Japan
	21	1990 Feb 07	155	H-1	H-21(F)	Osaki LC	1990 013A		Momo 1B (MOS 1B)	740	SSO	CIV	Japan
							1990 013B	A	Orizuru (Debut)	50	LEO (99)	CIV	Japan
							1990 013C	A	Fuji 2/Oscar 20 (JAS 1B)	50	LEO (99)	NGO	Japan
	22	Aug 28	202	H-1	H-22(F)	Osaki LC	1990 077A		Yuri 3A (BS 3A)	1115	GTO	CML	Japan
	23	1991 Aug 25	362	H-1	H-23(F)	Osaki LC	1991 060A		Yuri 3B (BS 3B)	1115	GTO	CML	Japan
	24	1992 Feb 11	170	H-1	H-24(F)	Osaki LC	1992 007A		Fuyo 1 (JERS 1)	1340	SSO	CIV	Japan
T	25	1994 Feb 03	723	H-2	H-II-1F	Yoshinobu LC	1994 007A	C	Ryusei (Orex)	865	LEO (30.5)	CIV	Japan
							1994 007B	C	Myojo (VEP)	2391	EEO (28.5)	CIV	Japan
T,S	26	Aug 28	206	H-2	H-II-2F	Yoshinobu LC	1994 056A		Kiku 6 (ETS 6)	3800	EEO (13)	CIV	Japan
T	27	1995 Mar 18	202	H-2	H-II-3F	Yoshinobu LC	1995 011A	C	SFU (Space Flyer Unit)	3571	LEO (28.5)	CIV	Japan
							1995 011B	C	Himawari 5 (GMS 5)	746	GTO	CIV	Japan
	28	1996 Aug 17	518	H-2	H-II-4F	Yoshinobu LC	1996 046A		Midori (ADEOS)	3547	SSO	CIV	Japan
							1996 046B	A	Fuji 3/Oscar 29 (JAS 2)	49	LEO (98.6)	NGO	Japan
	29	1997 Nov 27	467	H-2	H-II-6F	Yoshinobu LC	1997 074A	C	TRMM	3613	LEO (35)	CIV	USA
							1997 074B	C	Kiku 7/Hikoboshi-Orhime (ETS 7)	2944	LEO (35)	CIV	Japan
P	**30**	**1998 Feb 21**	**86**	**H-2**	**H-II-5F**	**Yoshinobu LC**	1998 011A		Kakehashi (COMETS)	3900	LEO (30)	CIV	Japan
F	**31**	**1999 Nov 15**	**632**	**H-2**	**H-II-8F**	**Yoshinobu LC**	1999 F04A		MTSat	2900	GTO	CIV	Japan
T	32	2001 Aug 29	653	H-2A202	H-IIA-F1	Yoshinobu LC	2001 038A		LRE	86	EEO (28)	CIV	Japan
							2001 038B		VEP 2		GTO	CIV	Japan
	33	2002 Feb 04	159	H-2A2024	H-IIA-F2	Yoshinobu LC	2002 003A		Tsubasa (MDS 1)	480	EEO (28.5)	CIV	Japan
S		Feb 04		H2A			2002 003B	A	DASH	89	EEO (88.9)	CIV	Japan
		Feb 04		H2A			2002 003C	A	VEP	33	EEO (28.5)	CIV	Japan
	34	Sep 10	218	H-2A2024	H-IIA-F3	Yoshinobu LC	2002 042A		USERS	1700	LEO (30.5)	CIV	Japan
		Sep 10		H2A			2002 042B		DRTS	2800	GTO	CIV	Japan
	35	Dec 14	95	H-2A	H-IIA-F4	Yoshinobu LC	2002 056A		Midori 2 (ADEOS 2)	3700	SSO	CIV	Japan
							2002 056B	A	FedSat	58	SSO	CIV	Australia
							2002 056C	A	Kantakun (WEOS)	50	SSO	CIV	Japan
							2002 056D	A	Micro LabSat	54	SSO	CIV	Japan
	36	2003 Mar 28	104	H2A 2024	H-IIA-F5	Yoshinobu LC	2003 009A	M	IGS 1A	850	SSO	MIL	Japan
							2003 009B	M	IGS 1B	1200	SSO	MIL	Japan
F	**37**	**Nov 29**	**246**	**H2A 2024**	**H-IIA-F6**	**Yoshinobu LC**	**2003 F02A**	**M**	**IGS 2A**	**850**	**SSO**	**MIL**	**Japan**
							2003 F02B	**M**	**IGS 2B**	**1200**	**SSO**	**MIL**	**Japan**

Osaki Launch Complex and Yoshinobu Launch Complex are at Tanegashima Space Center

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

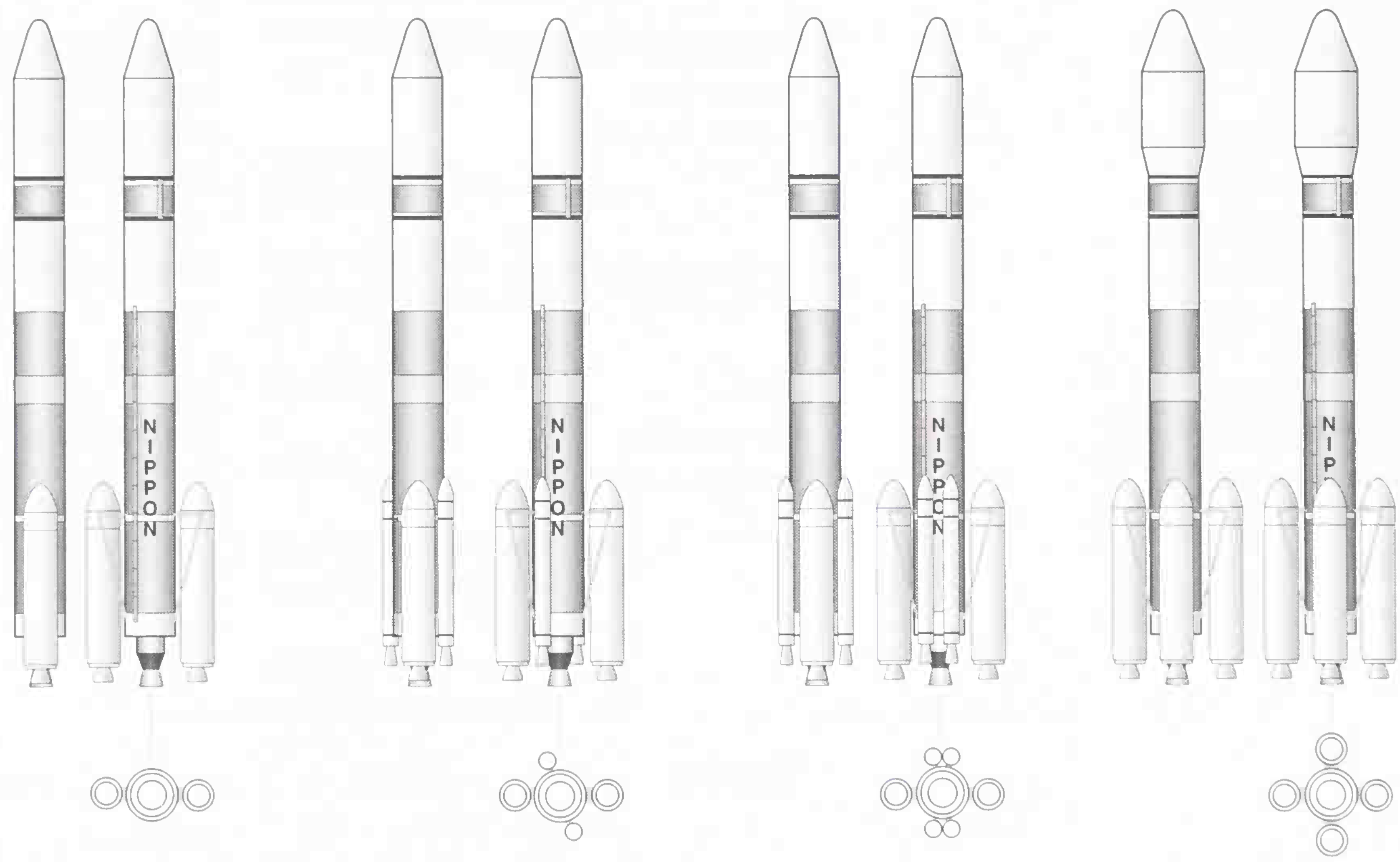
Flight History

Failure Descriptions:

F	1979 Feb 06	N-I N-5(F)	1979 009	Launch vehicle delivered payload to correct GTO orbit, but 10 s after apogee kick motor ignition, contact with the spacecraft was lost. The investigation concluded that a malfunction of the third stage yo-yo tumbler caused the stage to collide with the payload after separation, damaging the satellite.
S	1980 Feb 22	Ayame 2	1980 018A	Contact lost during burn of satellite's solid apogee kick motor likely due to a failure of the motor.
S	1994 Aug 28	Kiku 6	1994 056A	Spacecraft liquid apogee engine failed after 8-min burn stranding spacecraft in GTO.
P	1998 Feb 21	H-II-5F	1998 011	Faulty brazing in second-stage engine cooling system caused engine burn through and cable damage resulting in shutdown midway through the stage's second burn, leaving spacecraft in elliptical LEO instead of GTO. Spacecraft thrusters raised orbit enough to complete some communications experiments.
F	1999 Nov 15	H-II-8F	1999 F04A	Cavitation in the first stage hydrogen turbopump impeller caused an impeller blade to fracture, resulting in loss of fuel flow and rapid shutdown of the engine at T+239 s. The vehicle impacted the ocean 380 km NW of Chichi island.
S	2002 Feb 04	DASH	2002 003B	The DASH reentry experiment failed to separate itself from the launch vehicle due to faulty wiring.
F	2003 Nov 29	H-IIA-F6	2003 F02	A hot gas leak from one SRB-A motor destroyed its separation system. The strap-on did not separate as planned, and the weight of the spent motor prevented the vehicle from achieving its planned velocity.

Vehicle Design

Overall Vehicle



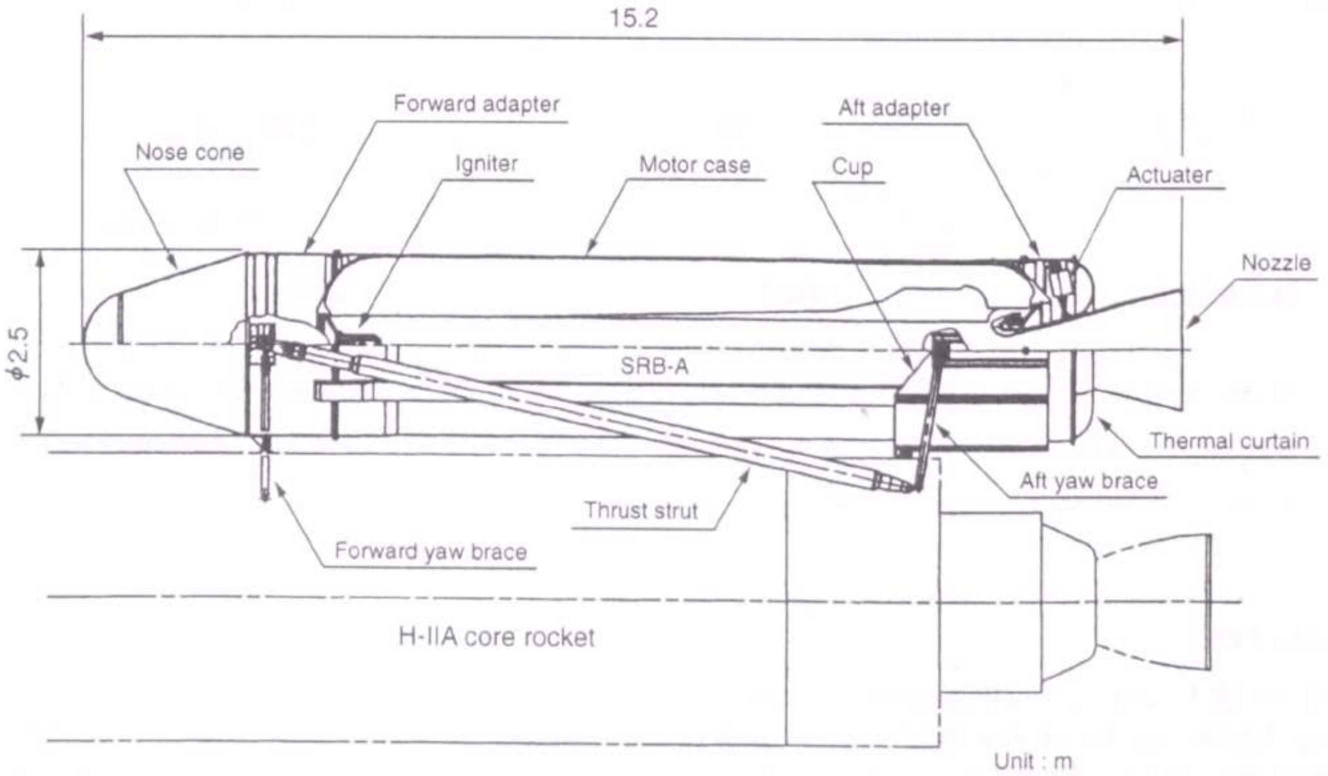
	H-IIA 202	2022	2024	204
Height	5.6 m (172.6 ft)	52.6 m (172.6 ft)	52.6 m (172.6 ft)	5.26 m (172.6 ft)
Gross Liftoff Mass	285 t (628 klbm)	316 t (697 klbm)	347 t (765 klbm)	436 t (961 klbm)
Thrust at Liftoff	4650 kN (1050 klbf)	?	?	?

Solid Rocket Boosters

While the primary stages of the H-IIA use LOX/LH2 propellants, much of the liftoff thrust is provided by a pair of solid rocket boosters (SRBs). SRBs were selected because they provide high thrust in a much smaller system than could be achieved with hydrogen. The H-IIA booster, designated SRB-A, incorporates several improvements over the SRB used on the original H-II. It is a shorter, monolithic motor, which simplifies handling and improves reliability. The motor casing is a filament-wound composite structure using technologies from ATK Thiokol, which reduces weight and permits a higher chamber pressure. The nozzle is smaller and is gimbaled using an electromechanical system rather than the hydraulic system of the original SRB. For larger payloads, the H-IIA can add two or four smaller solid strap-on boosters (SSBs), which are ATK Thiokol Castor IVAXL motors. For even heavier payloads, a second pair of SRB-A motors can be added in place of the SSBs.

Vehicle Design

	H-IIA SRB-A	H-IIA SSB
Dimensions		
Length	15.2 m (49.9 ft)	14.9 m (48.9 ft)
Diameter	2.5 m (8.2 ft)	1 m (3.3 ft)
Mass (each)		
Propellant Mass	65.04 t (143.4 klbm)	13.1 t (28.9 klbm)
Inert Mass	10.4 t (22.9 klbm)	2.1 t (4.6 klbm)
Gross Mass	75.4 t (166.3 klbm)	15.2 t (33.5 klbm)
Propellant Mass Fraction	0.86	0.86
Structure		
Type	Motor: filament-wound monocoque Nose cone: filament-wound monocoque Forward and aft adapters: monocoque	Monocoque
Material	Motor: graphite–epoxy Nose cone: graphite–epoxy Forward and aft adapters: aluminum	Steel
Propulsion		
Engine Designation	SRB-A (Nissan)	Castor IVAXL (ATK Thiokol)
Number of Engines	2	0, 2, or 4
Propellant	HTPB	HTPB
Number of Segments	1	1
Average Thrust (each)	Vacuum: 2260 kN (508 klbf) maximum	Vacuum: 630 kN (141 klbf)
Isp	Vacuum: 280 s	Vacuum: 283.4 s
Chamber Pressure	118 bar (1710 psi)	54 bar (780 psi)
Nozzle Expansion Ratio	17.7:1	15:1
Attitude Control		
Pitch, Yaw, Roll	Electromechanical nozzle gimbal ±5 deg	None, fixed 6 deg nozzle cant
Staging		
Nominal Burn Time	101 s	60 s
Shutdown Process	Burn to depletion	Burn to depletion
Stage Separation	2 separation rockets	



H-IIA Solid Rocket Booster A

Vehicle Design

Stage 1

The major elements of the H-IIA first stage are the aluminum isogrid propellant tanks and the engine section with a single LE-7A LOX/LH2 main engine. The first stage is closely based on the H-II first stage, but includes a number of modifications to reduce production costs. The propellant tank domes are elliptical, single-piece, spun-formed units, in contrast to the hemispherical domes made from individual welded triangular gores on the H-II. This simplifies construction and provides more propellant volume with shorter tanks. Both tanks are covered with PIF insulation to prevent propellant boil off. Pressurization of the LOX tank is provided by an autogenous system that generates gaseous oxygen using a heat exchanger on the engine, replacing a much more complex cryogenic helium system used on the H-II. First stage avionics are contained in the center body section between the tanks. The interstage is a new carbon fiber composite design, replacing the old aluminum structure. It is clearly recognizable in photographs because it is black, not orange like the insulation on the propellant tanks.

The LE-7A is a high-performance engine using a staged combustion cycle to achieve high chamber pressures. The LE-7A is derived from the LE-7 used on the H-II and has similar performance requirements but has a number of design changes to reduce costs and increase reliability. The LE-7A main engine operates at reduced preburner chamber temperatures permitting a simpler, less costly design at the expense of slightly reduced performance. The side-mounted turbopumps of the LE-7 are moved to the top on the LE-7A, making the engine longer but simplifying the plumbing. The number of engine welds on the LE-7 made it difficult to manufacture and caused explosive failures during testing, so the LE-7A is manufactured with advanced casting techniques to reduce the number of parts that must be welded. The LE-7A nozzle was designed with cooling tubes in the upper section only and a lower extension that is film-cooled. This design feature allows the lower extension to be removed to test low-throttle engine performance at sea level without suffering flow separation. However, during testing of the full nozzle it was discovered that ignition and shutdown transients could occasionally cause shocks at the joint of the two nozzle sections that were strong enough to damage the nozzle. As a result, the initial flights of H-IIA use a version of the LE-7A with only the upper section of the nozzle. The specific impulse is significantly reduced because of the lower expansion ratio. Work is in progress to develop a more robust version of the long-nozzle engine for future flights.

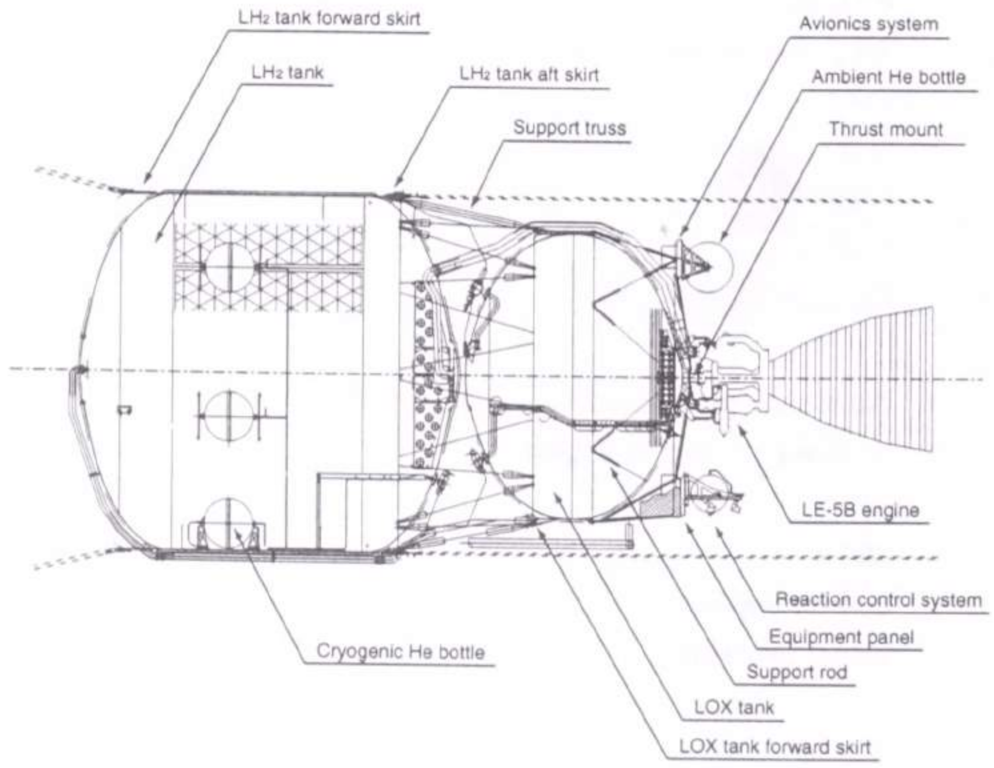
Dimensions	
Length	37.2 m (122.0 ft) with interstage
Diameter	4 m (13.1 ft)
Mass	
Propellant Mass	100.0 t (220.5 klbm)
Inert Mass	13.6 t (30.0 klbm) with interstage
Gross Mass	113.6 t (250.5 klbm)
Propellant Mass Fraction	0.88
Structure	
Type	Tanks: isogrid Engine section: semi-monocoque
Material	Stage 1: aluminum Interstage: graphite epoxy
Propulsion	
Engine Designation	LE-7A (Mitsubishi)
Number of Engines	1
Propellant	LOX/LH_2
Average Thrust	Short Nozzle: 1073 kN (241 klbf) Long Nozzle: 1098 kN (247 klbf)
Isp	Short Nozzle: 429 s Long Nozzle: 442 s
Chamber Pressure	121 bar (1750 psi)
Nozzle Expansion Ratio	Short Nozzle: 39:1 Long Nozzle: 51.9:1
Propellant Feed System	Staged combustion turbopump
Mixture Ratio (O/F)	5.9:1
Throttling Capability	100% or 72 ± 5%
Restart Capability	None
Tank Pressurization	Fuel: GH_2 Oxidizer: GOX
Attitude Control	
Pitch, Yaw	Blowdown hydraulic nozzle gimbal ±6.4 deg
Roll	Auxiliary roll control engine
Staging	
Nominal Burn Time	390 s
Shutdown Process	Burn to depletion
Stage Separation	Linear explosive

Vehicle Design

Stage 2

The second stage of the H-IIA consists of two independent propellant tanks for hydrogen and oxygen, an LE-5B engine, and an aft-mounted avionics shelf. The LE-5 series of engines date back to the upper stage of the H-I launch system. They are distinguished from the LE-7 engines by the use of an expander cycle instead of a staged combustion cycle. In the expander cycle, the turbopumps are powered by hydrogen heated in the cooling passages of the combustion chamber and nozzle, so there is no need for a preburner. In the LE-5B, the hydrogen is heated sufficiently by the combustion chamber alone, eliminating the need for channels in the nozzle skirt found on earlier engines.

Dimensions	
Length	9.2 m (30.2 ft)
Diameter	4.0 m (13.1 ft)
Mass	
Propellant Mass	16.6 t (36.6 klbm)
Inert Mass	3.0 t (6.6 klbm)
Gross Mass	19.6 t (43.2 klbm)
Propellant Mass Fraction	0.85
Structure	
Type	Isogrid
Material	Aluminum
Propulsion	
Engine Designation	LE-5B (Mitsubishi)
Number of Engines	1
Propellant	LOX/LH_2
Average Thrust	137 kN (30.8 klbf)
Isp	447 s
Chamber Pressure	36.3 bar (526 psi)
Nozzle Expansion Ratio	110:1
Propellant Feed System	Expander bleed cycle turbopump
Mixture Ratio (O/F)	5.0
Throttling Capability	100%, 60%, or 3% idle mode
Restart Capability	Multiple (3 starts demonstrated)
Tank Pressurization	Fuel: He during prestart, H_2 tapped from engine during burn Oxidizer: GHe
Attitude Control	
Pitch, Yaw	Electromechanical nozzle gimbal with ±3 deg range
Roll	Hydrazine ACS
Staging	
Nominal Burn Time	534 s
Shutdown Process	Command shutdown



H-IIA Stage 2

Vehicle Design

Attitude Control System

Attitude control during first-stage flight is provided using the nozzle gimbals on the main engine and the SRBs. The SRBs have a 5-deg gimbal capability using flexible nozzle seals and electromechanical actuators. The LE-7 main engine had a high-pressure hydraulic pump to provide continuous pressure for the gimbal actuators. On the LE-7A this is replaced by a simpler blow-down hydraulic system driven by pressurized helium. Nozzle gimbal angles up to 7 deg are possible. Following SRB separation, roll control of the first stage is provided by two tangential auxiliary engines, which generate approximately 1500 N (350 lbf) each using gases bled from the main engine preburner and nozzle cooling lines.

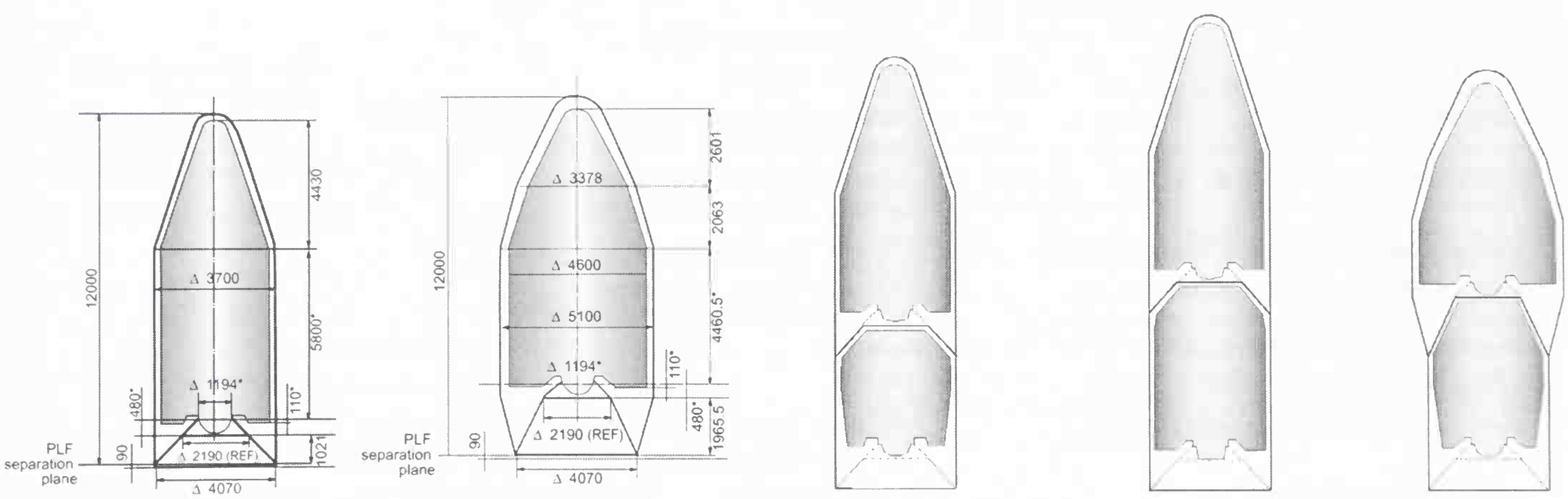
During second-stage burns, pitch and yaw are controlled by gimbaling the second-stage engine nozzle. The LE-5A had pumped-hydraulic actuators, while the LE-5B uses electromechanical actuators for this purpose. A modular hydrazine RCS system provides roll control during second-stage burns and three-axis control during coast and deployment phases. The standard RCS system includes six 50-N (11-lbf) thrusters for attitude control and two 50-N (11-lbf) thrusters for propellant management. Additional thruster modules can be added for missions requiring more ACS capability.

Avionics

The avionics equipment panel is mounted on the bottom of the second-stage oxygen tank. This allows avionics to be integrated on the top of the panel, with propulsion system components such as the RCS system mounted nearby on the bottom of the panel. The vehicle is controlled by a triple-redundant guidance control computer on the second stage (GCC2) that communicates with computers on the first stage through a MIL-STD-1553B data bus. The GCC2 computes inertial position, velocity, and attitude using data provided by an IMU through the data bus. By comparing these data to the planned trajectory, nozzle gimbal angles and engine shutdown times can be calculated. During the first-stage burn, GCC1 uses data from the first-stage lateral acceleration measurement unit and the rate gyro package to overlay additional nozzle gimbal commands.

Payload Fairing

H-IIA fairings are designated by the diameter (4 or 5 m) and the number of payload compartments (S for single or D for dual). The upper and lower compartments of the 4-m-diam dual payload fairing are each available in long or short sections. For example, the 4/4D-SL fairing designation refers to a fairing with two 4-m-diam payload compartments, with a short upper compartment and long lower compartment. Clamshell fairings are jettisoned using separation springs and swing hinges. The single-piece lower fairing of the dual payload fairings is jettisoned forward using springs. A clamshell lower structure is also available in the Model 4/4D-LC fairing. Access doors, radio-transparent windows, and acoustic blankets are standard features.



	Model 4S	Model 5S	Model 4/4D	Model 4/4D-LC	Model 5/4D
Length	12.0 m (39.4 ft)	12.0 m (39.4 ft)	14.5 m (47.6 ft)	16.0 m (52.5 ft)	14.0 m (45.9 ft)
Primary Diameter	4.07 m (13.4 ft)	5.1 m (16.7 ft)	4.07 m (13.4 ft)	4.07 m (13.4 ft)	Upper: 5.1 m (16.7 ft) Lower: 4.07 m (13.4 ft)
Mass	1397 kg (3087 lbm)	1716 kg (3792 lbm)	Long upper: 1226 kg (2709 lbm) Short upper: 1090 kg (2400 lbm) Long lower: 817 kg (1806 lbm) Short lower: 762 kg (1680 lbm)	Upper: 1226 kg (2709 lbm) Lower: 1101 kg (2433 lbm)	Upper: 1470 kg (3249 lbm) Lower: 817 kg (1806 lbm)
Sections	2	2	3 (upper clamshell, lower single unit)	4 (upper and lower clamshells)	3 (upper clamshell, lower single unit)
Structure	Cone: semi-monocoque Cylinder: honeycomb sandwich	Cone: semi-monocoque Cylinder: honeycomb sandwich	Cone: semi-monocoque Cylinder: honeycomb sandwich	Cone: semi-monocoque Cylinder: honeycomb sandwich	Cone: semi-monocoque Cylinder: honeycomb sandwich
Material	Aluminum	Aluminum	Upper: aluminum Lower: graphite–epoxy	Upper: aluminum Lower: graphite–epoxy	Upper: aluminum Lower: graphite–epoxy

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	3700 mm (145.7 in.) for 4-m fairings 4600 mm (181.1 in.) for 5-m fairings
Maximum Cylinder Length (may vary depending on adapter)	5800 mm (228.3 in.) for 4S fairing 4460 mm (175.6 in.) for 5S fairing 3800–4436 mm (149.6–174.6 in.) for dual payload fairings
Maximum Cone Length	4430 mm (174.4 in.) for 4S fairing 4664 mm (183.6 in.) for 5S fairing
Payload Adapter Interface Diameter	Model 937 M: 937 mm (36.9 in.) Model 1194 M: 1194 mm (47.0 in.) Model 1666 M: 1666 mm (65.6 in.) Model 1666 S: 1666 mm (65.6 in.) Model 2360 S: 2360 mm (92.9 in.) Model 3470 S: 3470 mm (136.6 in.)
Payload Integration	
Nominal Mission Schedule Begins	T−18 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T−7 min (next opportunity in 25 min)
On-Pad Storage Capability	24 h without propellant
Last Access to Payload	T−10 h
Environment	
Maximum Axial Load	4.0 *g*
Maximum Lateral Load	1.8 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz/30 Hz
Maximum Acoustic Level	H-IIA 202: 133 dB at 250 Hz
Overall Sound Pressure Level	H-IIA 202: 137.5 dB
Maximum Flight Shock	Clamp band separation system: 4000 *g* at 1500–3000 Hz Separation nut system: 2000 *g* at 800–3000 Hz
Maximum Dynamic Pressure on Fairing	4-m fairing: 52.7 kPA (1100 lbf/ft^2) 5-m fairing: 48.3 kPA (1010 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	GTO: −4.3 kPa/s (−0.62 psi/s) SSO: −4.5 kPa/s (−0.65 psi/s)
Cleanliness Level in Fairing	Class 5000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	SSO: 690 ± 10 km (372 ± 5.4 nmi), 98.6 deg ± 0.18 deg GTO: 250 ± 4 km×36,225 ± 180 km (135 ± 2×19560 ± 97 nmi), 28.5 deg ± 0.02 deg
Attitude Accuracy (3 sigma)	±3 deg, ±0.3 deg/s
Nominal Payload Separation Rate	As required, nominally 1 m/s (3.3 ft/s)
Deployment Rotation Rate Available	0–5 rpm, or up to 50 rpm with spin table
Loiter Duration in Orbit	7000 s after liftoff with optional kit
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Dual payload manifesting is common on H-IIA missions, and dual payload fairing structures are available. Multiple manifesting will be assessed on a case by case basis.
Auxiliary Payloads	Auxiliary payloads can be accommodated depending on available performance and volume.

Production and Launch Operations

Production

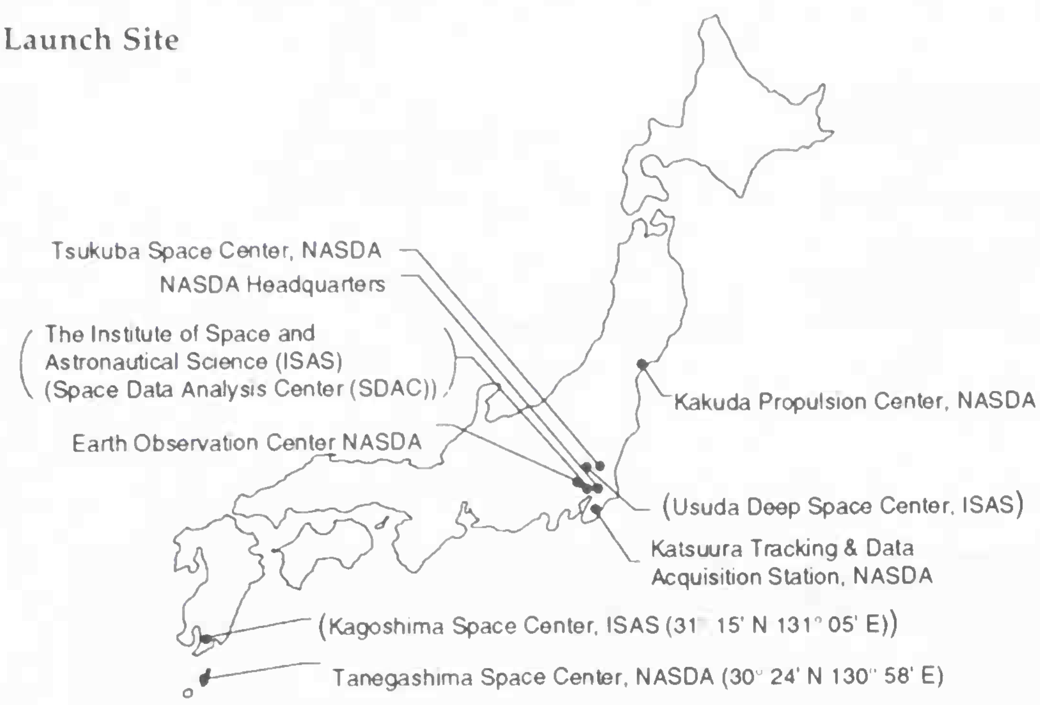
Mitsubishi Heavy Industries is the primary contractor for the H-II series. The engines and vehicle stages are manufactured at Mitsubishi's Nagoya plant and are shipped directly from Nagoya Bay to Tanegashima. The following major companies are involved in the H-II program.

Organization	Responsibility
Mistubishi Heavy Industries	Systems integration LE-7A and LE-5B engines, Stage 1 & 2, vehicle assembly
Ishikawajima Harima Heavy Industries	LE-7A and LE-5B turbopumps
Japan Aviation Electronics Industry	IMU
Kawasaki Heavy Industries	Payload fairings, adapters
Mitsubishi Precision Company	Avionics
NEC Corporation	Avionics
IHI Aerospace	SRB-A boosters
ATK Thiokol	SSB motors

Launch Operations–Tanegashima Space Center

The Tanegashima Space Center is located on the southeastern coast of Tanegashima Island, Kagoshima Prefecture. Tanegashima Island, which is 58 km (36 mi) long with a population of about 40,000, is located 40 km (25 mi) off the southern coast of Kyushu, the southernmost of the four major islands in the Japanese chain. The facilities include the Takesaki Range for sounding rockets and the Osaki Range for space launch vehicles. The center also includes the Masuda Tracking and Data Acquisition Station, the Masuda, Uchugaoka, and Tanegashima radar stations, and three optical tracking stations, as well as test firing facilities for solid and liquid engines. The center occupies approximately 8.6 km2 (3.3 mi2) of land and is the largest launch site in Japan. The major tasks of the center are to check, assemble, and launch rockets, and to perform tracking and control after launch. The center is Japan's primary site for launching applications satellites and test firing solid rocket motors and liquid rocket engines. As a result of objections from fishing unions to the hazards associated with the launches over their fishing grounds, launch windows are restricted to two launch periods each year. In the past these periods lasted from 15 January to 28 February and 1 August to 15 September. However, JAXA has negotiated an increase in the duration of the available launch periods, to support higher flight rates. The new periods cover 1 January through 28 February, and 22 July through 30 September. In addition, two 60-day periods encompassing June to July and November to December are available as launch windows for high-priority launches such as planetary or international missions.

Launch Site

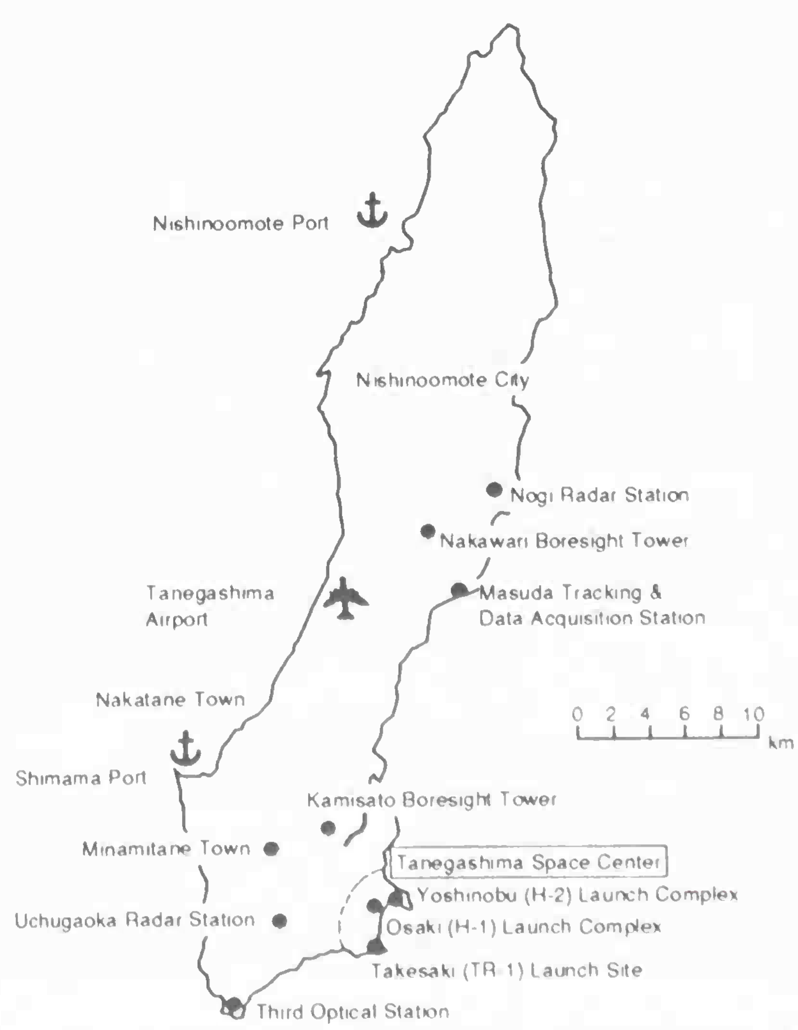


Launch Facilities

Construction of the Yoshinobu Launch Complex for launching H-II rockets at the Osaki Range was begun in 1985. The major facilities in the complex are the Vehicle Assembly Building (VAB), the mobile launcher (ML), two launch pads, the blockhouse, and propellants and high-pressure gas storage. Adjacent to those facilities, a test stand was constructed to conduct firing tests of the LE-7 and LE-7A. An x-ray computer tomography facility is available for nondestructive inspections of SRBs. Launch operations are controlled from within the octagonal Yoshinobu blockhouse and directed from the Takesaki Range Control Center.

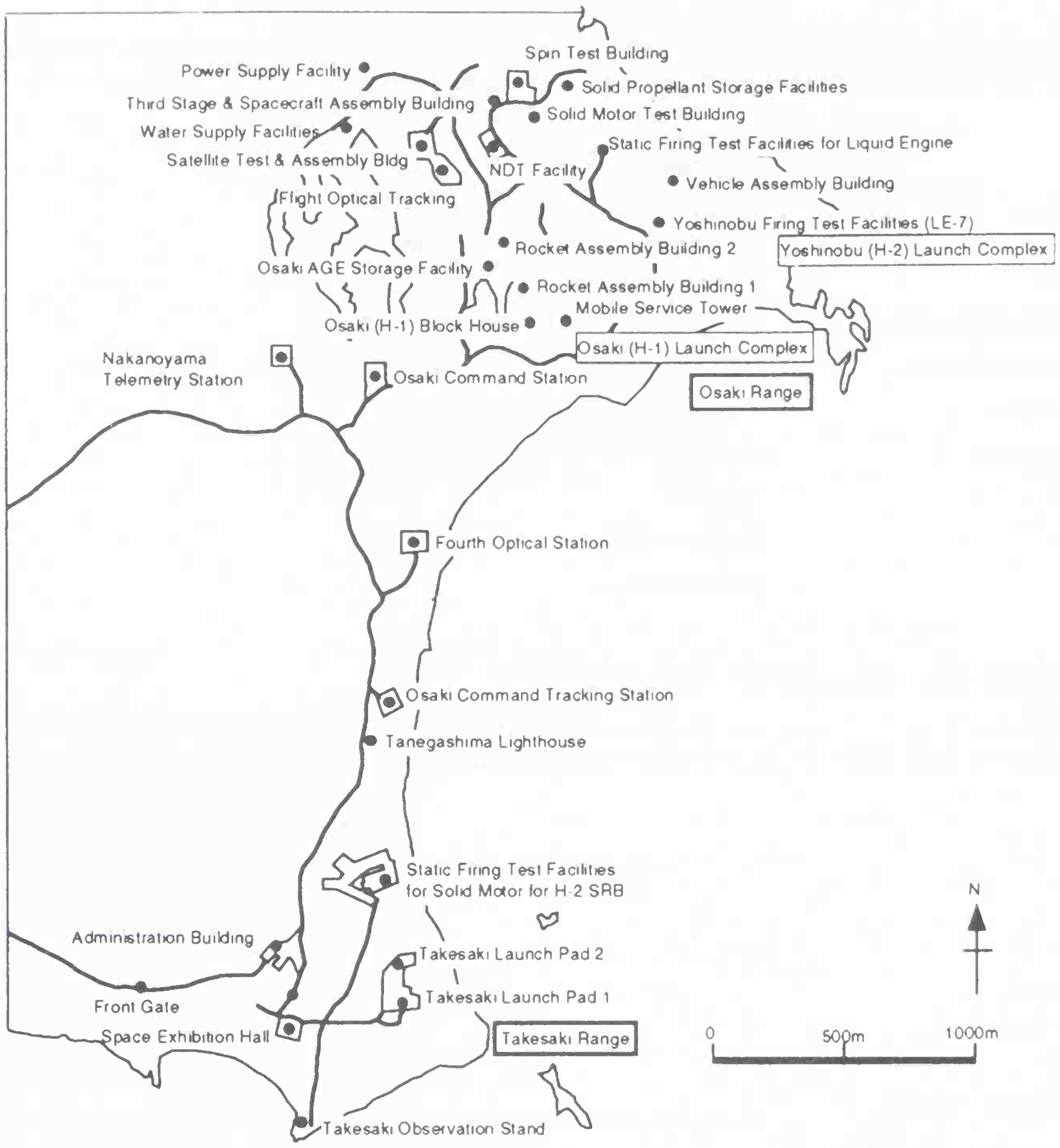
The VAB is a 66-m (215-ft) high structure with two bays for vertical stacking of launch vehicles. The second bay was added for the H-IIA program to support the planned increase in flight rates. While the H-II program performed only vehicle integration and basic checkout in the VAB with most tests performed after rollout to the launch pad, the H-IIA is designed to perform all tests in the VAB and is rolled out to the pad less than 24 h before launch. The launch vehicle is assembled on the deck of one of two MLs. Major umbilicals to the vehicle and payload are routed through service masts mounted on the ML. When vehicle integration and testing are complete and the spacecraft has been mated to the launch vehicle, the booster is carried by the ML along a 500-m (1600-ft) route to the launch pad.

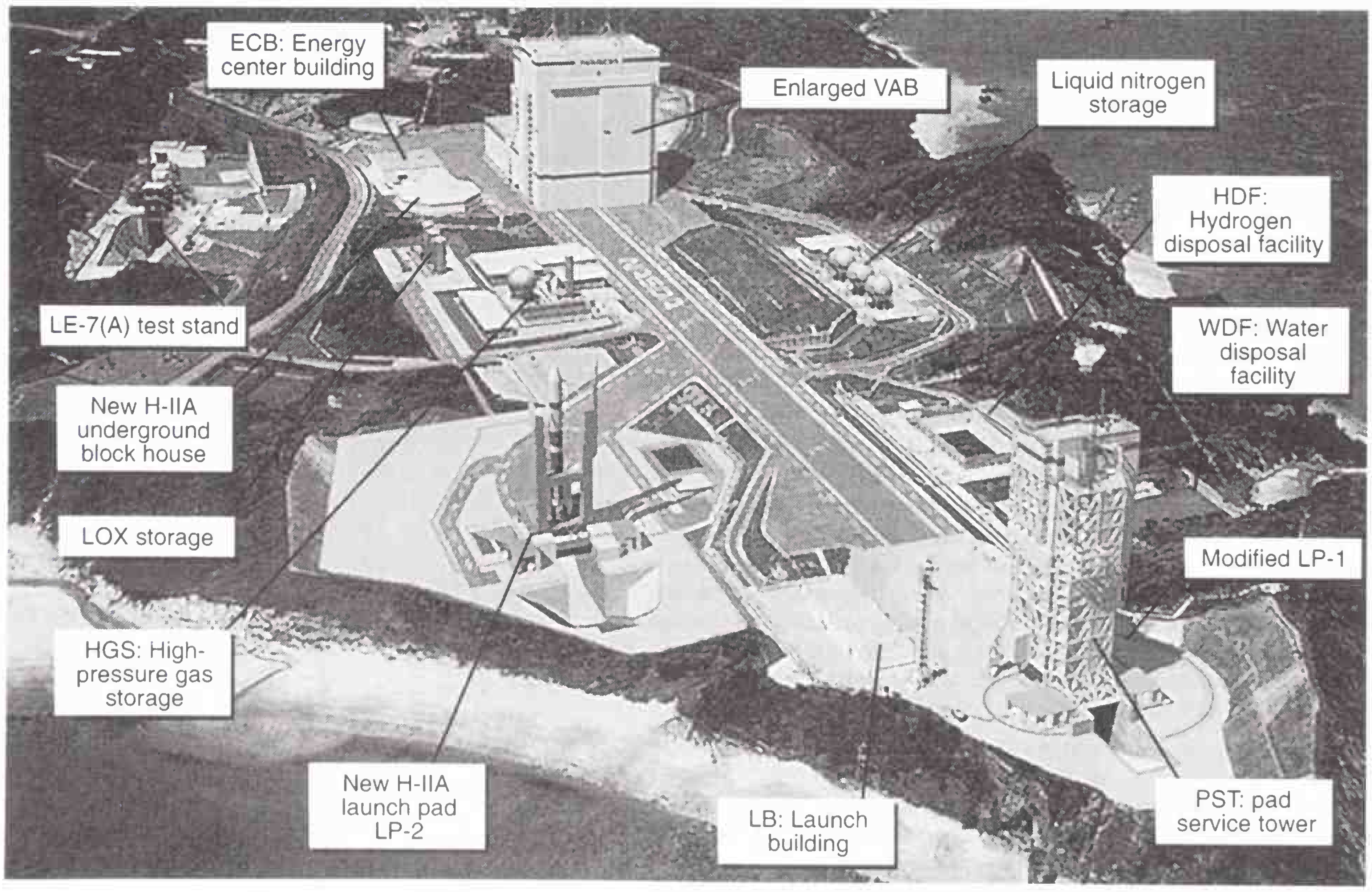
There are two launch pads at the Yoshinobu Launch Complex. Launch Pad 1, built for the H-II program, includes a large Pad Service Tower (PST). The PST stands 75 m (245 ft) high. It consists of a fixed service structure and two rotating service structures that can open to both sides before launch. The fixed service structure includes work platforms at different levels to access the vehicle, and a 20-ton bridge crane to lift the encapsulated fairing and payload into place. Launch Pad 2, built for the H-IIA, is a simple "clean pad" design with no access tower, reflecting the simplified operations sequence for the H-IIA. All interfaces to the launch vehicle are provided by the umbilical masts on the ML.



Location of JAXA (NASDA) Facilities on Tanegashima Island

Production and Launch Operations





Courtesy JAXA.

H-II Yoshinobu Launch Site Layout

Production and Launch Operations

H-IIA204 GTO Flight Sequence

H-IIA Flight Sequence

Event (Time, s)	H2A202	H2A2024
LE-7A Ignition	−6	−6
SRB-A Ignition	−0.6	−0.6
Liftoff	0	0
First SSB Pair Ignition	—	10
SRB-A Burnout	100	100
SRB-A Separation	110	110
First SSB Pair Burnout	—	70
Second SSB Pair Ignition	—	76
First SSB Pair Separation	—	111
Second SSB Pair Burnout	—	136
Second SSB Pair Separation	—	142
Fairing Jettison	264	220
First-Stage Main Engine Cutoff	389	389
First-Stage Separation	399	399
Second-Stage First Ignition	405	405
Second-Stage First Cutoff	732	713
Second-Stage Second Ignition	1466	1421
Second-Stage Second Cutoff	1651	1629
Payload Separation	1687	1664

Vehicle Upgrade Plans

There are a variety of improvements planned for the basic H-IIA configuration, including an enhanced main engine turbopump, the longer first stage engine nozzle, and modified SRBs with longer burn times. In addition, cost reduction efforts will continue.

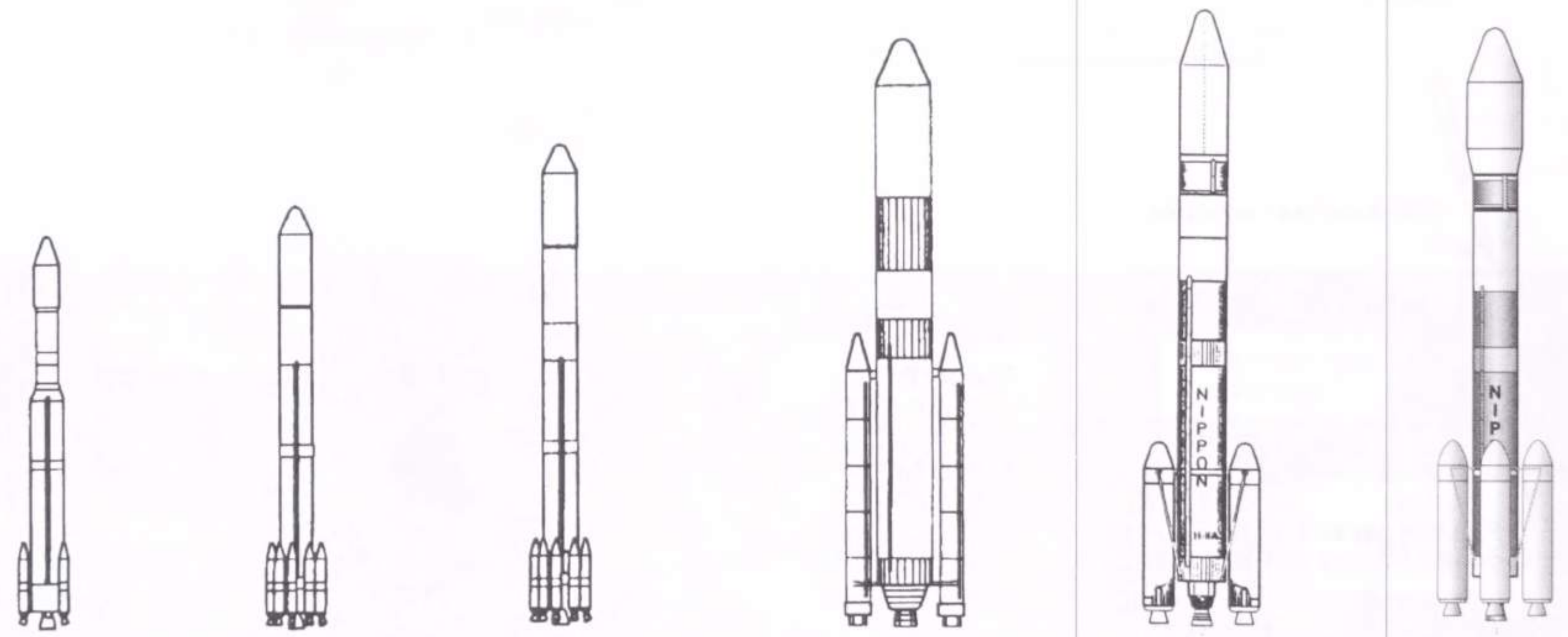
JAXA is planning to develop a 5-m-diam twin engine first stage with 4 SRB-As which is designated H-IIA204H. This configuration could lift 16,500 kg (36,3000 lbm) to LEO for launching H-IIA Transfer Vehicle (HTV), which is used to carry cargo to ISS.

H-IIA204H

Vehicle History

Vehicle Evolution

Retired | Operational | In Development



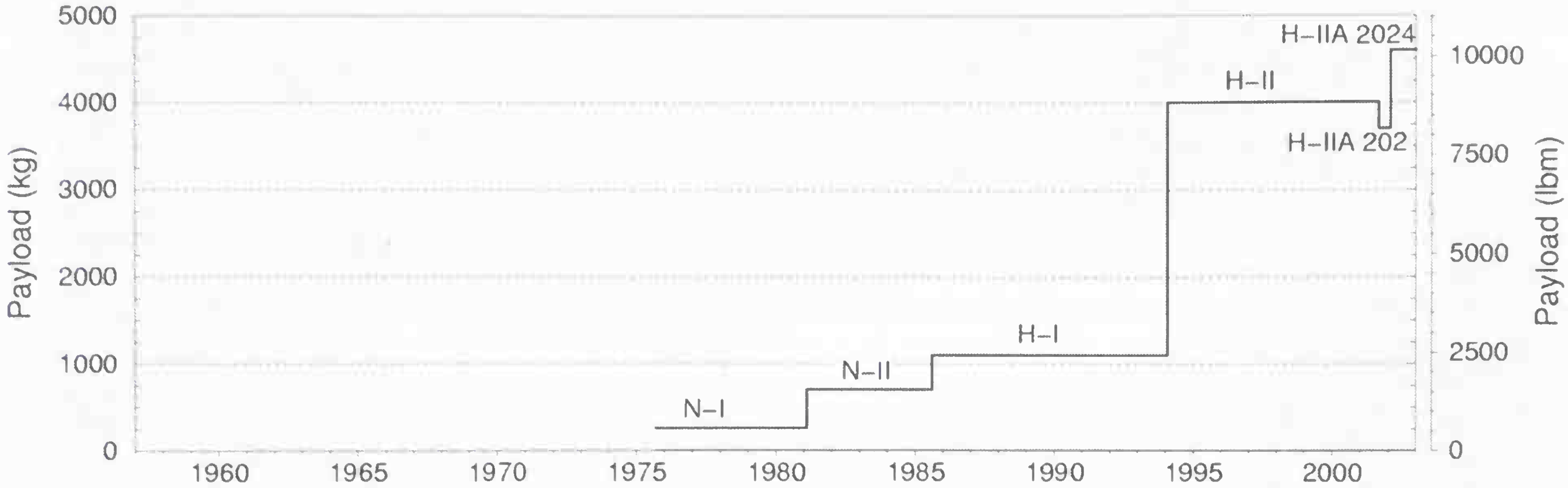
Vehicle	N-I	N-II	H-I	H-II	H-IIA 202	H-IIA 204
Period of Service	1975–1982	1981–1987	1986–1992	1994–1999	2001–Present	2005 or 2006
LEO Payload	1200 kg (2600 lbm)	2000 kg (4400 lbm)	3200 kg (7000 lbm)	10,000 kg (22000 lbm)	10,000 kg (22000 lbm)	?
GTO Payload	260 kg (753 lbm)	700 kg (1543 lbm)	1100 kg (2400 lbm)	4000 kg (8800 lbm)	3700 kg (8150 lbm)	5800 kg (12,800 lbm)

Vehicle Description

- **N-I** Derived from a version of the Thor-Delta launcher. Three solid Castor II strap-on boosters, LOX/RJ-1 first stage, N_2O_4/A-50 second stage, radio guidance, solid spinning upper stage, and 1.65-m-diam (5.4-ft) fairing.
- **N-II** Same as N-I except nine solid Castor II strap-ons, first-stage tank extended, second-stage engine improved, inertial guidance, and 2.44-m-diam (8.0-ft) fairing.
- **H-I** Same as N-II except new LOX/LH_2 second stage and engine, higher mass fraction third stage, and improved inertial guidance.
- **H-II** New vehicle fully developed with Japanese technology features two large solid strap-ons, LOX/LH_2 first stage, a LOX/LH_2 second stage derived from H-I, inertial guidance with ring laser gyros, and 4.07-m-diam (13.4-ft) fairing.
- **H-IIA** Similar to H-II, with lower cost LE-7A and LE-5B engines, new SRB-A and SSB solid boosters, simplified tank structures and lower cost avionics.

Vehicle History

Performance Evolution



Evolution of the N and H series vehicles performance to GTO.

Historical Summary

In 1955, Hideo Itokawa, at the University of Tokyo, assembled a team that produced the "pencil rocket," a small sounding rocket for collecting information about the atmosphere. In 1964, the Japanese government started taking more notice of Itokawa's work with rockets when satellite images of the Tokyo Olympics were broadcast by the United States. The Science and Technology Agency then created the National Space Development Center to investigate potential practical benefits from space development. In 1969, the center became the National Space Development Agency of Japan.

NASDA was responsible for space application missions such as communication satellites or Earth observation satellites. Japan's second space agency, the Institute for Space and Astronautical Science (ISAS), was responsible for space research and technology such as astronomical satellites. Each agency develops its own launch vehicles and operates its own launch site. The Space Activities Commission (SAC), which reports to the prime minister, is responsible for establishing space policy and coordinating.

In 1969 the Japanese government negotiated an agreement with the United States government for the transfer of Delta launch vehicle technology and assistance in development of the N-vehicle. This resulted in development of the N-I vehicle by NASDA, intended to launch much larger spacecraft than the M-vehicle, which was developed by ISAS. In 2003, ISAS, NAL, and NASDA were merged into one independent administrative institution, JAXA.

The Japanese were licensed by the U.S. Department of State to build the entire N-vehicle in Japan, but restricted from commercially offering the vehicle. For reasons of cost and convenience, many components were purchased from U.S. companies. McDonnell Douglas Corporation, the U.S. builder of the Delta launch vehicle, provided the bulk of the assistance on the N-vehicle, supporting overall design, production, and launch operations. Mitsubishi Heavy Industries was the prime contractor for JAXA on the N-vehicle program. Mitsubishi produced the Delta booster under license from McDonnell Douglas and the first-stage engine (MB-3) under license from Rocketdyne. Through the technical assistance of Rocketdyne, it developed the second-stage engine (LE-3), and with help from McDonnell Douglas, it developed the second-stage structure. Components such as Delta tankage (McDonnell Douglas), Castor II strap-ons (ATK Thiokol), solid propellant third stage (ATK Thiokol), control systems (Honeywell), and system analysis (TRW) were purchased from the United States.

From 1975 to 1982, seven N-1 vehicles were flown. The three-stage N-I was capable of delivering 130 kg (290 lbm) to GEO. Because the N-I performance was not sufficient for operation of commercial communication systems or other applications, the N-II was developed. The N-II used nine strap-ons instead of three and improved first, second, and third stages and inertial guidance system. From 1981 to 1987, eight N-II vehicles were successfully launched. All 15 N-vehicle launches were flown from the Osaki Range at the Tanegashima Space Center.

Although the N-II increased performance to GEO by 2.7 times the capability of N-I, application satellites required even larger launch vehicles. The research and development of a larger launch vehicle, beginning in 1977, led to the development of the H-vehicle program. The H-vehicles have significantly enhanced Japan's autonomous capability in the design and use of launch vehicles.

The H-I vehicle, developed as a successor to the N-series rockets, employed a new domestically developed cryogenic second stage, inertial guidance system, and third-stage solid motor, while the first-stage, strap-on boosters, and fairing continued to be manufactured under license. A three-stage H-I rocket could launch a 550-kg (1200-lbm) payload into GEO. A two-stage H-I was available for lower orbits. Two successful test flights of the H-I were flown, a two-stage version on 13 August 1986, and a three-stage version on 27 August 1987. A total of nine H-I launches were conducted through 1992.

Based on the experience gained through the H-I development program, NASDA developed the H-II launch vehicle entirely with Japanese technology. The H-II rocket was designed to serve as NASDA's main workhorse in the 1990s to meet the demand for larger satellite launches at lower cost and still maintain a high degree of reliability. The H-II vehicle consisted of cryogenic first and second stages and a pair of SRBs. The first-stage propulsion engine (LE-7) was designed so that the experience gained in the H-I second-stage engine (LE-5) could be utilized and extended. The SRBs were selected because of the experience with solid rocket motors developed in Japan. The H-II second stage was an enhanced version of the H-I second stage. A strap-down inertial guidance system with ring laser gyros was used for the guidance and control of the H-II, replacing the tuned platform gyros used on the H-I. The H-II had a 4-m-diam (13-ft) payload fairing that could accommodate either one or two payloads. A 5-m (16-ft) Space Shuttle-class fairing was also developed. The payload capability was 4000 kg (8800 lbm) into GTO, or four times the capability of H-I.

The H-II development program was approved by the Space Activities Commission in 1984 based on the results of trade studies by NASDA. Having verified the feasibility of the vehicle design with the systems study and component tests, NASDA began the development in 1985 and established the vehicle baseline configuration at the preliminary design review in May 1987—three months before the first H-I launch. To verify their analyses on the launch environment and SRB separation, NASDA launched a series of one-quarter scale, suborbital test rockets in 1988 and 1989. At the critical design review in July 1990, the H-II design was finalized. As a result of engineering development problems associated with the LE-7 first stage cryogenic engine, it was

Vehicle History

decided at the review to reduce the thrust of the LE-7 to 90% of the original design target and to delay the first launch from 1992 to 1994. Fortunately, the LE-7 performance loss was offset by higher performance values of the SRB and second stage demonstrated by test results.

The first H-II launch occurred on 4 February 1994 from the new Yoshinobu Launch Complex at Tanegashima Space Center. Two payloads were flown on this flight, the Vehicle Evaluation Payload (VEP), which measured the rocket's performance, and the orbital reentry vehicle (OREX), an aerodynamic shell body designed to collect atmospheric reentry data. The flight was termed a "100 percent success."

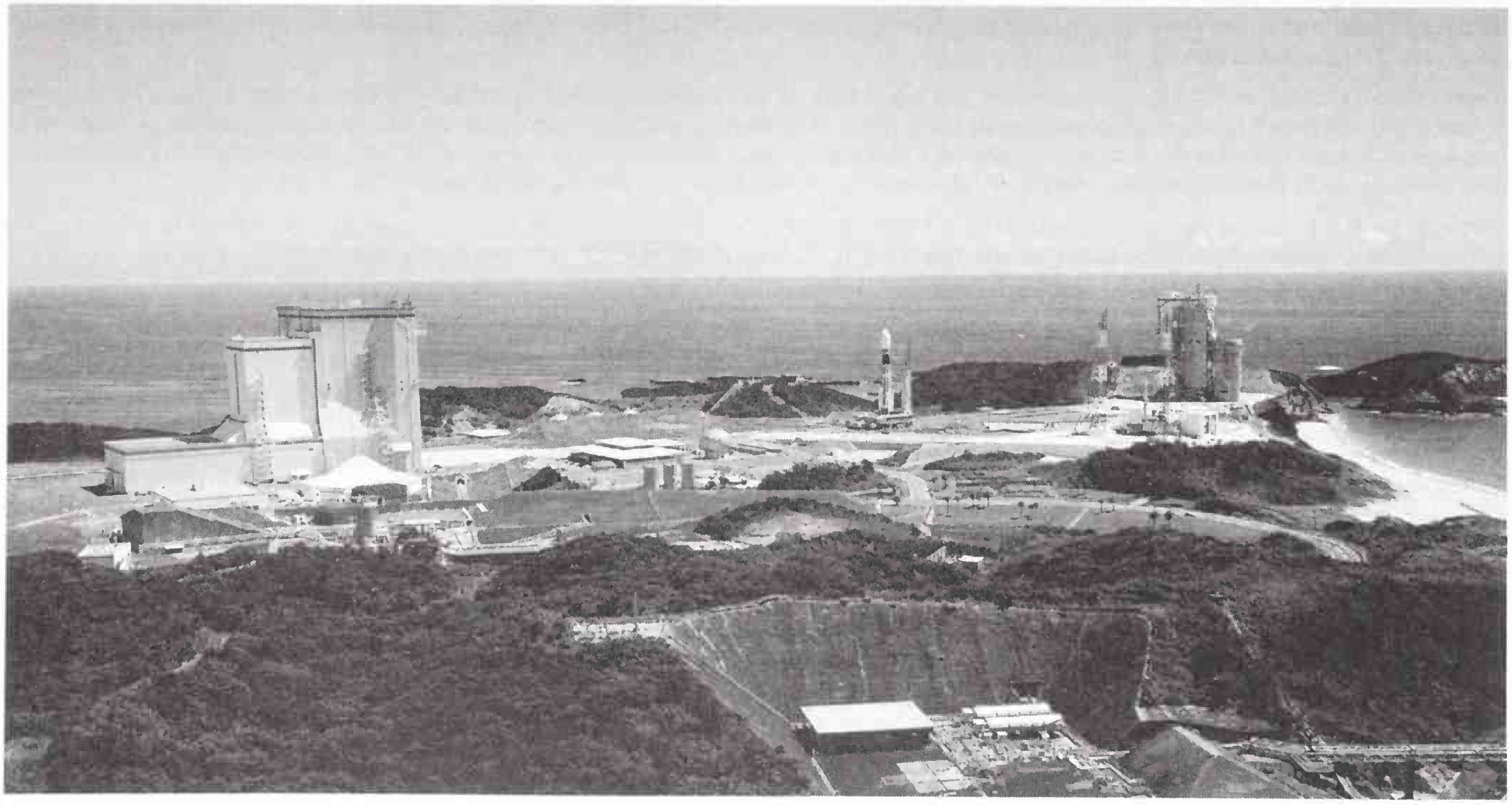
While the H-II was a technological triumph for Japan, it was not a commercial success. This resulted in large part from two factors beyond control of NASDA. In the mid 1980s, when the H-II was approved and preliminary design began, it was assumed that within a decade Japanese companies would be building a significant number of satellites and that the H-II would be the default choice for launching these satellites. However, in the late 1980s, the United States forced Japan to open its markets for high-tech products, including satellites and launch vehicles. As a result, most Japanese commercial satellites are now built by U.S. companies and launched on European or U.S. launch vehicles. The second problem was that the dramatic rise in the value of the yen made it impractical to sell launches to foreign customers. The value of the yen relative to the dollar nearly doubled during the time the H-II was being developed. By the time the H-II became operational in the mid 1990s the dollar-denominated price for a launch reached $190 million—twice the price of comparable Ariane or Atlas boosters. Consequently, the H-II was left to launch a few government payloads per year.

To overcome this problem, NASDA initiated the H-IIA development program, with the primary goal of cutting launch costs in half, from about 19 billion yen to 8.5 billion yen ($145 million to $65 million). One important change would be a higher launch rate. Early contracts from Hughes and Loral guaranteed the purchase of up to 20 launches if the target price could be met. Negotiations with local fishing unions also lengthened the available launch windows, allowing more launches per year. The higher launch rate made larger production lots possible, which are much more efficient than manufacturing one or two vehicles at a time. Costs were also reduced by redesigning the vehicle to simplify manufacturing and launch operations. The duration of each launch campaign was reduced from 50 days for the H-II to 20 days for the H-IIA. Another major change was the decision to purchase some components from lower-cost foreign suppliers. While Japanese technological independence was a primary purpose of the original H-II program, the overriding commitment to low cost in the H-IIA program was demonstrated by contracts with ATK Thiokol of the United States. Although Japanese companies have experience with solid motors, ATK Thiokol was selected to supply technologies for the SRB-A composite cases and fully assembled motors for the SSB. Boeing and Man Technologies of Germany were also selected to produce core-stage tank domes.

Although the future looked bright, NASDA soon faced obstacles on the road to the H-IIA. In 1998, H-II-5F failed to deliver its payload to the proper orbit because of a burn-through on the second stage engine. The next mission, H-II-8F, was a hybrid of the H-II and H-IIA, using the new second stage on an otherwise standard H-II. Unfortunately, mission controllers had to terminate the flight after a main engine failure on the first stage. The cause of the failure could not be conclusively determined from telemetry, so NASDA decided it would be necessary to examine the engine. In an unusual operation, the Japan Marine Science and Technology Center performed a deep ocean search in the impact area 380 km northwest of Chichi Island and recovered the LE-7 engine from a depth of 3000 m (10,000 ft). By examining the recovered hardware, engineers determined that resonance caused by cavitation in the impeller of the hydrogen turbopump caused the LE-7 to fail.

Japan's space launch systems had built a reputation for reliability based on a long track record of success. In fact, the contract to launch MT-SAT on H-II-8F included no provisions in the event of a launch failure. So the consecutive failures of two H-II launches and of an M-V rocket built by JAXA (ISAS) caused repercussions throughout Japan's space industry. The H-II program was terminated early, and the last remaining H-II, H-II-7F, was donated to Japan's National Science Museum. The H-IIA program was delayed by a year and a half by additional testing to ensure that its first flight would succeed. American satellite manufacturer Hughes cancelled its order for 10 H-IIA launches citing the failures and delays. The Ministry of Education and Science also listed the failures as one reason for its decision to begin merging Japan's three space agencies. This merger was completed in October 2003.

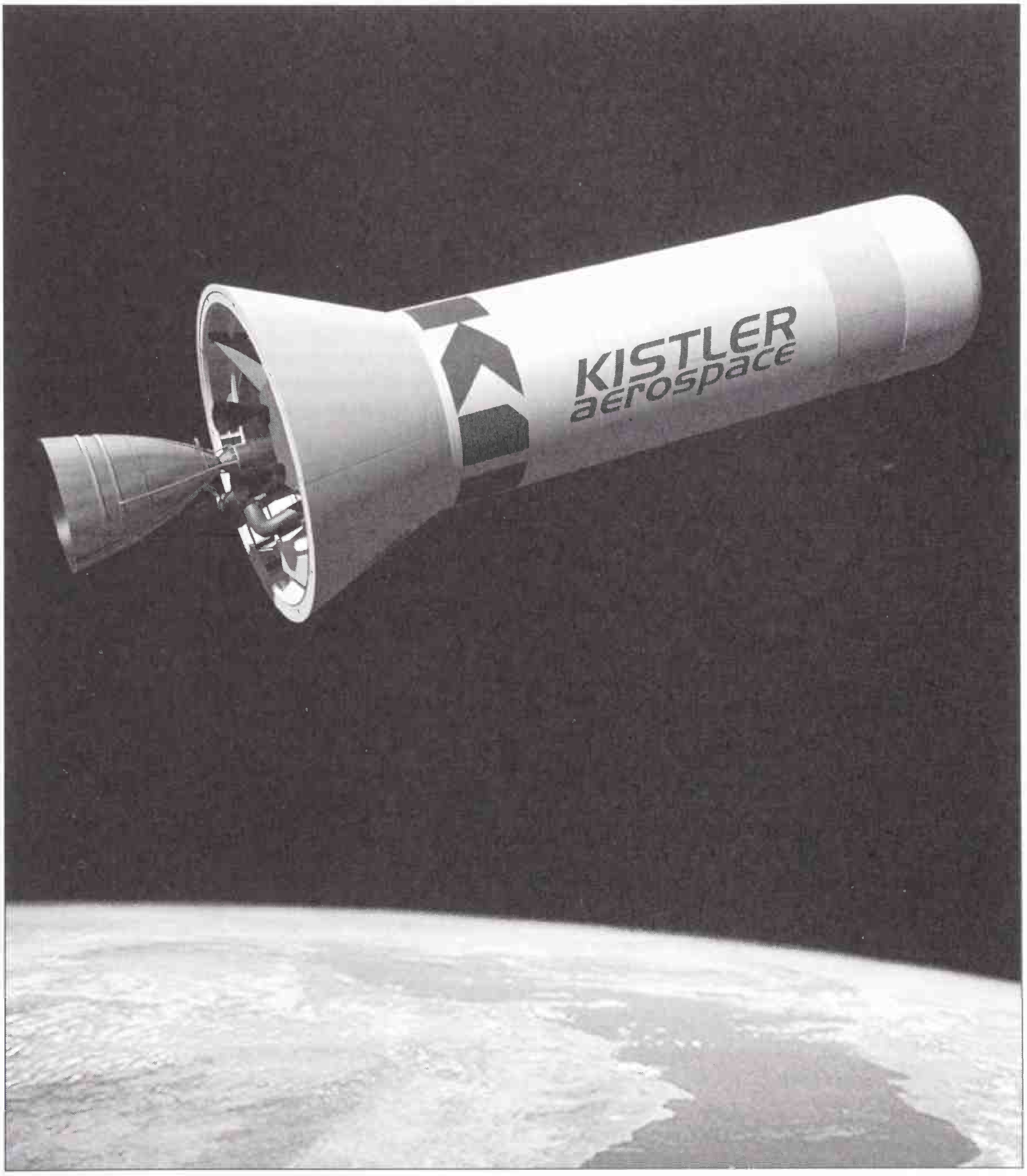
When the first H-IIA was readied for its demonstration launch in August 2001, a great deal was riding on the outcome. Fortunately, the launch was a success. In February 2002 a second test launch demonstrated the more powerful 2024 configuration and a number of modifications that had not been ready in time for the first launch. The two successful test flights cleared the way toward full operational capability for the H-IIA.



Courtesy JAXA.

The H-IIA Ground Test Vehicle (GTV) is shown being rolled out for pathfinder testing at the Yoshinobu Launch Complex before the first H-IIA launch. The Vehicle Assembly Building is at left. The two launch pads are at right.

K-1



Courtesy Kistler Aerospace Corporation.

The Kistler Aerospace Corporation's K-1 is a commercially developed, two-stage reusable launch vehicle, which will be capable of carrying small to medium payloads into LEO or to higher orbits with an optional upper stage.

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K-1

General Description

K-1

Summary

The K-1 is a new two-stage reusable launch vehicle being developed commercially by Kistler Aerospace Corporation. The K-1 can carry payloads up to 4600 kg. Both stages are recovered using parachutes and airbags. Launch operations will be conducted from new launch facilities in Australia and Nevada. Both stages burn LOX/kerosene propellants and are powered by Aerojet engines modified from the core Russian NK-33/43 engines. An optional expendable third stage, fueled by MMH/N_2O_4, can be used for high-orbit missions. An optional cargo module can be used for International Space Station (ISS) resupply missions.

Status

In development. First launch planned 13–18 months after completion of financing.

Origin

United States

Key Organizations

Marketing Organization	Kistler Aerospace Corporation
Launch Service Provider	Kistler Aerospace Corporation
Prime Contractor	Kistler Aerospace Corporation

Primary Missions

ISS servicing, LEO satellites, or small satellites to high-energy orbits with optional upper stage.

Estimated Launch Price

$17 million for LEO missions (Kistler 2002)
$25 million with upper stage (Kistler 2002)

Spaceports

Launch Site	Woomera, Australia
Location	31.1° S, 136.6° E
Available Inclinations	45–60 deg, 84–99 deg
Landing Site	Woomera, Australia
Launch Site	Nevada Test Site
Location	37.3° N, 116.5° W
Available Inclinations	45–60 deg, 84–99 deg
Landing Site	Nevada Test Site

Performance Summary

The following performance values are for a launch from the Woomera Spaceport using the standard payload module for LEO missions, the Active Dispenser Module for high orbits, and the ISS cargo module for ISS missions. Payload adapter mass must be subtracted to determine available spacecraft mass.

200 km (108 nmi), 45 deg	4600 kg (10,150 lbm)
200 km (108 nmi), 90 deg	3000 kg (6600 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	3750 kg (8250 lbm) upmass
	900 kg (2000 lbm) downmass
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	1250 kg (2750 lbm)
GTO: 200×35,786 km (108×19,323 nmi), 45 deg	1570 kg (3460 lbm)
Geostationary Orbit	800 kg (1760 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	0

Flight Rate

To be determined

Nomenclature

The K-1 designation refers to the first launch vehicle developed by Kistler Aerospace Corporation. The first stage of the K-1 is referred to as the Launch Assist Platform (LAP). The name stems from an early design in which the first stage was to be a rocket-powered platform that carried the second stage. The second stage is called the Orbital Vehicle (OV). Spacecraft are carried in a reusable payload module (PM) rather than a conventional payload fairing. Three types are available – the Standard Payload Module (SPM), Extended Payload Module (EPM), and Active Dispenser Payload Module (ADPM). The phrase Active Dispenser refers to an optional expendable upper stage. A cargo module (CM) is also available to support ISS resupply missions.

Cost

The K-1 development program has been financed commercially. More than $600 million in financing was raised through the mid 1990s before investment began to decline. The funding was provided by investors in the United States, Saudi Arabia, Asia, and Europe, and from vendors and subcontractors. For example, Northrup Grumman had a contract for $145 million to build structures for the vehicle, and advanced $30 million to Kistler in 1998 to continue work when funding slowed. In 2001 Kistler received a contract worth up to $135 million from NASA under the Space Launch Initiative. However, this consisted of a $10 million firm contract and options for up to $125 million for flight demonstrations of 13 technologies if the K-1 vehicle was developed. (These options had not been exercised at the time of publication.)

According to Kistler, LEO missions will be priced at $17 million per flight. Missions requiring the expendable upper stage will be priced at $25 million. These prices do not include mission integration.

Availability

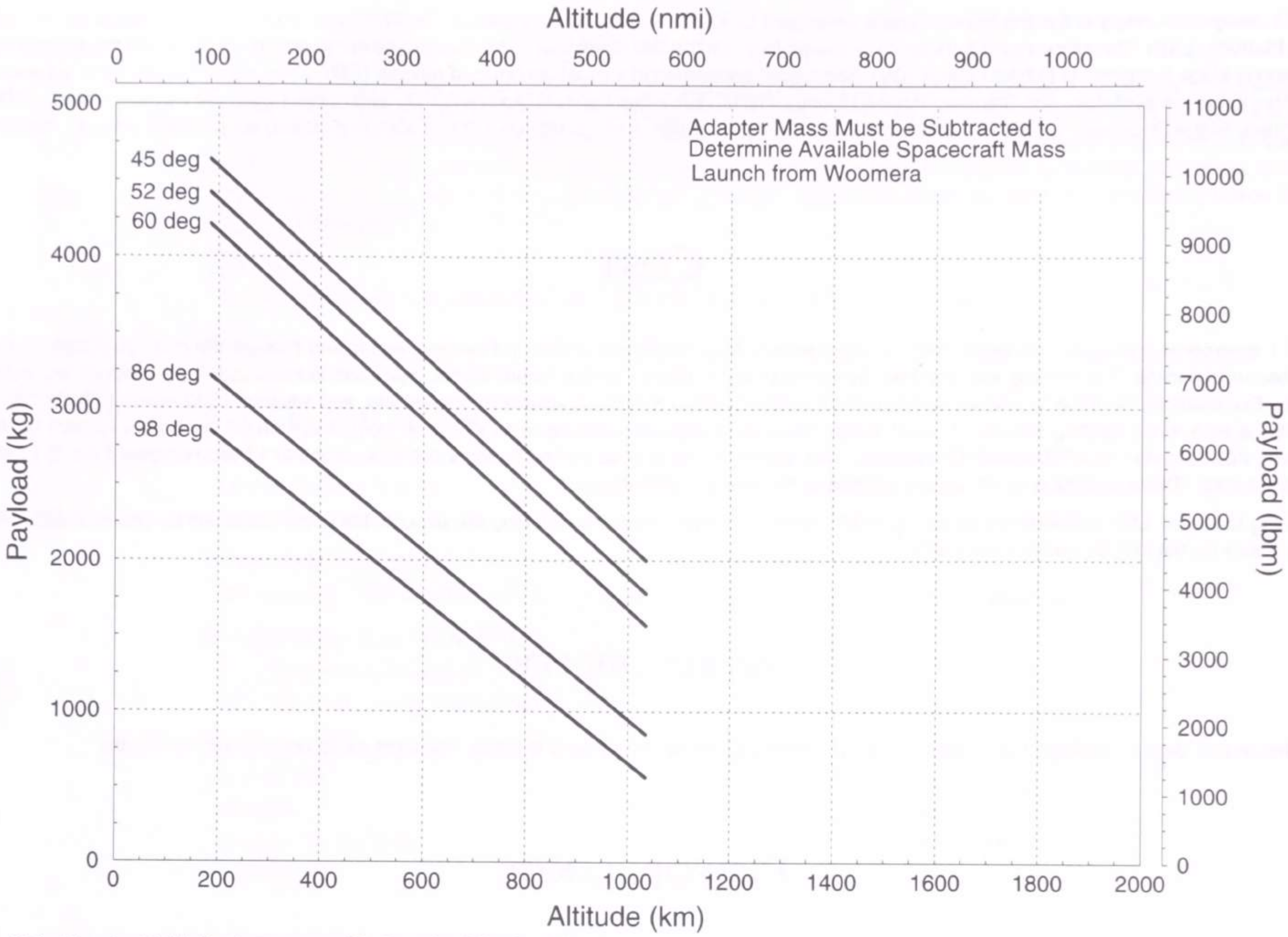
The first launch date is contingent on completion of fundraising. Once full financing is in place, first flight could occur within 18 months.

Performance

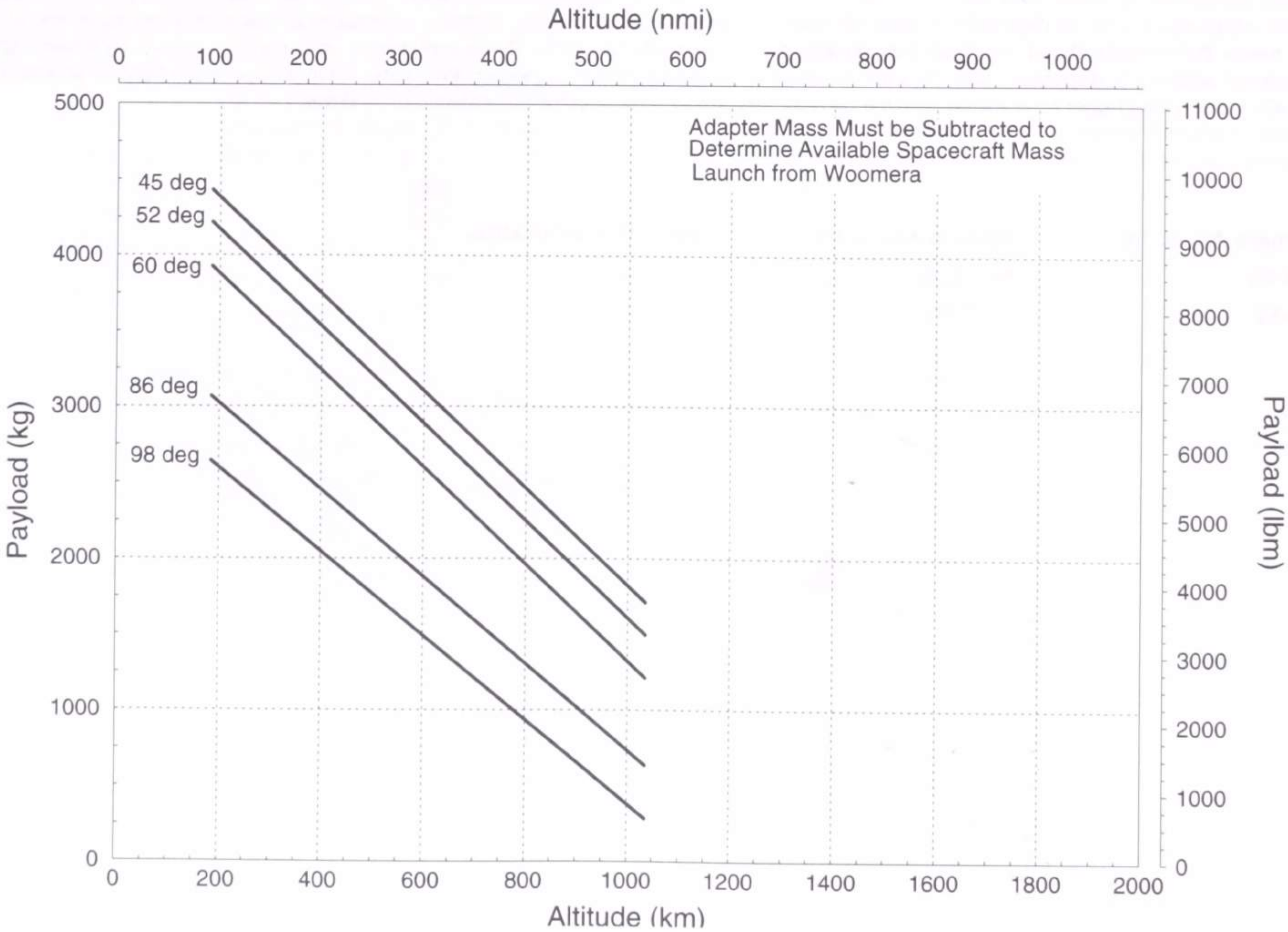
The K-1 will be launched from Spaceport Woomera in Australia. A second launch site is also planned at the Nevada Test Site in the United States. Both sites are landlocked, so launch azimuths are limited to those that do not endanger populated areas. The K-1 can reach inclinations in two sectors between 45–60 deg and 84–99 deg from both sites. As operational experience is gathered, additional inclinations can be considered depending on customer needs. Performance shown includes a 3-sigma flight performance reserve (FPR). These performance values do not account for mission-specific spacecraft adapters or dispensers. Launches are assumed to be conducted from Spaceport Woomera, the location of initial flights. Performance is up to 400 kg (880 lbm) higher for launches from the Nevada Test Site as a result of the higher launch site elevation.

Woomera Azimuths	Nevada Azimuths	Resulting Inclination
55–33 deg	62–36 deg	45–60 deg
5–14 deg	5–14 deg	84–99 deg

Performance

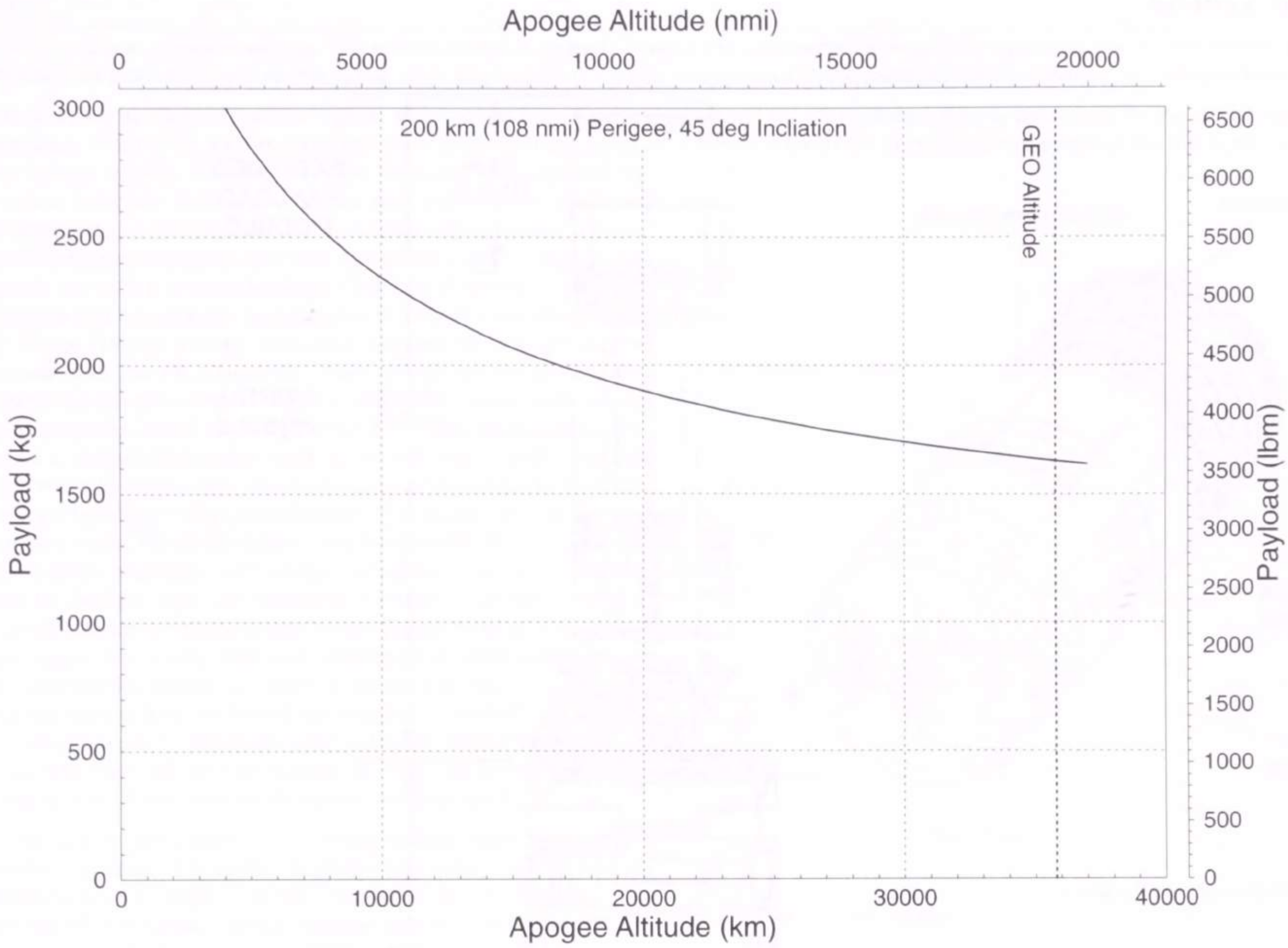


K-1: Circular Orbit Performance with Standard Payload Module

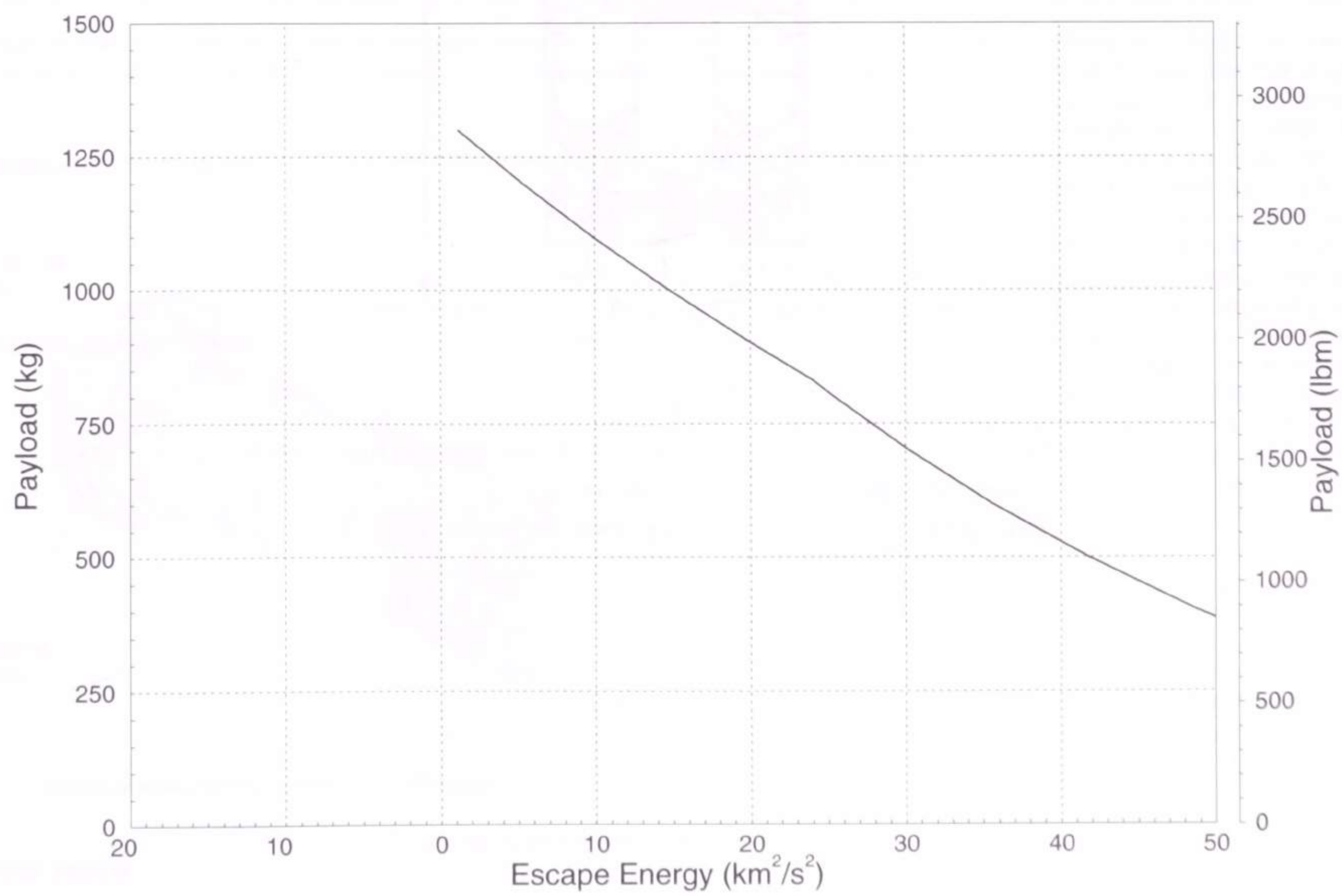


K-1: Circular Orbit Performance with Extended Payload Module

Performance



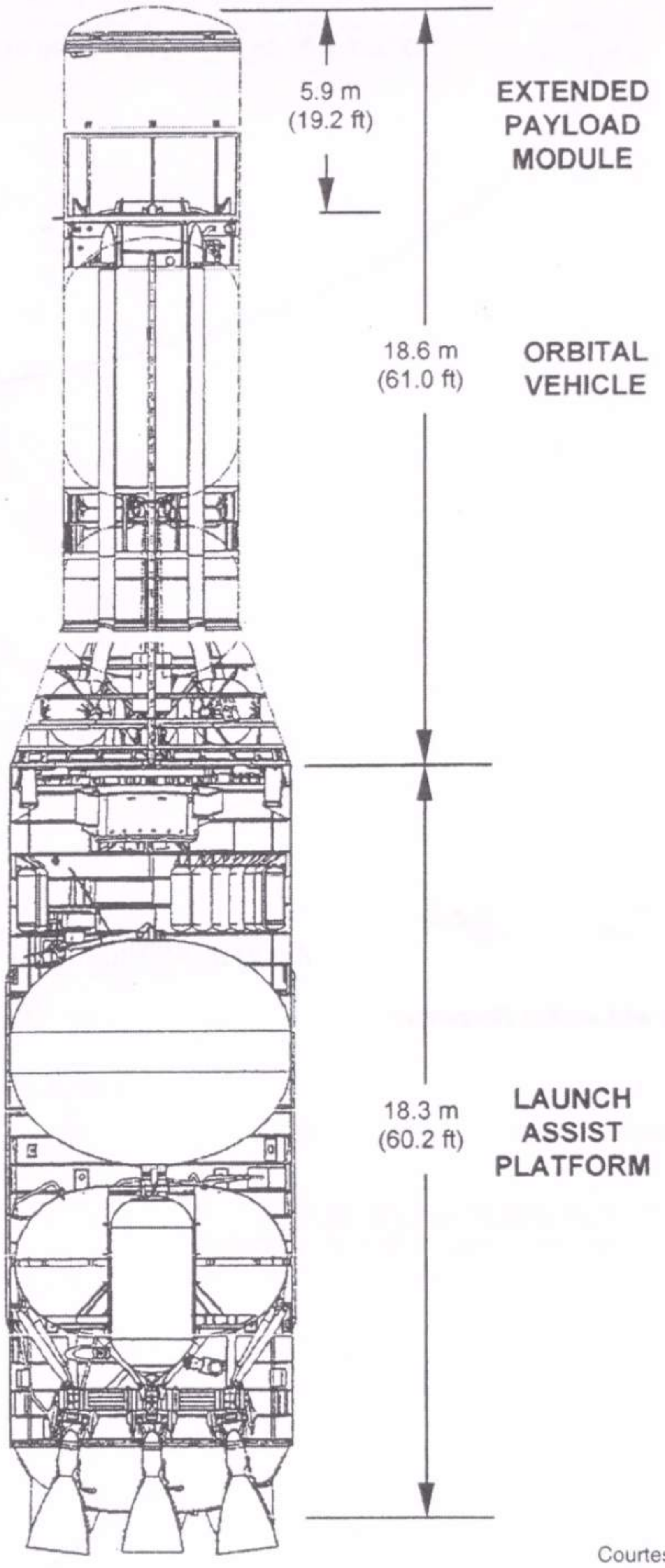
K-1: GTO Performance with Active Dispenser



K-1: Earth Escape Performance with Active Dispenser

Vehicle Design

Overall Vehicle



Courtesy Kistler Aerospace Corporation.

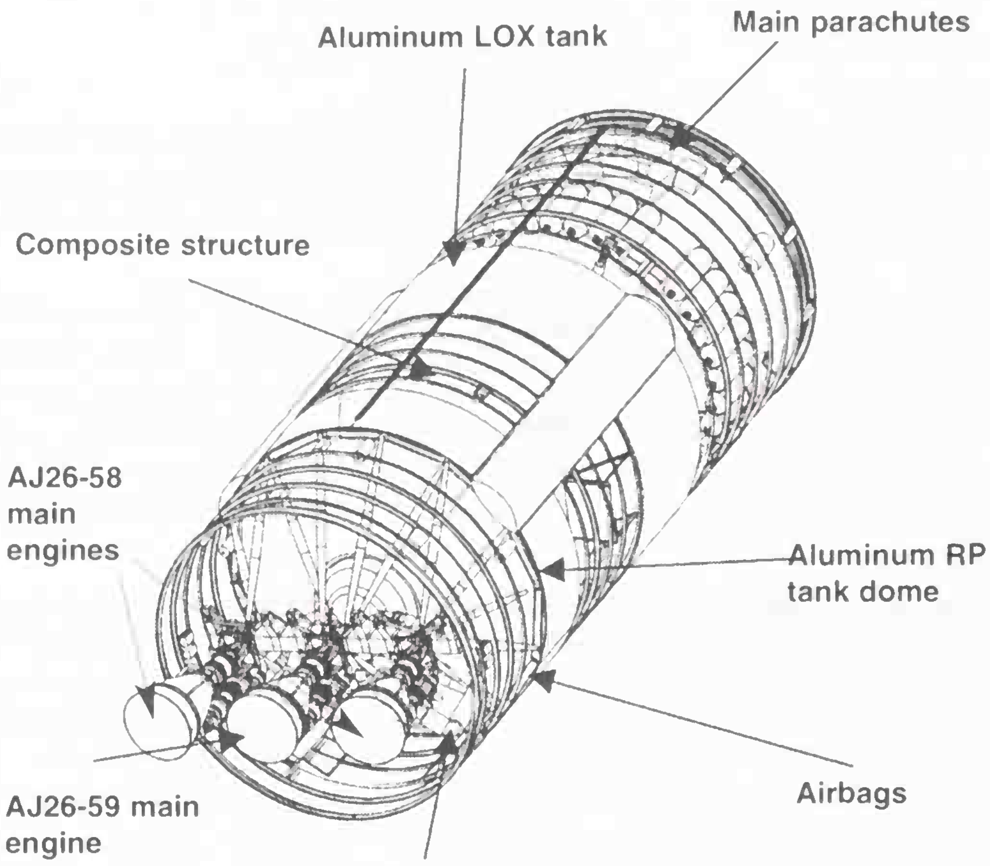
	K-1
Height	36.9 m (121.2 ft) with Extended Payload Module
Gross Liftoff Mass	382 t (841 klbm)
Thrust at Liftoff	4540 kN (1020 klbf)

Vehicle Design

Stages

The K-1 is a two-stage, reusable vehicle. The system design is strongly driven by requirements for reusability, system longevity (the vehicle is designed for 100 flights), low cost, and rapid maintenance. Both stages burn RP grade kerosene and LOX, avoiding the complexities of hydrogen-fueled systems.

The K-1 is powered by liquid-propellant engines from Aerojet based on the NK-33 and NK-43 engines originally developed for the Soviet manned lunar landing program. The NK-33 and NK-43 in turn were developed as upgrades to the NK-15 and NK-25 engines that powered the test flights of the Soviet N-1 heavy launch vehicle. The engines were designed and produced by ND Kuznetsov Scientific Technical Complex and underwent a very thorough development and test process to demonstrate high reliability, extended life, and reusability. A total of 250 NK33/43 engines were tested, accumulating over 108,000 s of test firing time. (This was in addition to the 581 NK-15 engines that had already accumulated 86,000 s of test firing time.) Up to 17 firings (16,000 s total) were demonstrated on a single engine between overhauls. The NK-33 and NK-43 are very similar, the primary distinction being that the NK-43 has a larger nozzle for improved performance in vacuum, while the NK-33 is designed for first-stage applications. The engines use a staged combustion cycle to achieve high specific impulse and a high thrust-to-weight ratio. The turbopump is driven by an oxidizer-rich preburner, resulting in high performance. The engines for the K-1 use the extensively tested NK-33/43 engine core (turbomachinery, hot gas systems, combustion chamber, and nozzle) combined with new systems developed by Aerojet, such as electronic controllers, ignition systems, electromechanical valve actuators, and a new gimbal bearing. The center first-stage engine has a new dual-start cartridge and other modifications required to restart the engine for return to the launch site. The K-1 first-stage engines derived from the NK-33 are designated the AJ26-58 (single-start) or AJ26-59 (restart). The second-stage engine derived from the NK-43 is designated the AJ26-60. The engines are intended to be refurbished after 10 flights, then flown 10 more times before being retired.



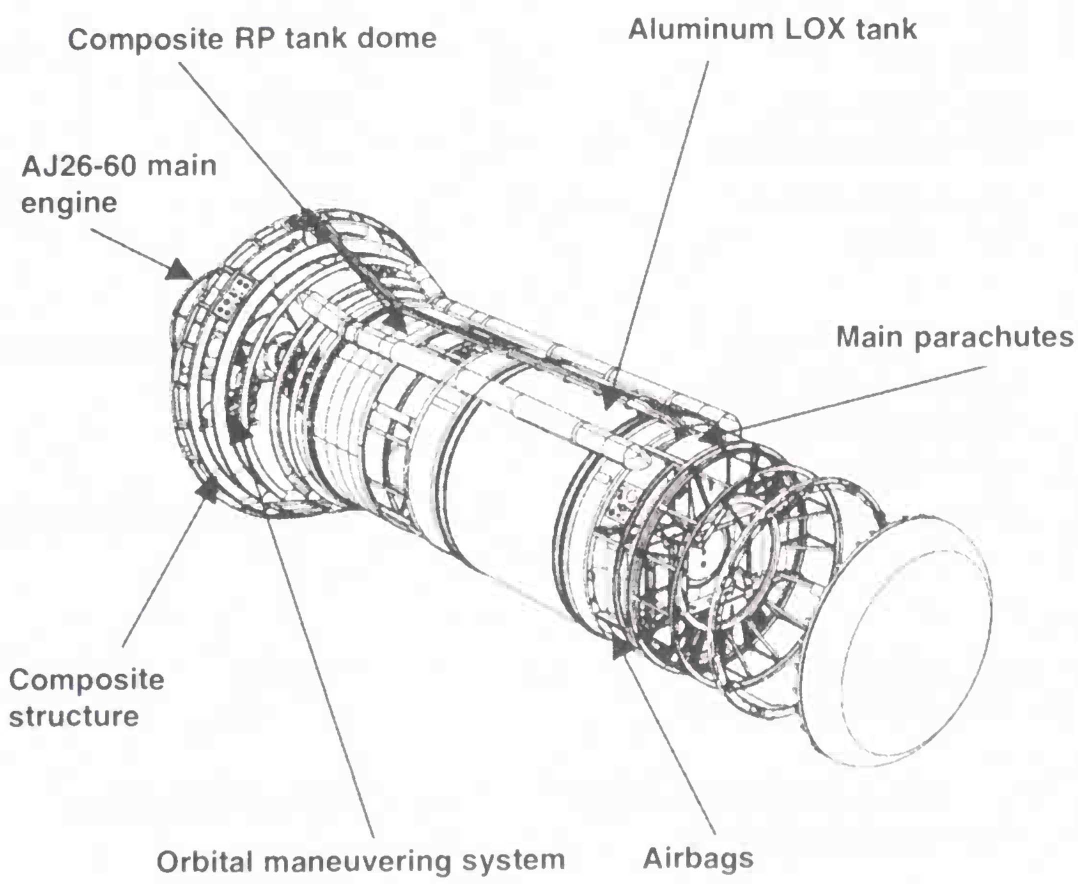
Courtesy Kistler Aerospace Corporation.

K-1 Launch Assist Platform

The K-1 includes a new Orbital Maneuvering System (OMS) that burns LOX/ethanol propellants. It is based on development work done at Aerojet in the 1980s intended to create a nontoxic propulsion system to replace the Space Shuttle ACS engines. Typical systems such as the Space Shuttle ACS engines use UDMH/N_2O_4 propellants, which are highly toxic and therefore the engines are expensive and time consuming to refurbish and maintain between flights. LOX/ethanol was selected for the K-1 because it is nontoxic and easier to maintain than UDMH/N_2O_4 systems. The K-1 OMS system consists of two spherical propellant tanks (one LOX, one ethanol) in the aft compartment of the OV second stage, and one electromechanically gimbaled OMS engine. The engine is fluid-film cooled, with a columbium combustion chamber and nozzle. An electrical igniter burning ethanol and gaseous oxygen is used to ignite the engines to avoid the maintenance requirements of pyrotechnic or hypergolic igniters. The OMS engine is used for velocity trim at the end of the main-engine burn, for circularization/orbit injection, and for maneuvering into the phasing orbit and reentry trajectory.

Each stage includes a propulsion system, a composite airframe, and propellant tanks. In addition to the two primary propellant tanks, the first stage includes a smaller LOX retention tank in the center of the toroidal kerosene tank. This tank retains the LOX needed to restart the main engine for the return of the first stage to the launch site. The retention tank is required because any remaining LOX in the primary tank would slosh toward the warm top of the tank during the pitch maneuver that turns the vehicle back toward the launch site. The LOX would vaporize at the top of the tank and could not be fed to the engines. The tanks are pressurized with helium stored at 400 bar (6000 psi) in composite overwrapped titanium pressure bottles. Nitrogen is stored in bottles of the same design for purging, valve actuation, and inflating the landing airbags. The stages are separated using the air pressure retained in the interstage area. At liftoff, the interstage contains air at ambient pressure. During flight this air is vented slowly, so that when the stages separate at approximately 43 km (140,000 ft) altitude, the internal air pressure is still about 43 kPa (6.3 psi). When the OV is released, this air pressure pushes the two stages apart, eliminating the need for springs or retro-rockets.

Both stages have recovery systems that consist of drogue and main parachutes for deceleration and airbags to cushion the landing. The parachutes can be reused six times. The airbags are pressurized using nitrogen. The OV is protected from reentry heating by ceramic TPS on the forward surface of the payload module and thermal blankets on areas that experience less heating.



Courtesy Kistler Aerospace Corporation.

K-1 Orbital Vehicle

Vehicle Design

	LAP (Stage 1)	OV (Stage 2)	Active Dispenser
Dimensions			
Length	18.4 m (60.2 ft)	23.6 m (77.3 ft) including bell nozzle	1.5 m (5 ft)
Diameter	6.7 m (22 ft)	4.3 m (14 ft)	?
Mass			
Propellant Mass	227 t (499.7 kblm)	116 t (257 klbm)	Variable
Inert Mass	22.7 t (50 klbm)	12.7 t (28 klbm)	?
Gross Mass	249 t (550 klbm)	129 t (285 klbm) (includes payload module)	4090 kg (9000 lbm)
Propellant Mass Fraction	0.91	0.90	?
Structure			
Type	Tanks: monocoque Thrust frame: truss Airframe: skin-stringer	Tanks: monocoque Airframe: skin-stringer	Struts, cylindrical tank, thrust cone
Material	Tanks: aluminum Thrust frame: composite Airframe: graphite composite	LOX tank: aluminum Airframe and RP tank: graphite composite	Composites
Propulsion			
Engine Designation	AJ26-58 and AJ26-59 (Aerojet, based on ND Kuznetsov NK-33)	Main engine: AJ26-60 (Aerojet, based on ND Kuznetsov NK-43) OMS: No designation (Aerojet)	No designation (Aerojet)
Number of Engines	3 (two AJ26-58, one AJ26-59)	1 main engine + 1 OMS	1
Propellant	LOX/RP (kerosene)	Main engine: LOX/RP (kerosene) OMS: LOX/ethanol	MMH/N_2O_4
Average Thrust (each)	Sea level: 1512 kN (339.9 klbf) Vacuum: 1683 kN (378.3 klbf)	Main engine: 1769 kN (397.7 klbf) OMS: 3870 N (870 lbf)	8960 N (2000 lbf)
Isp	Sea level: 297.2 s Vacuum: 331.3 s	Main engine: 348.3 s OMS: 305 s	?
Chamber Pressure	145.4 bar (2109 psi)	Main engine: 145.4 bar (2109 psi) OMS: 10.3 bar (150 psi)	?
Nozzle Expansion Ratio	27:1	Main engine: 80:1 OMS: 100:1	200:1
Propellant Feed System	Oxidizer-rich staged-combustion turbopump	Main engine: oxidizer-rich staged-combustion turbopump OMS: pressure fed	Pressure fed
Mixture Ratio (O/F)	2.586:1	Main engine: 2.592:1 OMS: 1.5:1	?
Throttling Capability	50–104%	Main engine: 55–104% OMS: None	None
Restart Capability	AJ26-58: none AJ26-59: 1 restart	Main engine: none OMS: 5 firings per flight	Multiple
Tank Pressurization	Helium	Main engine: helium OMS: helium	Helium
Attitude Control			
Pitch, Yaw	Pumped hydraulic gimbal ±6 deg	Main engine: Pumped hydraulic gimbal ±6 deg OMS: electromechanical gimbal ±6 deg	Electromechanical nozzle gimbal ±7 deg
Roll	GOX/ethanol ACS (main engines not used for roll control)	GOX/ethanol ACS	Monopropellant hydrazine ACS
Staging			
Nominal Burn Time	139 s for launch, plus 35 s using center engine only for recovery	Main engine: 233 s OMS: mission dependent	Variable
Shutdown Process	Command shutdown	Main engine: command shutdown OMS: command shutdown	Command shutdown
Stage Separation	Stored air pressure in interstage		V-band clamp
Recovery			
Deceleration	Two clusters of 3 main parachutes each, deployed by two drogue parachutes	3 main parachutes deployed by drogue parachute and stabilization parachute	Not recovered
Landing	Near-vertical landing on airbags	Near-vertical landing on airbags	

Vehicle Design

Attitude Control System

The original NK-33 and NK-43 engines have been modified by Aerojet to support engine gimbaling for attitude control during first- and second-stage main engine burns. The primary changes include modifications to the thrust structure, addition of a new gimbal block, and new TVC actuators. The hydraulic actuators are powered by high-pressure fuel tapped off the fuel turbopump outlet. The total gimbal range is ±6 deg. The second-stage OMS engine is also gimbaled using electromechanical actuators with a range of ±6 deg. Each stage has an independent ACS system (the first stage requires its own for recovery maneuvers). The thrusters are hot gas bipropellant engines that burn GOX/ethanol.

Avionics

The K-1 has two unique requirements that affect the design of the avionics systems. First, since both stages must be capable of independent flight, they must each have their own control systems. Second, the OV second stage must power down for approximately 18 h while in orbit to conserve power. Therefore, the system must be capable of powering up on orbit. Like other reusable launch systems, the K-1 must also be highly reliable to ensure that it can be reused over a large number of flights.

Each stage of the K-1 has its own independent guidance and control system. Because high reliability is needed, the guidance and control systems are triple-redundant and fault tolerant. Navigation data are provided by three integrated Embedded GPS/INS (Global Positioning System/Inertial Navigation System) devices called EGIs in each stage that combine both GPS receivers and Honeywell inertial measurement units. For the first stage, only the inertial data are used in the navigation solution. On the second stage, the GPS measurements are blended with the inertial data for a combined navigation solution. While a few other space launch systems are beginning to use GPS for tracking, the K-1 will be one of the first to use GPS for navigation in the control system. To save power during the long on-orbit phase before reentry, the avionics, with the exception of the inertial measurement unit, are turned off after payload deployment.

The flight computing and command functions are divided between redundant vehicle management computers (VMCs), subsystem management unit (SMU), a power distribution unit (PDU), and a payload module controller (PMC). The VMCs perform the guidance, navigation, and control computations, provide for communications and redundancy management, and command the propulsion and landing subsystems. The VMCs are organized as a redundant system with three identical, synchronized flight computers, each running the same software. Voting among the three channels detects computer faults, and only one functioning channel is required to perform the flight. The VMCs communicate with the engine controller units on the main engines to implement commands. Controls and power switching for most other subsystems are implemented through communications with the SMU, PDU, and PMC, which serve as the direct input/output interfaces with these subsystems. For example, the PMC performs low-level control of the payload deployment motors or airbag inflation systems. All flight software for the VMCs, SMU, PDU, and PMC is fully tested in a hardware-in-the-loop simulation at C.S. Draper Laboratory before flight.

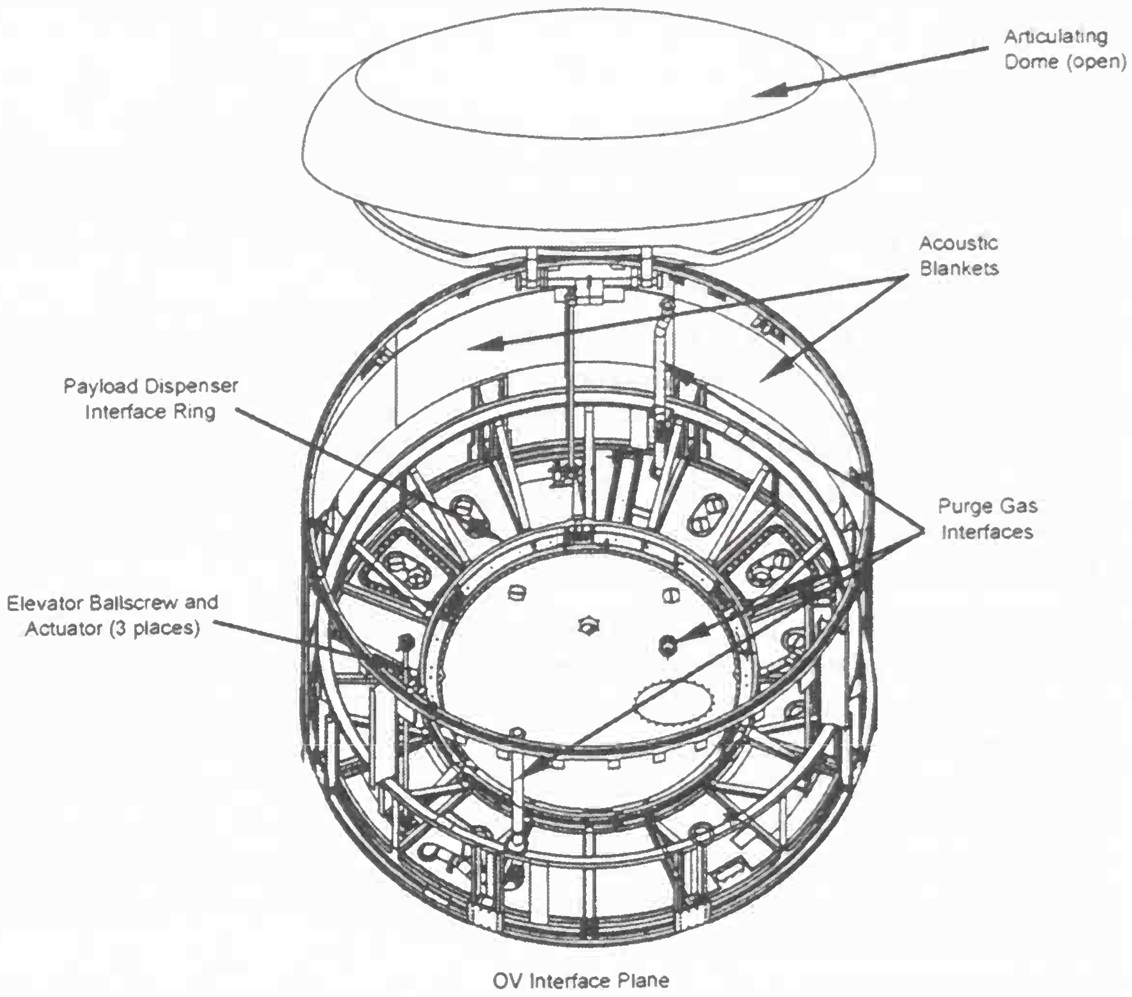
During flight the various subsystems communicate using three redundant MIL-STD-1553B data buses. Each of the three VMC channels controls one bus and monitors the others. The K-1 can transmit telemetry to the ground using the Tracking Data Relay Satellite System (TDRSS). The vehicle also has an FAA standard transponder. On the ground, before launch or after landing, each stage communicates with the ground-based mission control computer (MCC) through RS-422 serial data connections. The VMCs are capable of storing flight data in nonvolatile memory for download to the MCC after landing.

Power for the K-1 is provided by rechargeable batteries that remain inside the vehicle and are charged by ground-based charge control systems before launch. Each stage uses three lithium-ion batteries to power the primary electrical system and avionics, control motors and valves, and pyrotechnic systems.

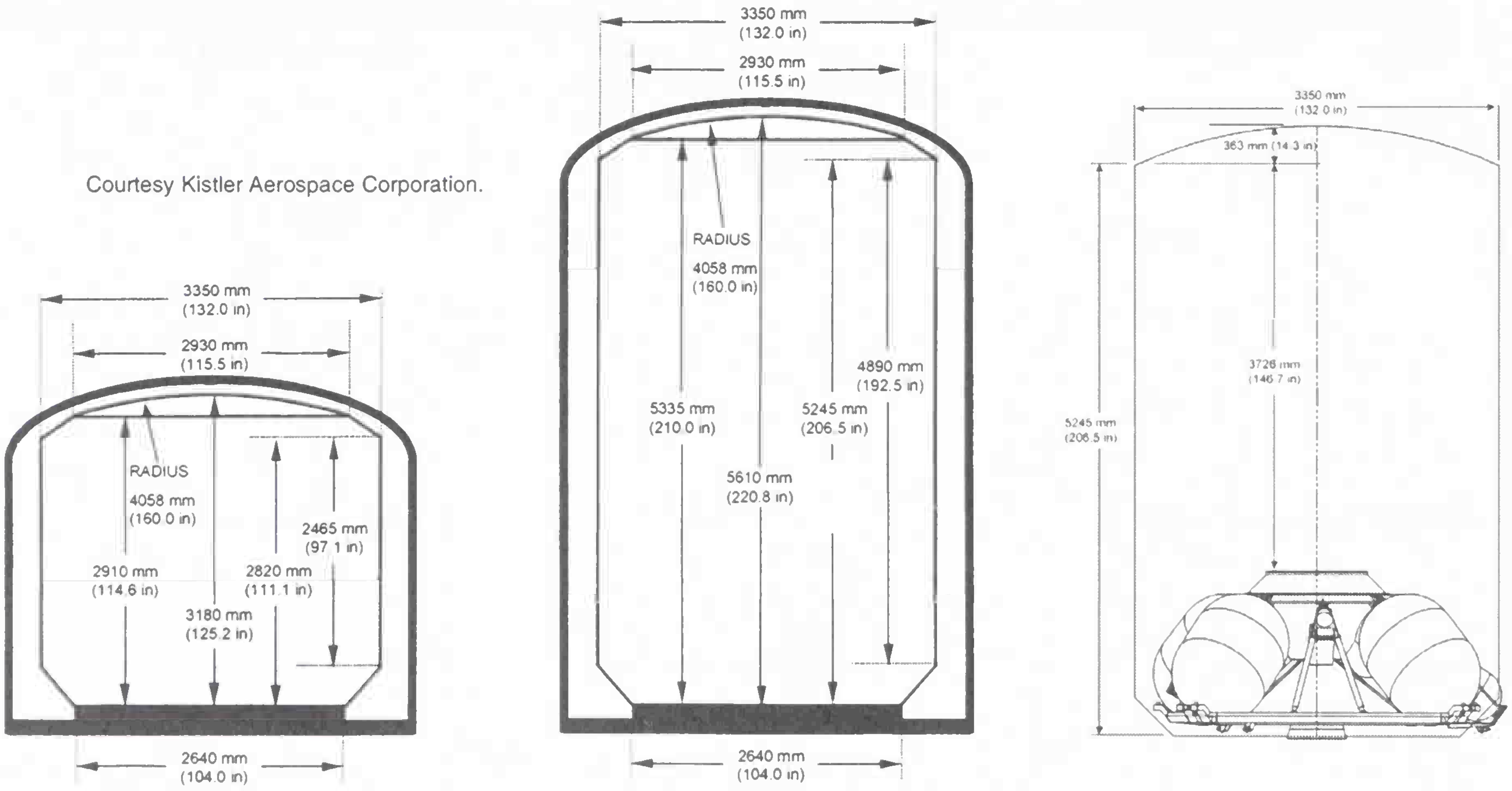
Vehicle Design

Payload Module

Like a conventional payload fairing, the function of the K-1 Payload Module (PM) is to enclose and protect the payload. However, its design differs significantly from typical fairings. The PM is a separate element of the launch vehicle, which completely encloses the payload. It is removed from the launch vehicle after landing to support processing and encapsulation of the next payload offline from the rest of the launch vehicle preparations. The forward dome of the PM has a blunt shape and is covered with thermal protective materials because it serves as the forward surface of the second stage during reentry. Unlike a conventional fairing the PM does not split open and separate from the vehicle to expose the payload during flight. Instead, the forward dome swings open on an articulated hinge to allow payload deployment, then swings back into place and latches shut for reentry. Single payloads can be deployed axially out of the fixed cylindrical section of the PM. Single or multiple payloads that are designed to be deployed radially can be pushed forward on an elevator plate to clear the cylindrical section, then deploy radially. When used, the payload elevator is driven by three electrically redundant electromechanical ball-screw actuators. Four interchangeable payload modules are available. The standard PM (SPM) can hold payloads approximately 3 m (10 ft) tall, while the extended PM (EPM) is roughly twice as tall. To maintain controllability during reentry, the upper half of the EPM retracts over the lower half after payload deployment, making it roughly the same height as the SPM. Acoustic blankets and environmental control are standard features of both PM designs. For missions with the expendable upper stage, the Active Dispenser Payload Module is used, which shares its mold like and retractable fairing with the EPM. A Cargo Module (CM), which has the same dimensions as the SP, is available to support ISS resupply missions. The CM is equipped with special sensor arrays for ISS rendezvous, a cold-gas ACS, and a grapple fixture and common berthing mechanism (CBM) for berthing operations with the ISS. It includes a 30-m^3 pressurized container for transportation of cargo to and from the ISS.



Courtesy Kistler Aerospace Corporation.
Standard Payload Module



Courtesy Kistler Aerospace Corporation.

	Standard Payload Module	Extended Payload Module	Active Dispenser
Length	3.5 m (11.5 ft)	5.9 m (19.2 ft)	3.7 m (12.2 ft)
Primary Diameter	4.3 m (14 ft)	4.3 m (14 ft)	3.4 m (11 ft)
Mass	1383 kg (3043 lbm)	1689 kg (3716 lbm)	1471 kg (3237 lbm)
Sections	1	1	1
Structure	Honeycomb	Honeycomb	Honeycomb
Material	Composite	Composite	Composite

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	3350 mm (132.0 in.)
Maximum Cylinder Length	SPM: 2465 mm (97.1 in.) EPM: 4890 mm (192.5 in.) ADPM: 3726 mm (146.7 in.)
Maximum Cone Length	445 mm (17.5 in.) total height above and below full-width volume
Payload Adapter Interface Diameter	2300 mm (90.55 in.)
Payload Integration	
Nominal Mission Schedule Begins	Approximately 15 months from contract to launch for a first launch, depending on dispenser; as little as 3 days for repeat launches.
Launch Window	
Last Countdown Hold Not Requiring Recycling	T–20 min
On-Pad Storage Capability	Not applicable—not stored on pad
Last Access to Payload	T–24 h
Environment	
Maximum Axial Load	+6.6 *g*
Maximum Lateral Load	±2 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz lateral/25 Hz axial
Maximum Acoustic Level	128.5 dB at 100 Hz
Overall Sound Pressure Level	138 dB
Maximum Flight Shock	Dependent on separation system
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	Payload module opens in orbit. Heating is near zero, depending on orbit altitude.
Maximum Pressure Change in Fairing	–3.7 kPa/s (0.54 psi)
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±10 km (5.4 nmi), ±0.05 deg ± 0.01 eccentricity, ±0.04 deg RAAN
Attitude Accuracy (3 sigma)	?
Nominal Payload Separation Rate	Typically 0.5 to 1 m/s (1.6–3.3 ft/s) depending on separation system
Deployment Rotation Rate Available	Dispenser dependent
Loiter Duration in Orbit	23 h
Maneuvers (Thermal/Collision Avoidance)	Yes, including complete deorbit
Multiple/Auxiliary Payloads	
Multiple or Comanifest	The K-1 will be capable of carrying multiple spacecraft and can provide either axial or radial deployment.
Auxiliary Payloads	Kistler is interested in carrying small payloads on missions with surplus capability. Dispensers for small spacecraft are under consideration. Rideshare opportunities are available on early K-1flights and in conjunction with NASA Space Launch Initiative missions. Contact Kistler for more information.

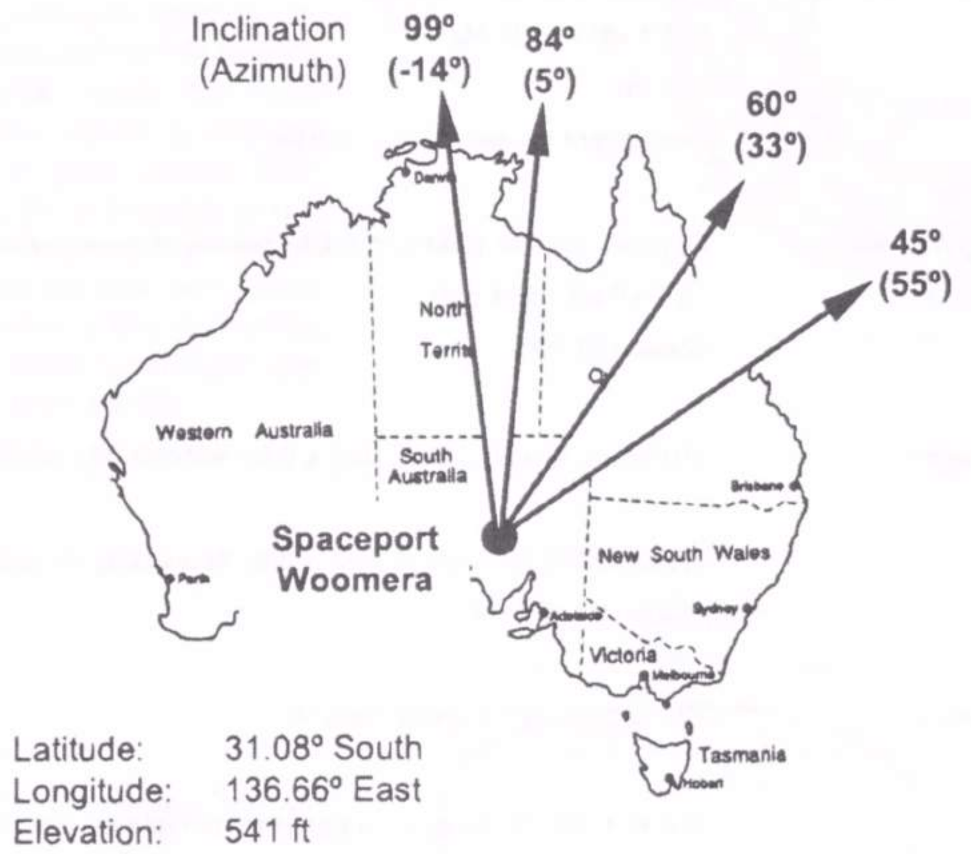
Production and Launch Operations

Production

Kistler Aerospace Corporation performs overall design, systems engineering, and operation of the K-1 launch vehicle. Detailed design and manufacturing are contracted to a number of established aerospace companies, each of which has relevant experience in the subsystems for which they are responsible.

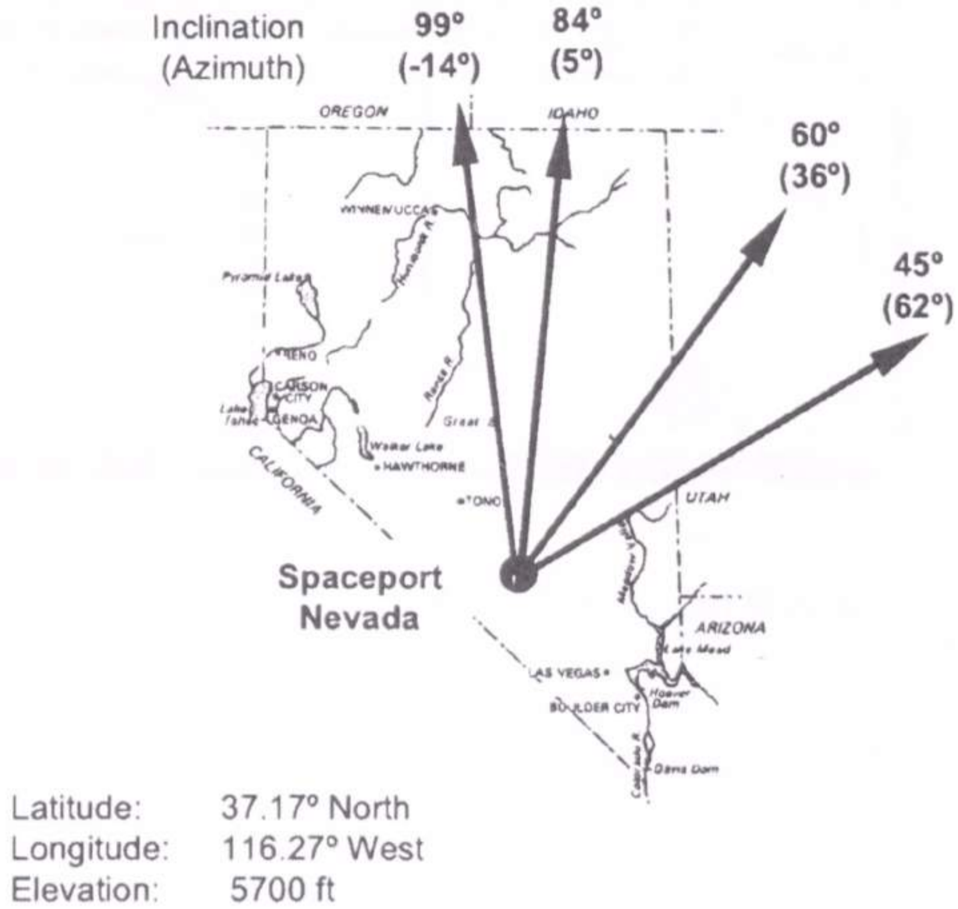
Organization	Responsibility
Kistler Aerospace Corporation	Systems engineering, program management, vehicle assembly, and operations
Aerojet	A-526-58/59/60 main engines, OMS, and subsystems
Lockheed Martin Michoud Space Systems	Aluminum tanks
Northrop Grumman Corporation	Composite RP tank and vehicle structure
Charles Stark Draper Laboratory	GNC system and flight software
Honeywell	Vehicle management system and flight computer
Irvin Aerospace, Inc.	Parachutes and airbags
Oceaneering Space Systems	Thermal protection system

Launch Facilities



Courtesy Kistler Aerospace Corporation.

Australia Launch Site, Spaceport Woomera



Courtesy Kistler Aerospace Corporation.

Nevada Launch Site, Spaceport Nevada

Production and Launch Operations

Launch Operations

The K-1 will be launched from Woomera, Australia. A second launch site is planned at the Nevada Test Site near Las Vegas, Nevada, in the United States. Both sites offer large, flat areas for parachute recovery of the stages, and low population density so that the vehicle can fly over land. Spaceport Woomera will be activated first. Launch operations from Woomera are described below and operations in Nevada will be similar.

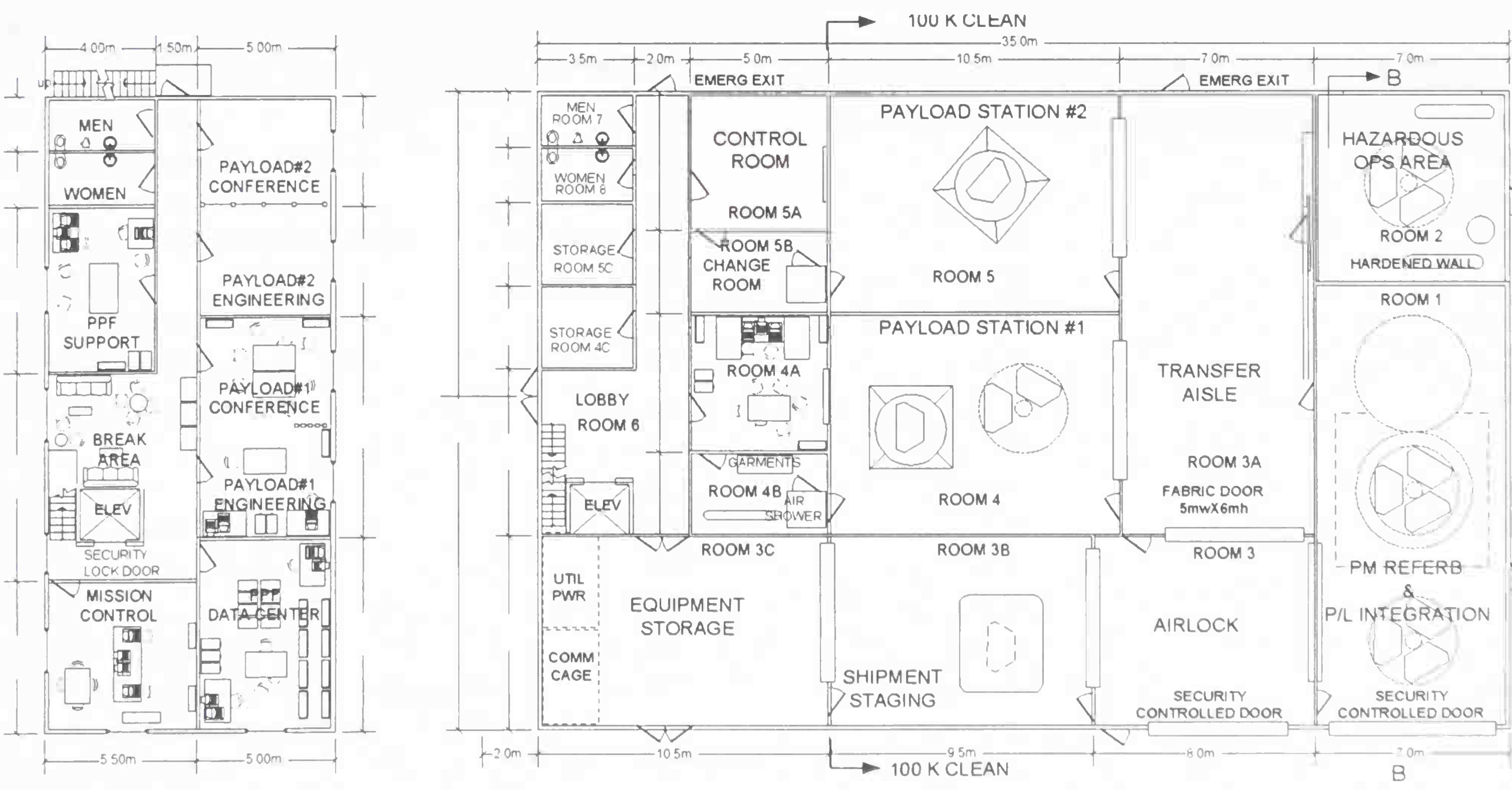
The K-1 will initially be launched from a new launch complex in the Woomera Prohibited Area, in South Australia, roughly 470 km (280 mi) north of Adelaide. Woomera was originally developed in 1946 as a joint British and Australian test site for long-range missiles, sounding rockets, and, eventually, of space launch vehicles. In 1967, Australia launched its first satellite from Woomera using a SPARTA rocket, a modified U.S. Redstone. Several failed tests of the joint European Europa space launch vehicle took place at Woomera from 1968–1970. In 1971 a British Black Arrow launch vehicle placed the Prospero satellite in orbit. However, following the cancellation of the Black Arrow program, the United Kingdom abandoned Woomera in 1976. With the exception of sounding rocket launches and support of Japanese reentry vehicle tests, there has been little space-related activity at Woomera since.

The Kistler facilities at Woomera are located along the Koolymilka–Woomera road at Ashton Hill, about 19 km (12 mi) northwest of the village of Woomera. Kistler has leased approximately 30 km^2 (12 mi^2) from the government of Australia. Within Spaceport Woomera are three primary areas: a 1.8-km (1.1-mi) diam flat, open area reserved for parachute landings of the returning stages, a payload processing facility and vehicle processing facility along the road, and the launch complex, 2.4 km (1.5 mi) down the road from the vehicle processing facility.

Launch operations begin when the vehicle stages arrive at the launch site. They can be transported by existing wide-body specialty cargo aircraft from the factory, but more commonly they will be returning from a previous flight and will land in the landing area using parachutes and airbags. The stages provide confirmation that they have been safed, and then can be approached by ground crews for recovery operations. Vehicle openings are covered to prevent dust contamination, and the airbags and parachutes are removed and packed for shipment back to the supplier for refurbishment. Each stage is lifted using the straddle lift vehicle, a tall, wheeled vehicle that straddles the stages and lifts them with a built-in crane. Each stage is mounted onto handling fixtures on a special tractor trailer truck, and transported horizontally to the vehicle processing facility (VPF) at the launch complex. At the VPF, all processing is done horizontally. The stages are vented and drained of residual fluids and gases, and are vacuum cleaned. The PM is removed from the OV, refurbished, and then moved to the payload processing facility (PPF) for offline payload encapsulation. Stage thermal protection systems are checked and refurbished if needed, and consumables and expended hardware are removed and replaced. Components scheduled for maintenance or that fail automated testing are removed for repair and replaced with fresh units. Finally, new airbags, parachutes, and pyrotechnics are installed.

Spacecraft processing is performed in the Kistler PPF. The facility includes two high bay work areas with an overhead bridge crane. In a separate hazardous processing area room in the same building, the PM is maintained and spacecraft can be fueled. Spacecraft are integrated onto an adapter or dispenser if required, and then lowered into the PM using a bridge crane. The encapsulated PM is then rotated to a horizontal position and transferred to the VPF for integration with the launch vehicle. The PPF building also includes control rooms for spacecraft testing and fueling operations, office space, and the mission control room for the K-1.

Once the stages and PM have been processed, the LAP and OV are aligned horizontally in the VPF and mated. The PM containing the payload is then aligned, connected, and then mated to the OV. At this point a flight readiness review is conducted to ensure that all systems are ready for launch. The assembled K-1 vehicle on its erector structure is transported horizontally on rails from the VPF to the launch stand approximately 2.25 km (1.4 mi) away. Umbilical connections for propellants, power, air conditioning, and gas purging are connected and checked. The vehicle is aligned with the launch mount and erected to the vertical position. Propellant loading is performed automatically under control of the ground control computer (GCC), and both the vehicle onboard health monitoring systems and GCC transmit information to the MCC so that the flight director can monitor vehicle status. After propellant



Level 2
Mission Control Center

Level 1
Payload Processing Facility

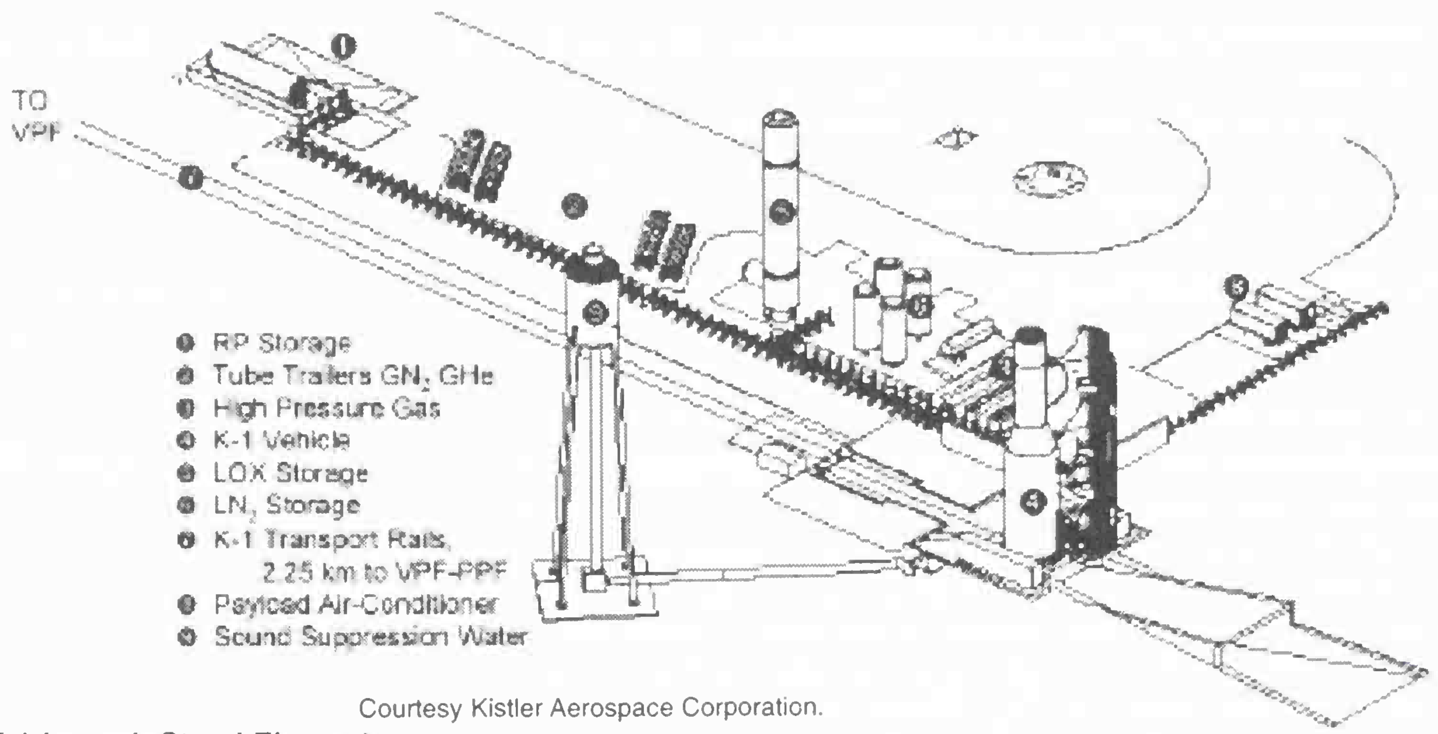
Courtesy Kistler Aerospace Corporation.

K-1 Launch Complex Overview

Production and Launch Operations

is fully loaded, the automated launch sequence begins, and the vehicle onboard computers complete the countdown procedure. Based on a review of all data, the flight director either initiates the launch at the appropriate time or terminates the countdown if necessary. If the launch is scrubbed, the vehicle is drained of propellant, lowered to the horizontal position, and returned to the VPF.

Following liftoff the K-1 rolls to the desired flight azimuth, pitches over, and follows a lofted trajectory during first-stage flight that reduces the propulsive requirements for the LAP to return to the launch site. Staging conditions are carefully controlled to occur at a predefined staging velocity. The LAP engines throttle down to 55%, then shut down. Attitude is stabilized and the two stages are separated using residual pressure in the interstage compartment. The LAP flips over, using its ACS thrusters assisted by the offset center of gravity of the stage. After the OV has cleared a safe distance, the LAP center engine restarts, finishes the pitch around maneuver, and provides enough thrust to put the stage on a ballistic trajectory back toward the launch site. After engine shutdown, the LAP is again flipped over so that engines are pointed forward. The first drogue parachute deploys at an altitude of approximately 7500 m (25,000 ft) above sea level. The six main canopy chutes deploy at approximately 4500 m (15,000 ft). The airbags inflate using pressurized nitrogen, and the ACS thrusters align the LAP's longitudinal axis with its velocity vector so that the vehicle will not roll over when it hits the ground. At impact, the airbags deflate, and the parachutes are released. All systems are safed and powered down, and the stage is retrieved by the ground crew to begin preparations for the next flight.



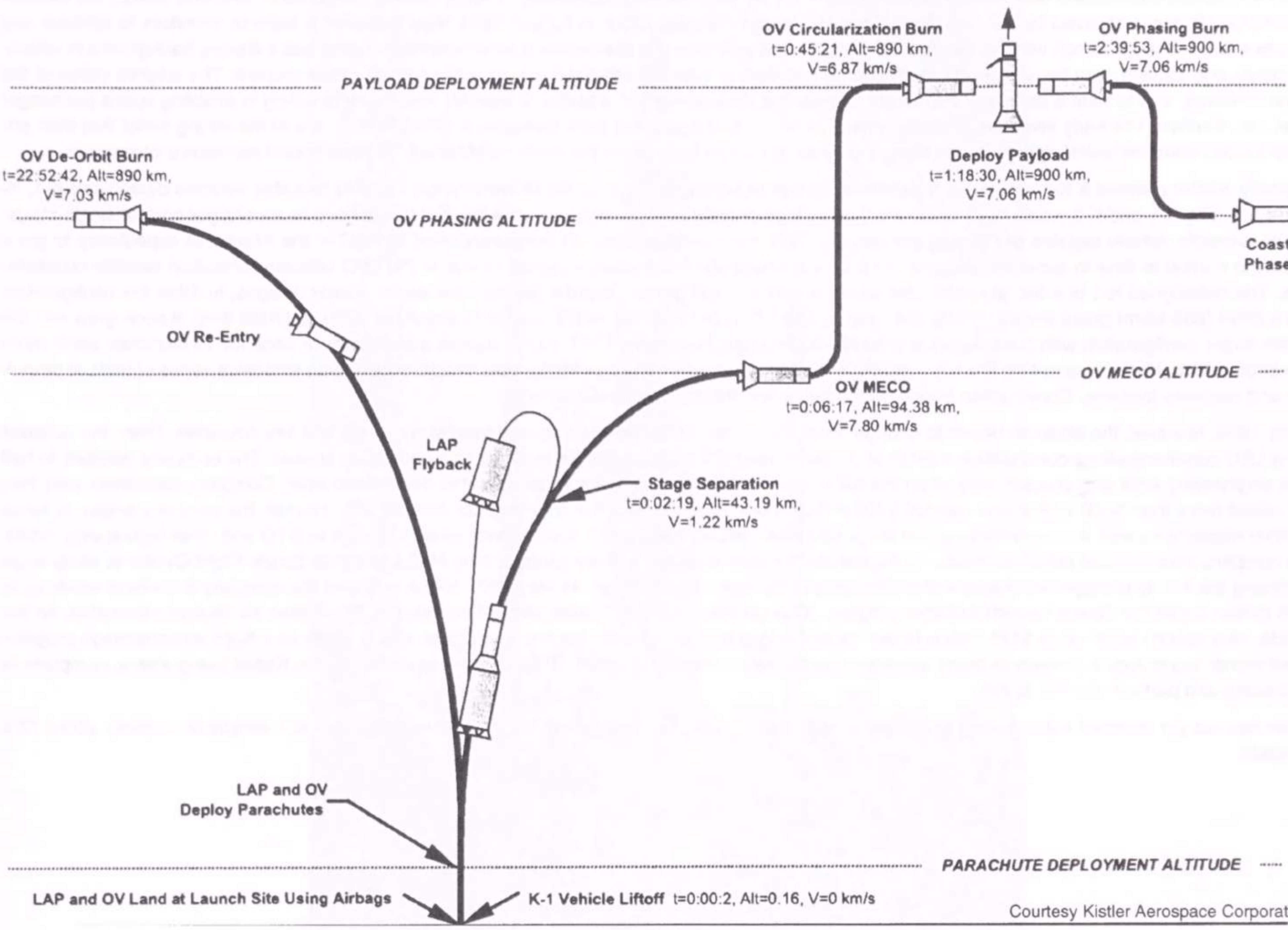
Courtesy Kistler Aerospace Corporation.

K-1 Launch Stand Elements

Meanwhile, the OV second-stage main-engine burn achieves the desired elliptical transfer orbit, with apogee at the target orbit altitude. The main engine throttles down to maintain acceptable acceleration loads, then shuts down at the desired cutoff conditions. Following burnout, residual main engine propellants are dumped. If necessary, a short OMS burn corrects velocity dispersions caused by main engine burnout dispersions or the fuel dump. The OV then coasts to apogee, where an OMS burn circularizes the orbit at the desired altitude. The ACS system orients the vehicle for payload deployment, and the top of the PM is opened to release the payload. Once the payload has separated to a safe distance, the OV performs a phasing orbit maneuver using the OMS engine. This orbit has its perigee above the landing site latitude, and an orbital period that ensures the ground track will pass over the landing site, rather than off to one side, when the Earth's rotation lines up the landing site with the OV orbital plane approximately 24 h later. After the maneuver is completed, the vehicle goes to sleep to conserve power. It wakes up once, halfway through the on-orbit phase, to correct the phasing orbit if necessary, then goes back to sleep. Before reentry, the vehicle again wakes up, reinitializes GPS and INS navigation systems, and executes a deorbit burn half an orbit before landing. Winds aloft data collected with weather balloons are transmitted from mission control to the vehicle so that the onboard guidance can correct for winds. The OV has an offset center of gravity, allowing the OV to be steered using roll control as was done for the Apollo spacecraft. At Mach 2.5 at an altitude of 25,000 m (81,000 ft) the stabilization chute is deployed, which maintains the stability of the OV until the drogue chutes are deployed at 8000 m (27,000 ft). Main chutes deploy at 4500 m (15,000 ft), and landing follows the same process described above for the LAP.

Production and Launch Operations

Flight Sequence



Courtesy Kistler Aerospace Corporation.

Mission event sequence for 900 km (486 nmi) orbit with launch from Woomera

Event	Time, s	Altitude, km	Earth-Relative Velocity, km/s
K-1 arrival at pad	–23,400	0.16	0.00
Launch vehicle raised to vertical	–15,000	0.16	0.00
Automatic pad checkout	–14,400	0.16	0.00
Vehicle chilldown and propellant loading	–13,500 to –4800	0.16	0.00
Final checkout	–3600	0.16	0.00
Vehicle switch to internal power	–75	0.16	0.00
Engine preignition sequence	–45	0.16	0.00
LAP (first-stage) engine start	0	0.16	0.00
Liftoff	2	0.16	0.00
Begin pitchover	11	0.32	0.03
Maximum dynamic pressure	82	11.74	0.39
Begin main-engine throttle down	134	38.61	1.18
Stage separation	139	43.19	1.22
OV (second-stage) engine starts	146	47.93	1.18
Begin g-limiting throttle of second-stage engine	345	94.29	5.96
OV main engine cutoff (MECO)	377	94.38	7.80
OMS velocity trim burn	437	98.29	7.79
OMS circularization burn	2721	890.00	6.87
Payload module opening	4320	900.00	7.06
Payload deployment	4710	900.00	7.06
Orbit phasing burn	9593	900.00	7.06
Deorbit burn	82362	890.00	7.03

Vehicle History

Historical Summary

Kistler Aerospace Corporation was formed in November 1993 by Walter Kistler, cofounder of Kistler-Morse Corporation, and Bob Citron, the founder of SPACEHAB, Inc. Catalyzed by the first flight of the McDonnell Douglas DC-X in August 1993, they gathered a team of investors to develop and operate a fully reusable launch vehicle, incorporating proven technologies in a low-cost and reliable system. Citron has a lifelong background in adventure travel and tourism, and he, along with the founding investors in Kistler Aerospace, share a passion for space tourism. The original vision of the Kistler investors was to form a company that would pioneer the development of a family of vehicles eventually resulting in enabling space passenger travel, i.e., tourism. The early investors in Kistler (many of whom had also been early investors in SPACEHAB), are of the strong belief that their privately funded entrepreneurial approach can bring a product to market faster than the more established, tradition-bound aerospace companies.

Originally, Kistler planned a K-0 vehicle as a pathfinder to test technologies for a series of increasingly capable reusable vehicles designated K-1, K-2, and K-3. The K-1 and K-2 would have been small, two-stage reusable cargo vehicles, while the K-3 would have been a larger conical single-stage-to-orbit reusable vehicle capable of carrying passengers. This step-by-step approach was abandoned in 1995 in the interest of expediency to get a product to market in time to serve the deployment and operations and maintenance launch needs of the LEO telecommunication satellite constellations. The redesigned K-1 is a two-stage-to-orbit reusable vehicle, with greater capabilities than the earlier Kistler designs. In 1996 the configuration was a 285-t (635-klbm) gross weight vehicle powered by five RD-120M engines, with a payload capacity of 3200 kg (7000 lbm). It soon grew into the current larger configuration with three Americanized NK-33 engines. In January 1997, Kistler signed a contract with Loral for 10 launches worth more than $100 million. Kistler was raising financing rapidly and concurrently proceeding with detailed design work and an extensive series of tests of propulsion and recovery systems. Construction began on the tanks and major structural elements.

In late 1998, however, the situation began to change. First, the Asian economic crisis slowed fundraising in several key countries. Then, the collapse of the LEO communications constellation market shut down many of Kistler's potential customers. Fundraising slowed. The company decided to halt most engineering work and proceed only when the full remaining funding was in hand, so as not to overextend itself. Company executives said they had raised more than $500 million and needed $400 million more. As a result of the shrinking commercial LEO market, the company began to focus on other missions as well. An expendable upper stage was designed so that the K-1 could launch small payloads to GTO and other high-energy orbits. The company also pursued NASA contracts. In August 2000 Kistler received a study contract from NASA Marshall Space Flight Center to study ways of utilizing the K-1 to resupply the space station, carrying cargo both up and down. In May 2001 NASA awarded the company a contract worth up to $135 million under the Space Launch Initiative program. This contract provided Kistler with a firm value of $10 million for design information on the vehicle, plus options worth up to $125 million to test 13 embedded technologies on the first four flights, and to serve as a flight demonstration program for additional technology experiments being developed separately in the SLI program. The money was contingent on Kistler being able to complete its fundraising and perform the first flights.

Kistler has not yet obtained full financing to complete its launch system, but fundraising efforts continue. The first K-1 vehicle is currently about 75% complete.

KAITUOZHE

The Kaituozhe is a new family of small, solid propellant launch vehicles developed by China. It is based on the DF-31 ICBM. This photograph has appeared on various web sites and appeared in CASIC materials.

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KAITUOZHE KT-1

GENERAL DESCRIPTION

Kaituozhe KT-1

Summary

The KT-1 is the first member of a new family of small launch vehicles based on the DF-31 solid-propellant ICBM. It is also the first Chinese launch vehicle not related to the Long March family. At the time of publication, China has released very little information on this launch system.

Status

In flight testing. First launch in 2002.

National Origin

China

Key Organizations

Marketing Organizations	Not yet marketed
Launch Service Provider	Space Solid Fuel Rocket Carrier Co. Ltd.
Prime Contractor(s)	China Aerospace Industry Corporation?

Primary Missions

Chinese government and university microsatellites

Estimated Launch Price

?

Space Port

Launch Site	Taiyuan Satellite Launch Center (TSLC)
Location	37.8° N, 111.5° W
Available Inclinations	?

Performance Summary

The KT-1 payload capability is approximately 100 kg (220 lbm) to unspecified polar orbits with a maximum capability around 300 kg (660 lbm) for lower inclination orbits

Flight Record (through 31 December 2003)

Total Flights	2
Successes	0
Partial Failures	0
Failures	2

Flight Rate

?

Nomenclature

The word Kaituozhe is commonly translated as explorer or trailblazer. The first vehicle in the family is designated KT-1. When first announced, it was also designated SLV-1, but this designation is no longer used. The Kaituozhe family is derived from the DF-31 (Dong Feng, East Wind) missile, called the CSS-9 by NATO.

Cost

No information on the cost of the Kaituozhe program is available. However, it appears that the program is structured and partially funded in a commercial manner with investments by the industrial partners. This would be a change from China's traditional government-managed Long March program.

Availability

When the Kaituozhe program was first announced, Space Solid Fuel Rocket Carrier Co. Ltd. expressed the intent to market launch services commercially. However, no significant effort has been made to market or publicize the vehicle outside of China and no point of contact for business development is advertised. It is possible that commercialization will be pursued after the vehicle has been successfully demonstrated, but it is also possible that marketing plans have been cancelled in response to security concerns or the collapse of the small satellite market.

Performance

No detailed performance data is available. Press reports suggest that KT-1 can carry 100 kg (220 lbm) to polar orbits, and up to 300 kg (660 lbm) to lower inclination orbits.

Flight History

In addition to the orbital launch attempt shown below, a suborbital launch of a DF-31 type vehicle occurred in January 2002. Some reports have suggested that this was a test launch using the KT-1 configuration; this can not be confirmed. The third stage exploded in flight, but this may have been intentional.

Orbital Flights Per Year

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Successes	0
Partial Failures	0
Failures	2

		Date (UTC)	Launch Interval (Days)	Model	Vehicle Designation	Site	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit	Market	Country
F	2002	Sep 15		KT 1		T		2002 F01A	Unknown	50	LEO	MIL	China
F	2003	Sep 16	366	KT 1	KT 1-2	T		2003 F01A	PS 2	40	SSO	MIL	China

T = Taiyuan Satellite Launch Center

Failure Descriptions

F	2002	Sep 15	KT 1	2002 F01	The second stage is reported to have failed.
F	2003	Sep 16	KT 1	2003 F01	Failed to reach orbit.

Vehicle Design

Height	~20 m (65 ft)
Gross Liftoff Mass	~50 t (110 klbm)
Thrust at Liftoff	?

Stages

Very little technical information is available on the KT-1 design. It consists of three solid propellant stages. The first stage has a 2.0 m (6.6 ft) diameter, while upper stages are smaller. The first stage has four nozzles. The first two stages are taken directly from the DF-31 design, while the third stage was developed or modified for KT-1. The third stage may be related to a new third stage believed to be under development for an upgraded missile called the DF-31A or for the DF-41. An inertial guidance system is used.

Production and Launch Operations

Production

The Space Solid Fuel Rocket Carrier Co. was created to develop the Kaituozhe launch system. Several organizations participated in the creation of SSRC and the development of KT-1, including China Machinery & Electronics Engineering Integrated Design Department, China Space Machinery & Electronics Co., the Space Solid Fuel Rocket Propulsion Technology Research Institute, Controls & Electronics Technology Research Institute and the Chenguang Co. Ltd.

Launch Operations

No details of KT-1 launch operations are available. The DF-31, on which the KT-1 is based, is a road-mobile, truck-launched ICBM, and initial descriptions of the KT-1 indicated that it, too, could be launched by a mobile launching system. A photo of the first KT-1 shows it mounted on a conventional launch stand with two small electrical umbilical towers.

Vehicle Upgrade Plans

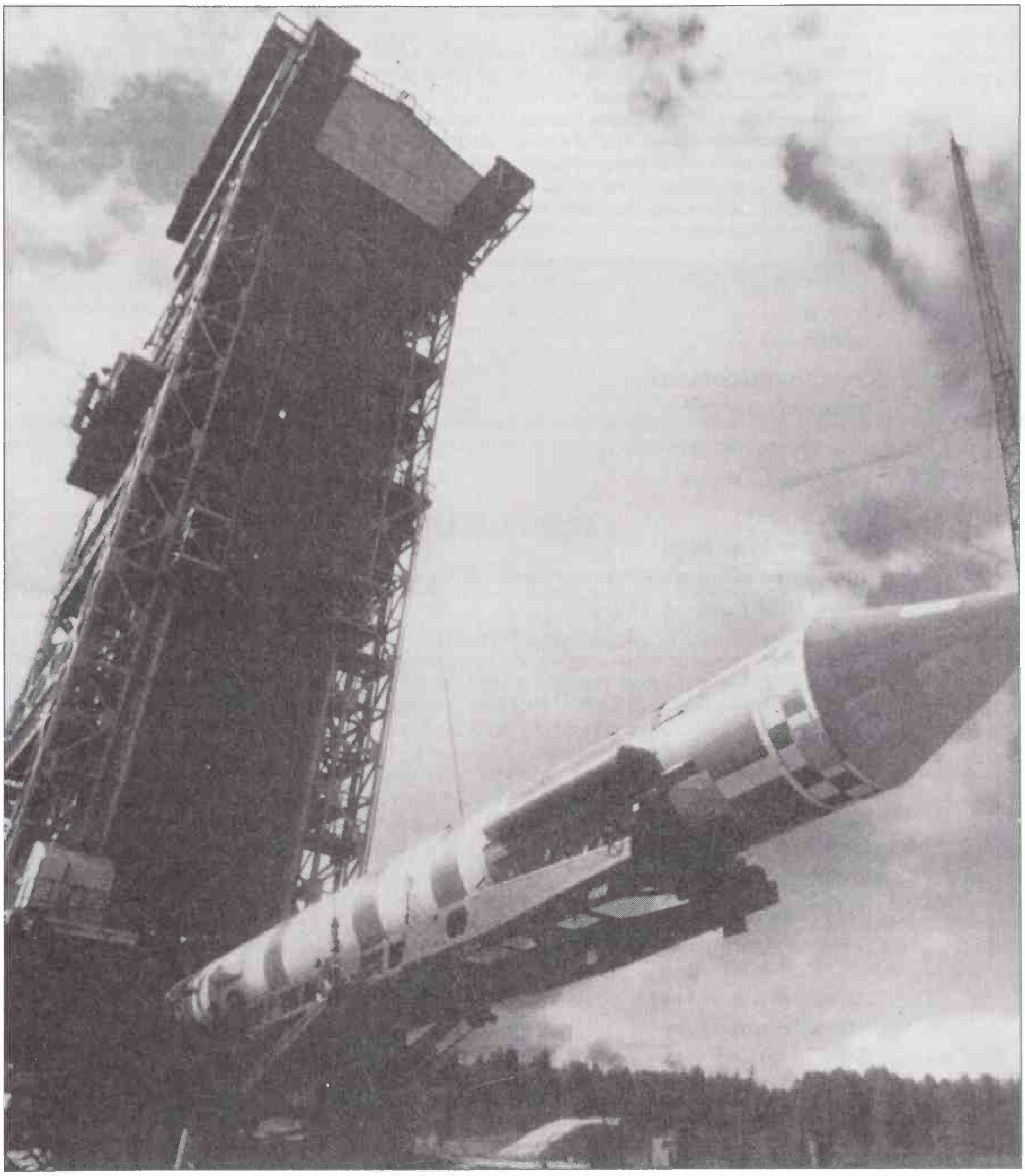
SSRC has expressed plans to develop seven solid motors for three launch vehicles. At the China International Aviation & Aerospace Exposition 2002 in Zhuhai, models of the KT-1 and future members of the Kaituozhe family were displayed. The exhibit included a larger vehicle designated KT-2 and a KT-2A, which is similar to the KT-2 but with a pair of strap-on motors and larger second stage. No data is available on the performance or schedule of these variants.

Vehicle History

Following China's deployment of its first generation liquid-fueled ICBMs, development of solid-propellant ICBM's began in the 1970s. The DF-31 program began around 1985 in conjunction with the JL-2 (Ju Lang-2 or Great Wave) SLBM project. The first test flight occurred in 1992 but failed. The first successful flight was in 1995, and operational deployment appears to have been initiated around 2000.

The Kaituozhe program was publicly announced in May 2000. Development progressed rapidly, presumably a result of the high degree of commonality with proven ICBM hardware. The vehicle passed a major design review in November, 2000, and the third stage motor was tested in February 2001. The first launch attempt is believed to have occurred in September 2002, resulting in failure. The schedule for future launches is unknown.

Kosmos



Courtesy United Start Corporation.

The Kosmos is a small Russian launch vehicle that historically has been one of the most active in the world. It is now being marketed internationally for commercial payloads.

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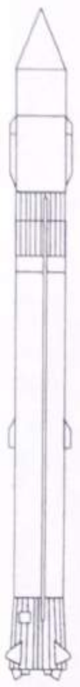
Kosmos 3M

General Description



Kosmos 3M

Summary

Kosmos is a small, two-stage, liquid-fueled launch vehicle that has been used to launch Soviet and Russian military payloads to LEO since the early 1960s. It is the only space launch vehicle to have flown from all three of the former Soviet launch centers—Baikonur, Plesetsk, and Kapustin Yar. The Kosmos 3M is the operational version of the Kosmos, and has been in use since 1967. Historically, Kosmos has had a very high flight rate, with more than 400 orbital launches and more than 300 suborbital missions. Use of Kosmos declined after the fall of the Soviet Union, but has stabilized with a mix of Russian government and foreign commercial payloads. Kosmos is now being marketed commercially to foreign customers, and several organizations in the United States and Europe have signed contracts for launch services. United Start Corporation, based in the United States, is the official marketing agency for non-Russian payloads.

Status

Operational. First launch in 1967.

Origin

USSR/Russia

Key Organizations

Marketing Organizations	United Start (International) Puskovie Uslugi (Russia) Cosmos International (Europe)
Launch Service Provider	Puskovie Uslugi, AKO Polyot
Prime Contractor	AKO Polyot (manufacturer)

Primary Missions

Small Russian military and Western commercial LEO satellites.

Estimated Launch Price

$12 million (United Start Corporation, 1999)

Spaceports

Launch Site	Plesetsk LC 132
Location	62.7° N, 40.3° E
Available Inclinations	66, 74, and 83 deg
Launch Site	Kapustin Yar LC 107
Location	48.6° N, 46.6° E
Available Inclinations	50.6 deg

Performance Summary

The standard adapter is accounted for in the performance values.

250 km (135 nmi), 51.6 deg	1500 kg (3300 lbm)
200 km (108 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	1400 kg (3090 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	775 kg (1700 lbm)
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Launches	432
Launch Vehicle Successes	409
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	22

Flight Rate

1–4 per year

Nomenclature

Following standard practice, the name "Kosmos" comes from the name of the Kosmos 38, 39, and 40 payloads carried on the first flight. The spelling "Cosmos" is also common. Under the Soviet nomenclature, Kosmos was designated 11K65M. The "article number," also called the model number or the index number is a unique alphanumeric designator assigned by the Soviet Union, and later by Russia, to each type of long-range missile, space launch vehicle, rocket engine, and spacecraft. The "K65" portion of the article number is carried over from the article number of the original R14 (SS-5 Skean) missile from which the Kosmos is derived. There is a variant of the Kosmos 3M, called the K65M-RB or 65MP that is used for suborbital launches.

The Kosmos name was also used for a different Soviet launch vehicle. Between 1962 and 1977, a small launch vehicle derived from the R-12 IRBM (SS-4 Sandal) was used to launch military payloads. Because Soviet launch vehicles were typically named after their first payloads, the SL-7 is sometimes referred to as "Kosmos," after its Kosmos 1 payload. Because more than 2300 military satellites have used the Kosmos cover name, it is unsurprising that two different launch vehicles would both have a Kosmos payload on their first launch. Despite having the same name, this launch vehicle is not directly related to the Kosmos 3M, although both were originally developed at the Yangel Design Bureau and manufactured at PO Polyot.

Large Russian engineering organizations are typically divided into distinct engineering companies and manufacturing companies. The name PO Polyot refers to the production organization of Polyot, and KB Polyot is a subunit of PO Polyot. There is also a separate Polyot engineering design bureau. AKO Polyot is the name of the collective organization.

Cost

According to United Start Corporation in 1999, the typical price for a commercial Kosmos launch is $12 million. Prices for secondary payloads typically range from $500,000 to $1.5 million, depending on factors such as volume, mass, and integration requirements.

Availability

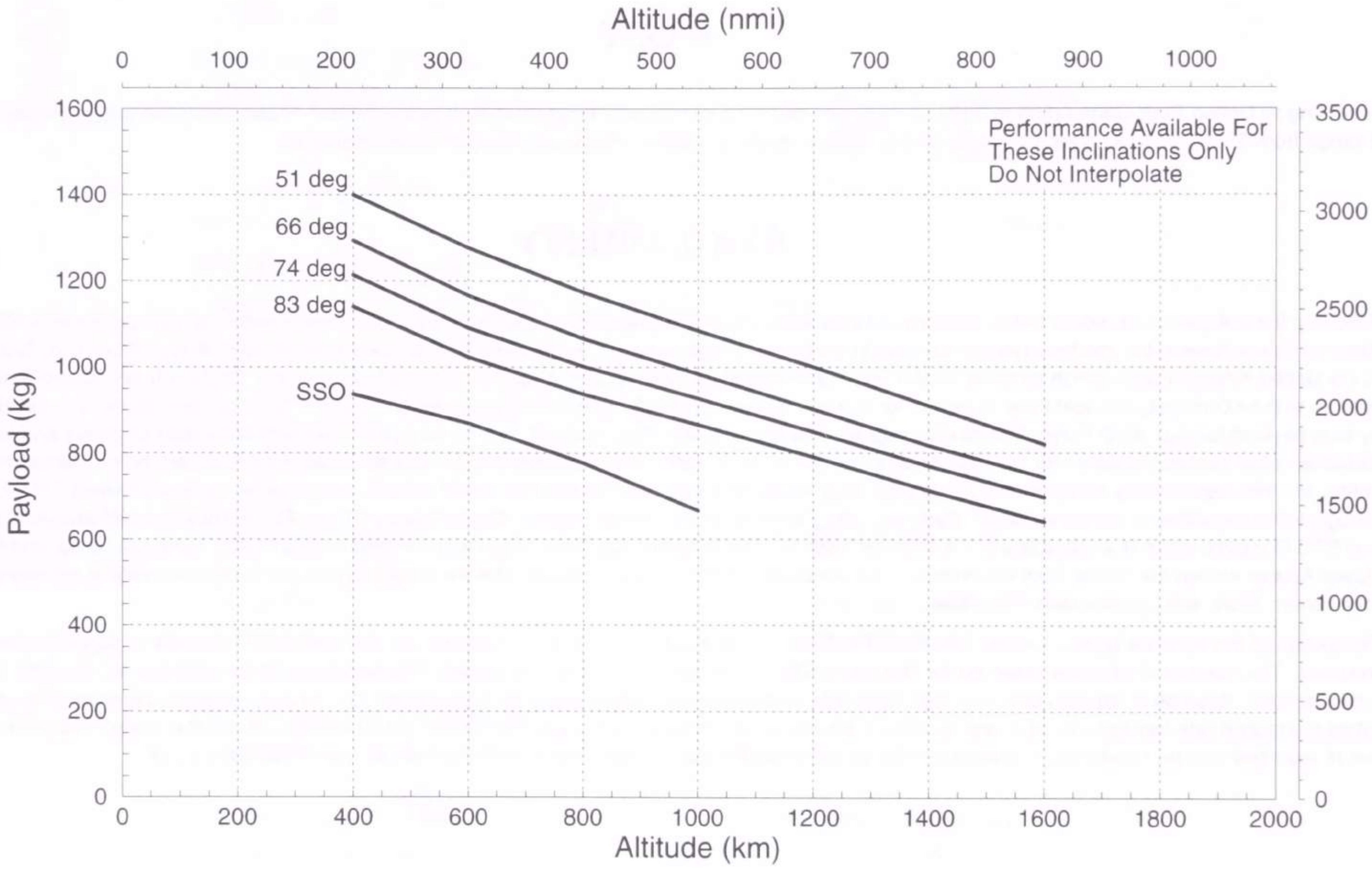
Following the collapse of the Soviet Union, Kosmos vehicles were offered commercially by several companies in cooperation with the producer, AKO Polyot. Assured Space Access, Inc. marketed launch services in the United States as part of the Cosmos USA joint venture with AKO Polyot. Plowshare Technology in the United Kingdom sold launch services for the Swedish microsatellite Astrid. OHB-System in Germany formed the Cosmos International GmbH joint venture with AKO Polyot, and sold three launches for German science payloads. Launches performed on behalf of Cosmos International can be identified by blue payload fairings. AKO Polyot also directly negotiated a contract with Final Analysis, Inc. for launches of several secondary payloads and four dedicated launches carrying satellites for the Final Analysis constellation. In 1995 Assured Space Access was designated the sole partner for international marketing. In 1998 responsibility for the Kosmos and Start small Russian launch vehicles was combined under a new Russian company named ZAO Puskovie Uslugi (which translates to launch services). Puskovie Uslugi is jointly owned by the Russian Space Agency (RKA—Rosiyskoye Kosmicheskoye Agenstvo) and STC Complex, which is a subsidiary of the Moscow Institute of Heat Technology. To market these vehicles internationally, Puskovie Uslugi and Assured Space Access formed the United Start Corporation joint venture in 1998. United Start is based in the United States and is the international marketing agent for Kosmos, Start, and Cyclone launch services.

Production of the Kosmos launch vehicle was halted in 1995, but at least 12 unassigned vehicles are still available in storage to support commercial missions. The number of vehicles reserved for Russian military missions has not been disclosed. Production could be restarted for an order of 10 or more vehicles. Kosmos is capable of a very high flight rate, and has performed as many as 42 launches (orbital and suborbital combined) in one year. While the launch rate has been much lower recently, a maximum rate of 30 launches per year would still be feasible should the market require it. A number of launches can be conducted in a short period, as two boosters can be launched from the same pad only three days apart.

Performance

Kosmos is launched from the Plesetsk Cosmodrome and Kapustin Yar missile range. Because both sites are landlocked, specific impact zones away from populated areas are reserved for jettisoned launch vehicle stages. For Kosmos, these zones are roughly 740 km (400 nmi) downrange. Therefore, Kosmos can only launch along preapproved launch azimuths that allow for safe disposal of the first stage and payload fairing. These azimuths permit launches to inclinations of 51 deg from Kapustin Yar and 66, 74, and 83 deg from Plesetsk. In 2000 permission was granted to launch into sun-synchronous orbits for the first time. The Kosmos flight control system does not have the capability for yaw steering, which prevents missions to inclinations between those listed above. Therefore, it is not valid to interpolate between the inclinations shown in the performance graph. The second stage is capable of a short second burn to insert into circular orbits as high as a few thousand kilometers. The propellant reserve provides at least 99.3% probability of reaching the target orbit. The mass of the standard adapter is accounted for in the performance values below.

Kosmos is also used for ballistic flights of hypersonic test vehicles (up to Mach 28) or microgravity systems. For information on capabilities for these missions, please contact United Start Corporation.



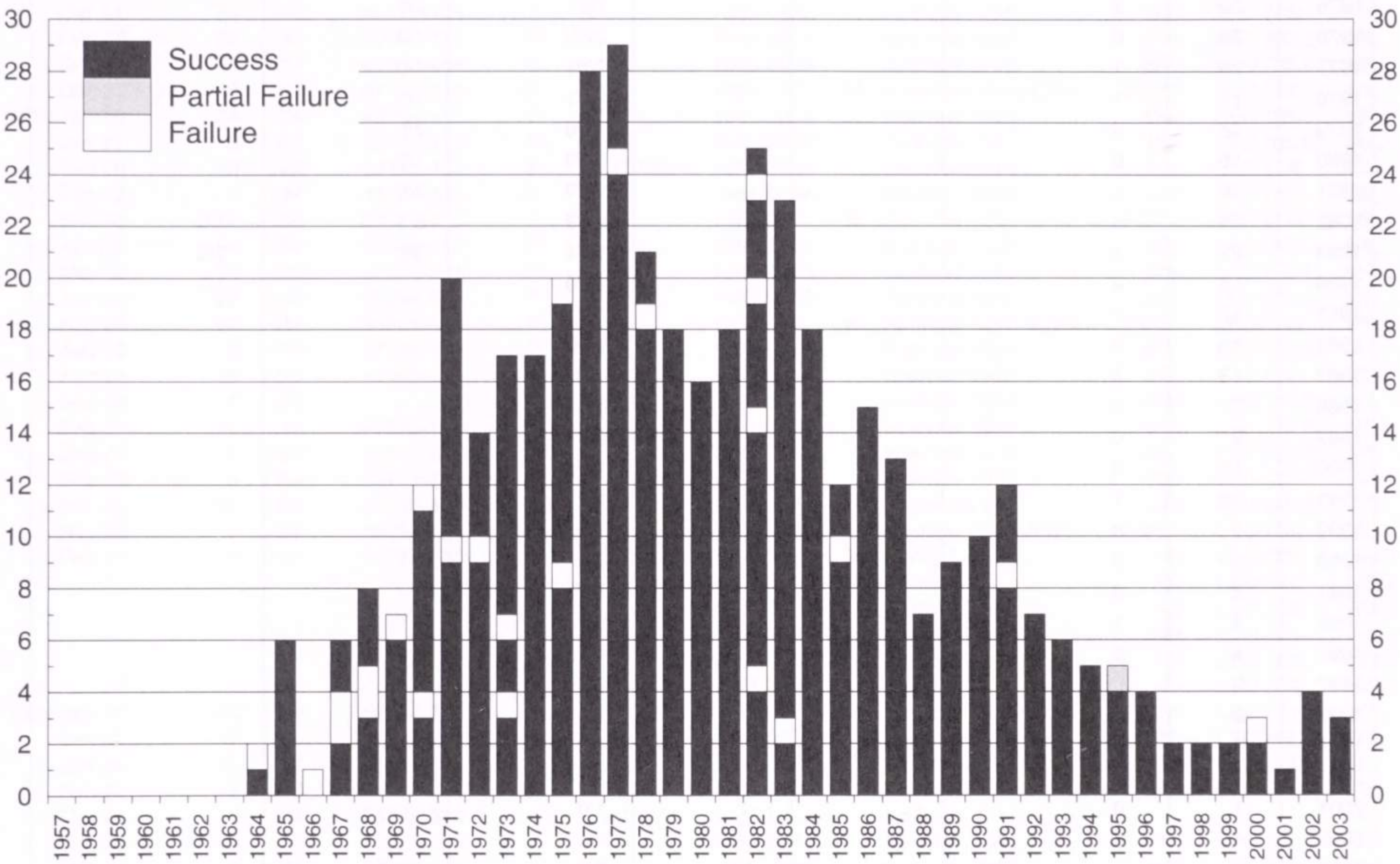
Kosmos: Performance for Circular Orbits

Flight History

In addition to orbital flights, Kosmos has flown over 330 suborbital missions. With the exception of the suborbital BOR-5 spaceplane tests, the suborbital missions are not included in the data below. The suborbital BOR-5 tests are included in the table for reference, but are not counted in the tally of orbital launches.

According to Russian definitions, certain payloads are considered suborbital if their perigee is inside the upper atmosphere, even if they complete a few revolutions of the Earth before reentering. By U.S. conventions, these trajectories are considered to be orbital, though unstable. As a result, some missions such as the BOR-4 spaceplanes are counted as orbital flights in this log, but are listed as suborbital missions by PO Polyot. (For more information on the BOR program, see the History section). In addition, this list includes a few early vehicle failures that occured on the launch pad, which are not counted as "launches" in some other references. These variations in definitions cause annual and cumulative flight totals to differ among sources.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Kosmos 3M	Combined Kosmos Family
Total Orbital Launches	432	448
Launch Vehicle Successes	409	422
Launch Vehicle Partial Failures	1	1
Launch Vehicle Failures	22	25

FLIGHT HISTORY

Year	Total	Failures or Partial Failures	Kosmos 2 *Baikonur* T/F	Kosmos 3 *Baikonur* T/F	Kosmos 3 M *Plesetsk* T/F	Kosmos 3 M *Kapustin* Yar T/F	Kosmos 3MP *Kasputin* Yar T/F
Total	448	26	8/1	4/2	414/21	18/2	4/0
1964	2	1	2/1				
1965	6	0	6/0				
1966	1	1		1/1			
1967	6	2		1/0	5/2		
1968	8	2		2/1	6/1		
1969	7	1			7/1		
1970	12	2			12/2		
1971	20	1			20/1		
1972	14	1			14/1		
1973	17	2			16/2	1/0	
1974	17	0			16/0	1/0	
1975	20	2			18/1	2/1	
1976	28	0			27/0	1/0	
1977	29	1			28/1	1/0	
1978	21	1			20/1	1/0	
1979	18	0			16/0	2/0	
1980	16	0			15/0	1/0	
1981	18	0			17/0	1/0	
1982	25	4			21/3	3/1	1/0
1983	23	1			19/1	2/0	2/0
1984	18	0			17/0		1/0
1985	12	1			12/1		
1986	15	0			15/0		
1987	13	0			12/0	1/0	
1988	7	0			7/0		
1989	9	0			9/0		
1990	10	0			10/0		
1991	12	1			12/1		
1992	7	0			7/0		
1993	6	0			6/0		
1994	5	0			5/0		
1995	5	1			5/1		
1996	4	0			4/0		
1997	2	0			2/0		
1998	2	0			2/0		
1999	2	0			1/0	1/0	
2000	3	1			3/1		
2001	1	0			1/0		
2002	4	0			4/0		
2003	3				3/0		

T = Total Launch Attempts; F = Failures and Partial Failures

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
209	1979 Jan 16	21	3M	47182-437	P		1979 003A		Kosmos 1072	810	LEO (82.9)	MIL	USSR
210	Feb 08	23	3M	53782-417	P		1979 010A		Kosmos 1075	1080	LEO (65.8)	MIL	USSR
211	Feb 27	19	3M	47155-107	P	LC 132/2	1979 020A		Interkosmos 19	1015	LEO (74)	CIV	USSR
212	Mar 15	16	3M	47168-312	P		1979 024	M	Kosmos 1081–1088	8@61	LEO (74)	MIL	USSR
213	Mar 21	6	3M	47172-413	P		1979 026A		Kosmos 1089	810	LEO (83)	MIL	USSR
214	Apr 07	17	3M	65082-424	P		1979 028A		Kosmos 1091	810	LEO (82.9)	MIL	USSR
215	Apr 11	4	3M	53782-423	P		1979 030A		Kosmos 1092	810	LEO (82.9)	MIL	USSR
216	May 31	50	3M	65082-418	P		1979 046A		Kosmos 1104	810	LEO (82.9)	MIL	USSR
217	Jun 07	7	3M	65055-109	KY	LC 107	1979 051A		Bhaskara 1	441	LEO (50.7)	CIV	India
218	Jun 28	21	3M	53768-304	P		1979 060A		Kosmos 1110	875	LEO (74)	MIL	USSR
219	Jul 06	8	3M	65075-124	KY	LC 107	1979 063A		Kosmos 1112	1190	LEO (50.7)	MIL	USSR
220	Jul 11	5	3M	47155-104	P		1979 065A		Kosmos 1114	900	LEO (74)	MIL	USSR
221	Aug 28	48	3M	47182-419	P		1979 078A		Kosmos 1125	875	LEO (74)	MIL	USSR
222	Sep 25	28	3M	47147-243	P		1979 084	M	Kosmos 1129–1137	8@61	LEO (74)	MIL	USSR
223	Oct 11	16	3M	47172-407	P		1979 089A		Kosmos 1140	875	LEO (74.1)	MIL	USSR
224	Oct 16	5	3M	47172-410	P		1979 090A		Kosmos 1141	810	LEO (83)	MIL	USSR
225	Nov 01	16	3M	53755-105	P	LC 132/2	1979 096A		Interkosmos 20	1100	LEO (74)	CIV	USSR

132/1 and 132/2, and LC 133 are at the Plesetsk Cosmodrome, LC 107 is at Kapustin Yar.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	226	Dec 05	34	3M	47168-309	P		1979 100A		Kosmos 1146	1000	LEO (65.9)	MIL	USSR
	227	1980 Jan 14	40	3M	65067-248	P		1980 003A		Kosmos 1150	810	LEO (83)	MIL	USSR
	228	Jan 25	11	3M	47164-249	P		1980 007A		Kosmos 1153	810	LEO (82.9)	MIL	USSR
	229	Feb 11	17	3M	53772-402	P		1980 012	M	Kosmos 1156–1163	8@61	LEO (74)	MIL	USSR
	230	Mar 17	35	3M	65098-314	P		1980 022A		Kosmos 1168	810	LEO (82.9)	MIL	USSR
	231	Mar 27	10	3M	65098-311	P		1980 023A		Kosmos 1169	900	LEO (65.8)	MIL	USSR
	232	Apr 03	7	3M	47198-315	P		1980 026A		Kosmos 1171	750	LEO (65.8)	MIL	USSR
	233	May 14	41	3M	53783-456	P		1980 037A		Kosmos 1179	810	LEO (82.9)	MIL	USSR
	234	May 20	6	3M	47175-116	P		1980 039A		Kosmos 1181	810	LEO (82.9)	MIL	USSR
	235	Jun 06	17	3M	47168-306	P		1980 047A		Kosmos 1186	900	LEO (74)	MIL	USSR
	236	Jul 01	25	3M	47182-416	P		1980 056A		Kosmos 1190	875	LEO (74)	MIL	USSR
	237	Jul 09	8	3M	65072-409	P		1980 058	M	Kosmos 1192–1199	8@61	LEO (74)	MIL	USSR
	238	Jul 31	22	3M	47175-122	KY	LC 107	1980 067A		Kosmos 1204	900	LEO (50.7)	MIL	USSR
	239	Oct 14	75	3M	47172-401	P		1980 083A		Kosmos 1215	900	LEO (74)	MIL	USSR
	240	Dec 05	52	3M	65098-317	P		1980 097A		Kosmos 1225	810	LEO (82.9)	MIL	USSR
	241	Dec 10	5	3M	65083-464	P		1980 099A		Kosmos 1226	680	LEO (82.9)	MIL	USSR
	242	Dec 23	13	3M	47198-318	P		1980 102	M	Kosmos 1228–1235	8@61	LEO (74)	MIL	USSR
	243	1981 Jan 16	24	3M	53798-319	P		1981 003A		Kosmos 1238	550	LEO (83)	MIL	USSR
	244	Jan 21	5	3M	65082-477	P		1981 006A		Kosmos 1241	750	LEO (65.8)	MIL	USSR
	245	Feb 06	16	3M	53793-478	P	LC 132/1	1981 011A		Interkosmos 21	550	LEO (74)	CIV	USSR
	246	Feb 12	6	3M	65098-320	P		1981 013A		Kosmos 1244	810	LEO (83)	MIL	USSR
	247	Mar 06	22	3M	53767-247	P		1981 022	M	Kosmos 1250–1257	8@61	LEO (74)	MIL	USSR
	248	Apr 09	34	3M	53798-322	P		1981 033A		Kosmos 1263	550	LEO (83)	MIL	USSR
	249	May 07	28	3M	47198-321	P		1981 041A		Kosmos 1269	875	LEO (74.1)	MIL	USSR
	250	Jun 04	28	3M	65098-323	P		1981 053A		Kosmos 1275	810	LEO (83)	MIL	USSR
	251	Aug 06	63	3M	65055-112	P		1981 074	M	Kosmos 1287–1294	8@61	LEO (74)	MIL	USSR
	252	Aug 12	6	3M	47198-324	P		1981 077A		Kosmos 1295	810	LEO (82.9)	MIL	USSR
	253	Aug 28	16	3M	47155-101	P		1981 084A		Kosmos 1302	875	LEO (74)	MIL	USSR
	254	Sep 04	7	3M		P	LC 132	1981 087A		Kosmos 1304	810	LEO (82.9)	MIL	USSR
	255	Sep 18	14	3M	47195-113	P		1981 091A		Kosmos 1308	810	LEO (82.9)	MIL	USSR
	256	Sep 23	5	3M	53719-527	P		1981 095A		Kosmos 1310	900	LEO (65.8)	MIL	USSR
	257	Sep 28	5	3M	53775-114	P		1981 097A		Kosmos 1311	550	LEO (83)	MIL	USSR
	258	Nov 20	53	3M	47193-468	KY	LC 107	1981 115A		Bhaskara 2	444	LEO (50.6)	CIV	India
	259	Nov 28	8	3M	65093-121	P		1981 116	M	Kosmos 1320–1327	8@61	LEO (74)	MIL	USSR
	260	Dec 17	19	3M	53775-120	P		1981 120A	M	Radio 3	40	LEO (83)	NGO	USSR
								1981 120B	M	Radio 8	40	LEO (83)	NGO	USSR
								1981 120C	M	Radio 5	40	LEO (83)	NGO	USSR
								1981 120D	M	Radio 4	40	LEO (83)	NGO	USSR
								1981 120E	M	Radio 7	40	LEO (83)	NGO	USSR
								1981 120F	M	Radio 6	40	LEO (83)	NGO	USSR
	261	1982 Jan 07	21	3M	53755-102	P		1982 001A		Kosmos 1331	875	LEO (74)	MIL	USSR
	262	Jan 14	7	3M	53767-250	P		1982 003A		Kosmos 1333	810	LEO (82.9)	MIL	USSR
	263	Jan 29	15	3M	65075-115	P		1982 007A		Kosmos 1335	550	LEO (74)	MIL	USSR
	264	Feb 17	19	3M	65067-261	P		1982 012A		Kosmos 1339	810	LEO (82.9)	MIL	USSR
F	**265**	**Mar 04**	**15**	**3M**	**53739-530**	**KY**	**LC 107**	1982 F01A		Kosmos			MIL	USSR
	266	Mar 24	20	3M	53767-253	P		1982 024A		Kosmos 1344	810	LEO (82.9)	MIL	USSR
	267	Mar 31	7	3M	53747-241	P		1982 026A		Kosmos 1345	900	LEO (74)	MIL	USSR
	268	Apr 08	8	3M	65047-242	P		1982 030A		Kosmos 1349	810	LEO (82.9)	MIL	USSR
	269	Apr 21	13	3M	53767-256	KY	LC 107	1982 034A		Kosmos 1351	550	LEO (50.6)	MIL	USSR
	270	Apr 28	7	3M	47175-125	P		1982 037A		Kosmos 1354	875	LEO (74)	MIL	USSR
	271	May 06	8	3M	53775-123	P		1982 040	M	Kosmos 1357–1364	8@61	LEO (74)	MIL	USSR
	272	Jun 01	26	3M	47167-258	P		1982 051A		Kosmos 1371	875	LEO (74)	MIL	USSR
	273	Jun 03	2	3MP		KY	LC 107	1982 054A		BOR 4-1 (Kosmos 1374)	1074	LEO (50.6)	MIL	USSR
	274	Jun 06	3	3M	65067-257	P		1982 055A		Kosmos 1375	750	LEO (65.8)	MIL	USSR
F	**275**	**Jun 18**	**12**	**3M**	**53783-460**	**P**	**LC 132**	1982 061A		Kosmos 1380	810	LEO (82.9)	MIL	USSR
	276	Jun 29	11	3M	47183-457	P	LC 132	1982 066A		Kosmos 1383	810	LEO (82.9)	MIL	USSR
	277	Jul 07	8	3M	65093-473	P		1982 069A		Kosmos 1386	810	LEO (83)	MIL	USSR
	278	Jul 21	14	3M	65055-103	P		1982 073	M	Kosmos 1388–1395	8@61	LEO (74)	MIL	USSR
	279	Jul 29	8	3M	53793-472	P		1982 076A		Kosmos 1397	550	LEO (50.6)	MIL	USSR
F	**280**	**Aug 30**	**32**	**3M**	**65047-236**	**P**	**LC 132**	1982 F05A		Kosmos			MIL	USSR
	281	Oct 19	50	3M	53734-137	P		1982 102A		Kosmos 1417	810	LEO (83)	MIL	USSR
	282	Oct 21	2	3M	47193-471	KY	LC 107	1982 104A		Kosmos 1418	550	LEO (50.6)	MIL	USSR
	283	Nov 11	21	3M	47167-252	P		1982 109A		Kosmos 1420	875	LEO (74)	MIL	USSR
F	**284**	**Nov 24**	**13**	**3M**	**47183-465**	**P**	**LC 132**	1982 F07A		Kosmos			MIL	USSR
	285	Dec 29	35	3M	57783-454	P		1982 121A		Kosmos 1427	750	LEO (65.8)	MIL	USSR
	286	1983 Jan 12	14	3M	65093-479	P		1983 001A		Kosmos 1428	810	LEO (82.9)	MIL	USSR
	287	Jan 19	7	3M	53793-481	P		1983 002	M	Kosmos 1429 - 1436	8@61	LEO (74)	MIL	USSR
F	**288**	**Jan 25**	**6**	**3M**	**47193-480**	**P**	**LC 132**	1983 F01A		Kosmos			MIL	USSR
	289	Mar 15	49	3MP		KY	LC 107	1983 017A		BOR 4-2 (Kosmos 1445)	1000	LEO (50.6)	MIL	USSR
	290	Mar 24	9	3M	53739-548	P		1983 021A		Kosmos 1447/KOSPAS	810	LEO (82.9)	MIL	USSR
	291	Mar 30	6	3M	65019-522	P		1983 023A		Kosmos 1448	810	LEO (83)	MIL	USSR

132/1 and 132/2, and LC 133 are at the Plesetsk Cosmodrome, LC 107 is at Kapustin Yar.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Flight History

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
292	Apr 06	7	3M	53719-512	P		1983 027A		Kosmos 1450	750	LEO (65.9)	MIL	USSR
293	Apr 12	6	3M	65019-519	P		1983 031A		Kosmos 1452	875	LEO (74)	MIL	USSR
294	Apr 19	7	3M	47119-520	P		1983 034A		Kosmos 1453	550	LEO (74)	MIL	USSR
295	May 06	17	3M	65019-513	P		1983 042A		Kosmos 1459	810	LEO (82.9)	MIL	USSR
296	May 19	13	3M	47119-511	P		1983 046A		Kosmos 1463	550	LEO (82.9)	MIL	USSR
297	May 24	5	3M	47139-541	P		1983 048A		Kosmos 1464	810	LEO (82.9)	MIL	USSR
298	May 26	2	3M	65083-458	KY	LC 107	1983 049A		Kosmos 1465	550	LEO (50.6)	MIL	USSR
	Jul 04	39	3MP		KY	LC 107			BOR 5-1	1400	suborbital	MIL	USSR
299	Jul 06	2	3M	47183-459	P		1983 069	M	Kosmos 1473–1480	8@61	LEO (74)	MIL	USSR
300	Aug 03	28	3M	53783-463	P		1983 079A		Kosmos 1486	875	LEO (74)	MIL	USSR
301	Aug 31	28	3M	47139-547	KY	LC 107	1983 091A		Kosmos 1494	550	LEO (50.6)	MIL	USSR
302	Sep 30	30	3M	65093-482	P		1983 101A		Kosmos 1501	550	LEO (82.9)	MIL	USSR
303	Oct 05	5	3M	65039-540	P		1983 102A		Kosmos 1502	750	LEO (65.8)	MIL	USSR
304	Oct 12	7	3M	65039-543	P		1983 103A		Kosmos 1503	875	LEO (74)	MIL	USSR
305	Oct 26	14	3M	47122-331	P		1983 108A		Kosmos 1506	810	LEO (82.9)	MIL	USSR
306	Nov 11	16	3M	65093-470	P		1983 111A		Kosmos 1508	550	LEO (82.9)	MIL	USSR
307	Dec 08	27	3M	65019-528	P		1983 120A		Kosmos 1513	810	LEO (82.9)	MIL	USSR
308	Dec 27	19	3MP		KY	LC 107	1983 125A		BOR 4-3 (Kosmos 1517)	1000	LEO (50.6)	MIL	USSR
309	1984 Jan 05	9	3M	47119-529	P		1984 001	M	Kosmos 1522–1529	8@61	LEO (74)	MIL	USSR
310	Jan 11	6	3M	65039-546	P		1984 003A		Kosmos 1531	810	LEO (82.9)	MIL	USSR
311	Jan 26	15	3M	53793-469	P		1984 007A		Kosmos 1534	2000	LEO (65.8)	MIL	USSR
312	Feb 02	7	3M	47139-532	P		1984 010A		Kosmos 1535	810	LEO (83)	MIL	USSR
313	Feb 21	19	3M	65039-537	P		1984 019A		Kosmos 1538	875	LEO (74)	MIL	USSR
314	May 11	80	3M	53639-533	P		1984 043A		Kosmos 1550	810	LEO (83)	MIL	USSR
315	May 17	6	3M	47139-538	P		1984 046A		Kosmos 1553	810	LEO (82.9)	MIL	USSR
316	May 28	11	3M	65039-531	P		1984 052	M	Kosmos 1559–1566	8@61	LEO (74)	MIL	USSR
	Jun 06	9	3MP		KY	LC 107			BOR 5-2	1400	suborbital	MIL	USSR
317	Jun 08	2	3M	53739-542	P		1984 056A		Kosmos 1570	875	LEO (74.1)	MIL	USSR
318	Jun 21	13	3M	53719-521	P		1984 062A		Kosmos 1574	810	LEO (83)	MIL	USSR
319	Jun 27	6	3M	65039-549	P		1984 067A		Kosmos 1577	810	LEO (83)	MIL	USSR
320	Jun 28	1	3M	47134-133	P		1984 068A		Kosmos 1578	550	LEO (50.7)	MIL	USSR
321	Sep 13	77	3M	53739-539	P		1984 100A		Kosmos 1598	810	LEO (82.9)	MIL	USSR
322	Sep 27	14	3M	53796-180	P		1984 104A		Kosmos 1601	550	LEO (65.8)	MIL	USSR
323	Oct 11	14	3M	53734-125	P		1984 109A		Kosmos 1605	810	LEO (82.9)	MIL	USSR
324	Nov 15	35	3M	47139-124	P		1984 118A		Kosmos 1610	810	LEO (82.9)	MIL	USSR
325	Dec 19	34	3MP		KY	LC 107	1984 126A		BOR 4-4 (Kosmos 1614)	1000	LEO (50.7)	MIL	USSR
326	Dec 20	1	3M	47134-127	P		1984 127A		Kosmos 1615	550	LEO (65.8)	MIL	USSR
327	1985 Jan 17	28	3M	65034-138	P		1985 006A		Kosmos 1624	875	LEO (74)	MIL	USSR
328	Feb 01	15	3M	47144-445	P		1985 011A		Kosmos 1627	810	LEO (82.9)	MIL	USSR
329	Feb 27	26	3M	47134-139	P		1985 018A		Kosmos 1631	550	LEO (65.9)	MIL	USSR
330	Mar 14	15	3M	53744-146	P		1985 022A		Kosmos 1634	810	LEO (82.9)	MIL	USSR
331	Mar 21	7	3M	53734-134	P		1985 023	M	Kosmos 1635–1642	8@61	LEO (74.1)	MIL	USSR
	Apr 17	27	3MP		KY	LC 107			BOR 5-3	1400	suborbital	MIL	USSR
332	May 30	43	3M	47144-148	P		1985 041A		Kosmos 1655	810	LEO (83)	MIL	USSR
333	Jun 19	20	3M	65044-147	P		1985 050A		Kosmos 1662	550	LEO (65.8)	MIL	USSR
334	Sep 04	77	3M	65011-241	P		1985 079A		Kosmos 1680	875	LEO (74.1)	MIL	USSR
335	Oct 02	28	3M	65044-153	P		1985 089A		Kosmos 1688	550	LEO (50.7)	MIL	USSR
F 336	**Oct 23**	**21**	**3M**	**65034-135**	**P**	**LC 133**	1985 F04A		Kosmos			MIL	USSR
337	Nov 28	36	3M	47122-346	P		1985 110A		Kosmos 1704	810	LEO (82.9)	MIL	USSR
338	Dec 19	21	3M	65022-348	P		1985 116A		Kosmos 1709	810	LEO (82.9)	MIL	USSR
339	1986 Jan 09	21	3M	47122-337	P		1986 002	M	Kosmos 1716–1723	8@61	LEO (74)	MIL	USSR
340	Jan 16	7	3M	53744-152	P		1986 005A		Kosmos 1725	810	LEO (82.9)	MIL	USSR
341	Jan 23	7	3M	47122-334	P		1986 008A		Kosmos 1727	810	LEO (82.9)	MIL	USSR
342	Apr 17	84	3M	53726-198	P		1986 030A		Kosmos 1741	875	LEO (74)	MIL	USSR
343	May 23	36	3M	53772-362	P		1986 037A		Kosmos 1745	810	LEO (82.9)	MIL	USSR
344	Jun 06	14	3M	47172-364	P		1986 042	M	Kosmos 1748–1755	8@61	LEO (74)	MIL	USSR
345	Jun 18	12	3M	53722-341	P		1986 047A		Kosmos 1759	810	LEO (82.9)	MIL	USSR
346	Jul 16	28	3M	54722-365	P		1986 052A		Kosmos 1763	875	LEO (74)	MIL	USSR
347	Sep 03	49	3M	65022-342	P		1986 067A		Kosmos 1776	550	LEO (74)	MIL	USSR
348	Sep 10	7	3M	47196-164	P		1986 070A		Kosmos 1777	875	LEO (74)	MIL	USSR
349	Oct 27	47	3M	53748-411	P		1986 083A		Kosmos 1788	550	LEO (65.8)	MIL	USSR
350	Nov 13	17	3M	54748-414	P		1986 086A		Kosmos 1791	810	LEO (82.9)	MIL	USSR
351	Nov 21	8	3M	53796-168	P		1986 092	M	Kosmos 1794–1801	8@61	LEO (74)	MIL	USSR
352	Nov 24	3	3M	47122-349	P		1986 093A		Kosmos 1802	810	LEO (82.9)	MIL	USSR
353	Dec 17	23	3M	53772-353	P		1986 100A		Kosmos 1808	810	LEO (82.9)	MIL	USSR
	Dec 25	8	3MP		KY	LC 107			BOR 5-4	1400	sub orbital	MIL	USSR
354	1987 Jan 21	27	3M	47172-361	P		1987 006A		Kosmos 1814	875	LEO (74.1)	MIL	USSR

LC 132L and R, and LC 133 are at the Plesetsk Cosmodrome, LC 107 is at Kapustin Yar.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Flight History

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
355	Jan 22	1	3M	65048-415	KY	LC 107	1987 007A		Kosmos 1815	550	LEO (50.7)	MIL	USSR
356	Jan 29	7	3M	65096-172	P		1987 009A		Kosmos 1816	810	LEO (82.9)	MIL	USSR
357	Feb 18	20	3M	65072-354	P		1987 017A		Kosmos 1821	810	LEO (82.9)	MIL	USSR
358	Jun 09	111	3M	47196-176	P		1987 049A		Kosmos 1850	875	LEO (74)	MIL	USSR
359	Jun 16	7	3M	65026-184	P		1987 051	M	Kosmos 1852–1859	8@61	LEO (74)	MIL	USSR
360	Jun 23	7	3M	65026-202	P		1987 054A		Kosmos 1861	810	LEO (82.9)	MIL	USSR
361	Jul 06	13	3M	53796-177	P		1987 057A		Kosmos 1864	810	LEO (82.9)	MIL	USSR
362	Jul 14	8	3M	47196-161	P		1987 061A		Kosmos 1868	550	LEO (74)	MIL	USSR
	Aug 27	44	3MP		KY	LC 107			BOR 5-5	1400	suborbital	MIL	USSR
363	Oct 14	48	3M	53726-186	P		1987 087A		Kosmos 1891	810	LEO (82.9)	MIL	USSR
364	Dec 01	48	3M	65035-621	P		1987 098A		Kosmos 1898	875	LEO (74)	MIL	USSR
365	Dec 15	14	3M	53726-183	P		1987 103A		Kosmos 1902	550	LEO (65.8)	MIL	USSR
366	Dec 23	8	3M	53711-281	P		1987 106A		Kosmos 1904	810	LEO (82.9)	MIL	USSR
367	1988 Mar 11	79	3M	47161-263	P		1988 016	M	Kosmos 1924–1931	8@61	LEO (74)	MIL	USSR
368	Mar 22	11	3M	47111-221	P		1988 023A		Kosmos 1934	810	LEO (83)	MIL	USSR
369	Apr 05	14	3M	47126-188	P		1988 029A		Kosmos 1937	875	LEO (74)	MIL	USSR
370	Jun 21	77	3M	47126-191	P		1988 053A		Kosmos 1954	875	LEO (74)	MIL	USSR
	Jun 22	1	3MP		KY	LC 107			BOR 5-6	1400	suborbital	MIL	USSR
371	Jul 14	22	3M	47111-224	P		1988 060A		Kosmos 1958	550	LEO (65.8)	MIL	USSR
372	Jul 18	4	3M	65096-175	P		1988 062A		Kosmos 1959	810	LEO (83)	MIL	USSR
373	Jul 28	10	3M	65026-187	P		1988 065A		Kosmos 1960	550	LEO (65.8)	MIL	USSR
374	1989 Jan 26	182	3M	65079-812	P		1989 005A		Kosmos 1992	875	LEO (74.1)	MIL	USSR
375	Feb 14	19	3M	53726-192	P		1989 012A		Kosmos 2002	550	LEO (65.8)	MIL	USSR
376	Feb 22	8	3M	47111-236	P		1989 017A		Kosmos 2004	810	LEO (82.9)	MIL	USSR
377	Mar 24	30	3M	65011-232	P		1989 025	M	Kosmos 2008–2015	8@61	LEO (74)	MIL	USSR
378	Apr 04	11	3M	47126-194	P		1989 028A		Kosmos 2016	810	LEO (82.9)	MIL	USSR
379	Jun 07	64	3M	53711-225	P		1989 042A		Kosmos 2026	810	LEO (82.9)	MIL	USSR
380	Jun 14	7	3M	53761-249	P		1989 045A		Kosmos 2027	900	LEO (65.8)	MIL	USSR
381	Jul 04	20	3M	65061-250	P		1989 050A		Nadezhda 1/KOSPAS	825	LEO (83)	CIV	USSR
382	Jul 25	21	3M	65061-256	P		1989 059A		Kosmos 2034	810	LEO (82.9)	MIL	USSR
383	1990 Jan 18	177	3M	65061-247	P		1990 004A		Kosmos 2056	875	LEO (74)	MIL	USSR
384	Feb 06	19	3M	53711-237	P		1990 012A		Kosmos 2059	550	LEO (65.8)	MIL	USSR
385	Feb 27	21	3M	53711-228	P		1990 017A		Nadezdha 2/KOSPAS	825	LEO (83)	CIV	USSR
386	Mar 20	21	3M	47161-245	P		1990 023A		Kosmos 2061	810	LEO (82.9)	MIL	USSR
387	Apr 06	17	3M	65061-259	P		1990 029	M	Kosmos 2064–2071	8@61	LEO (74)	MIL	USSR
388	Apr 20	14	3M	53726-195	P		1990 036A		Kosmos 2074	810	LEO (82.9)	MIL	USSR
389	Apr 25	5	3M	65061-253	P		1990 038A		Kosmos 2075	550	LEO (63.2)	MIL	USSR
390	Aug 28	125	3M	53711-240	P		1990 078A		Kosmos 2098	550	LEO (83)	MIL	USSR
391	Sep 14	17	3M	65048-403	P		1990 083A		Kosmos 2100	810	LEO (82.9)	MIL	USSR
392	Dec 10	87	3M	47126-203	P		1990 111A		Kosmos 2112	875	LEO (74)	MIL	USSR
393	1991 Jan 29	50	3M	53744-158	P		1991 006A		Informatr 1	800	LEO (82.9)	NGO	USSR
394	Feb 05	7	3M	47148-407	P		1991 007A		Kosmos 2123	810	LEO (82.9)	MIL	USSR
395	Feb 12	7	3M	47187-204	P		1991 009	M	Kosmos 2125–2132	8@61	LEO (74)	MIL	USSR
396	Feb 26	14	3M	47168-254	P		1991 013A		Kosmos 2135	810	LEO (82.8)	MIL	USSR
397	Mar 12	14	3M	65048-406	P		1991 019A		Nadezhda 3/KOSPAS	825	LEO (82.9)	CIV	USSR
398	Mar 19	7	3M	53448-417	P		1991 021A		Kosmos 2137	900	LEO (65.8)	MIL	USSR
399	Apr 16	28	3M	47148-401	P		1991 029A		Kosmos 2142	810	LEO (83)	MIL	USSR
400	Jun 11	56	3M	47159-819	P		1991 041A		Kosmos 2150	875	LEO (74)	MIL	USSR
F 401	**Jun 25**	**14**	**3M**	**65061-262**	**P**	**LC 132**	1991 F02A		Kosmos			MIL	USSR
402	Aug 22	58	3M	53778-429	P		1991 059A		Kosmos 2154	810	LEO (82.9)	MIL	USSR
403	Oct 10	49	3M	53778-420	P		1991 072A		Kosmos 2164	900	LEO (73.9)	MIL	USSR
404	Nov 26	47	3M	65048-409	P		1991 081A		Kosmos 2173	810	LEO (83)	MIL	USSR
405	1992 Feb 17	83	3M	65078-418	P		1992 008A		Kosmos 2180	810	LEO (82.9)	MIL	Russia
406	Mar 09	21	3M	53778-423	P		1992 012A		Kosmos 2181	810	LEO (82.9)	MIL	Russia
407	Apr 15	37	3M	47178-419	P		1992 020A		Kosmos 2184	810	LEO (82.9)	MIL	Russia
408	Jun 03	49	3M	53742-114	P		1992 030	M	Kosmos 2187–2194	8@61	LEO (74)	MIL	Russia
409	Jul 01	28	3M	65072-427	P		1992 036A		Kosmos 2195	810	LEO (82.9)	MIL	Russia
410	Aug 12	42	3M	65078-421	P		1992 053A		Kosmos 2208	875	LEO (74)	MIL	Russia
411	Oct 29	78	3M	65078-424	P		1992 073A		Kosmos 2218	810	LEO (82.9)	MIL	Russia
412	1993 Jan 12	75	3M	53778-426	P		1993 001A		Kosmos 2230	810	LEO (82.9)	MIL	Russia
413	Feb 09	28	3M	47178-428	P		1993 008A		Kosmos 2233	810	LEO (82.9)	MIL	Russia
414	Apr 01	51	3M	47178-431	P		1993 020A		Kosmos 2239	810	LEO (82.9)	MIL	Russia
415	Jun 16	76	3M	47135-601	P		1993 036A		Kosmos 2251	875	LEO (74)	MIL	Russia
416	Oct 26	132	3M	65065-624	P	LC 132	1993 067A		Kosmos 2265	500	LEO (82.9)	MIL	Russia
417	Nov 02	7	3M	53735-608	P	LC 132	1993 070A		Kosmos 2266	810	LEO (82.9)	MIL	Russia
418	1994 Apr 26	175	3M		P	LC 133/3	1994 024A		Kosmos 2279	810	LEO (82.9)	MIL	Russia
419	Jul 14	79	3M		P	LC 133/3	1994 041A		Nadezhda 4/KOSPAS	825	LEO (82.9)	CIV	Russia

LC 132L and R, and LC 133 are at the Plesetsk Cosmodrome, LC 107 is at Kapustin Yar.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	420	Aug 02	19	3M		P	LC 132/1	1994 045A		Kosmos 2285	875	LEO (74)	MIL	Russia
	421	Sep 27	56	3M		P	LC 132/1	1994 061A		Kosmos 2292	500	LEO (82.9)	MIL	Russia
	422	Dec 20	84	3M		P	LC 132/1	1994 083A		Kosmos 2298	875	LEO (74)	MIL	Russia
	423	1995 Jan 24	35	3M		P	LC 132/1	1995 002A		Tsikada	825	LEO (82.9)	CIV	Russia
								1995 002B	A	Astrid 1	28	LEO (82.9)	CIV	Sweden
								1995 002C	A	FAISat	114	LEO (82.9)	CML	USA
	424	Mar 02	37	3M		P	LC 132/1	1995 008A		Kosmos 2306	1000	LEO (65.5)	MIL	Russia
	425	Mar 22	20	3M		P	LC 132/1	1995 012A		Kosmos 2310	810	LEO (82.9)	MIL	Russia
	426	Jul 05	105	3M		P	LC 132/1	1995 032A		Kosmos 2315	810	LEO (82.9)	MIL	Russia
P	**427**	**Oct 06**	**93**	**3M**		**P**	**LC 132/1**	1995 052A		Kosmos 2321	810	LEO (82.9)	MIL	Russia
	428	1996 Jan 16	102	3M		P	LC 132/1	1996 004A		Kosmos 2327	810	LEO (83)	MIL	Russia
	429	Apr 24	99	3M		P	LC 132/1	1996 025A		Kosmos 2332	3250	LEO (71)	MIL	Russia
	430	Sep 05	134	3M		P	LC 132/1	1996 052A		Kosmos 2334	810	LEO (82.9)	MIL	Russia
								1996 052B	A	UNAMSAT B	17	LEO (82.9)	NGO	Mexico
	431	Dec 20	106	3M		P	LC 132/1	1996 071A		Kosmos 2336	810	LEO (82.9)	MIL	Russia
	432	1997 Apr 17	118	3M		P	LC 132/1	1997 017A		Kosmos 2341	825	LEO (82.9)	MIL	Russia
	433	Sep 23	159	3M		P	LC 132/1	1997 052A		Kosmos 2336	900	LEO (82.9)	MIL	Russia
								1997 052B	A	FAISat 2V	115	LEO (82.9)	CML	USA
	434	1998 Dec 10	443	3M		P		1998 072A		Nadezhda 5/KOSPAS	850	LEO (83)	CIV	Russia
								1998 072B	A	Astrid 2	35	LEO (83)	CIV	Sweden
	435	Dec 24	14	3M		P		1998 076A		Kosmos 2361	795	LEO (82.9)	MIL	Russia
S	436	1999 Apr 28	125	3M		KY	LC 107	1999 022A		ABRIXAS	550	LEO (48.5)	CIV	Germany
								1999 022B	A	Megsat 0	35	LEO (48.5)	NGO	Italy
	437	Aug 26	120	3M		P	LC 132/1	1999 045A		Kosmos 2366	795	LEO (83)	MIL	Russia
	438	2000 Jun 28	307	3M		P	LC 132	2000 033A		Nadezhda 6/KOSPAS	825	SSO	CIV	Russia
								2000 033B	A	Tsinghua 1	49	SSO	CIV	China
								2000 033C	A	SNAP 1	6	SSO	CML	UK
	439	Jul 15	17	3M	47136-414	P	LC 132	2000 039A		CHAMP	550	SSO	CIV	Germany
								2000 039B	A	Nina (MITA)	170	LEO (87.3)	CIV	Italy
								2000 039C	A	Rubin	14	LEO (87.3)	CIV	Germany
F	**440**	**Nov 20**	**128**	**3M**	**47136414**	**P**	**LC 132/1**	**2000 074A**		**QuickBird 1**	**950**	**LEO (66)**	**CML**	**USA**
	441	2001 Jun 08	200	3M		P	LC 132/1	2001 023A		Kosmos 2378	795	LEO (82.9)	MIL	Russia
	442	2002 May 28	354	3M		P	LC 132/1	2002 026A		Kosmos 2389	825	LEO (83)	MIL	Russia
	443	Jul 08	41	3M		P	LC 132/1	2002 036A	M	Kosmos 2390	225	LEO (83)	MIL	Russia
								2002 036B	M	Kosmos 2391	225	LEO (83)	MIL	Russia
	444	Sep 26	80	3M		P	LC 132	2002 046A		Nadezhda 7/KOSPAS	825	LEO (82.9)	CIV	Russia
	445	Nov 28	63	3M		P	LC 132/1	2002 054A		AlSat 1 (DMC-1)	80	SSO	CIV	Algeria
								2002 054B	A	Mozhaetz	64	SSO	MIL	Russia
								2002 054C	A	Rubin 3-DSI	45	SSO	CIV	Germany
	446	2003 Jun 04	188	3M		P	LC 132/1	2003 023A		Kosmos 2398	820	LEO (83)	MIL	Russia
	447	Aug 19	76	3M		P	LC 132/1	2003 037A	C	Kosmos 2400	230	LEO (83)	MIL	Russia
								2003 037B	C	Kosmos 2401	230	LEO (83)	MIL	Russia
	448	Sep 27	39	3M		P	LC 132/1	2003 042A	A	Mozhaets 4	64	SSO	CIV	Russia
								2003 042C	A	NigeriaSat 1	80	SSO	CIV	Nigeria
								2003 042D	A	BNSCSat 1	80	SSO	CIV	UK
								2003 042E	A	BilSat 1	130	SSO	CIV	Turkey
								2003 042F	A	Larets	21	SSO	MIL	Russia
								2003 042G		STSat 1	110	SSO	CIV	Korea, S.
								2003 042G	A	Rubin 4-DSI	45	SSO	CML	Germany

LC 132L and R, and LC 133 are at the Plesetsk Cosmodrome, LC 107 is at Kapustin Yar.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Failure Descriptions:

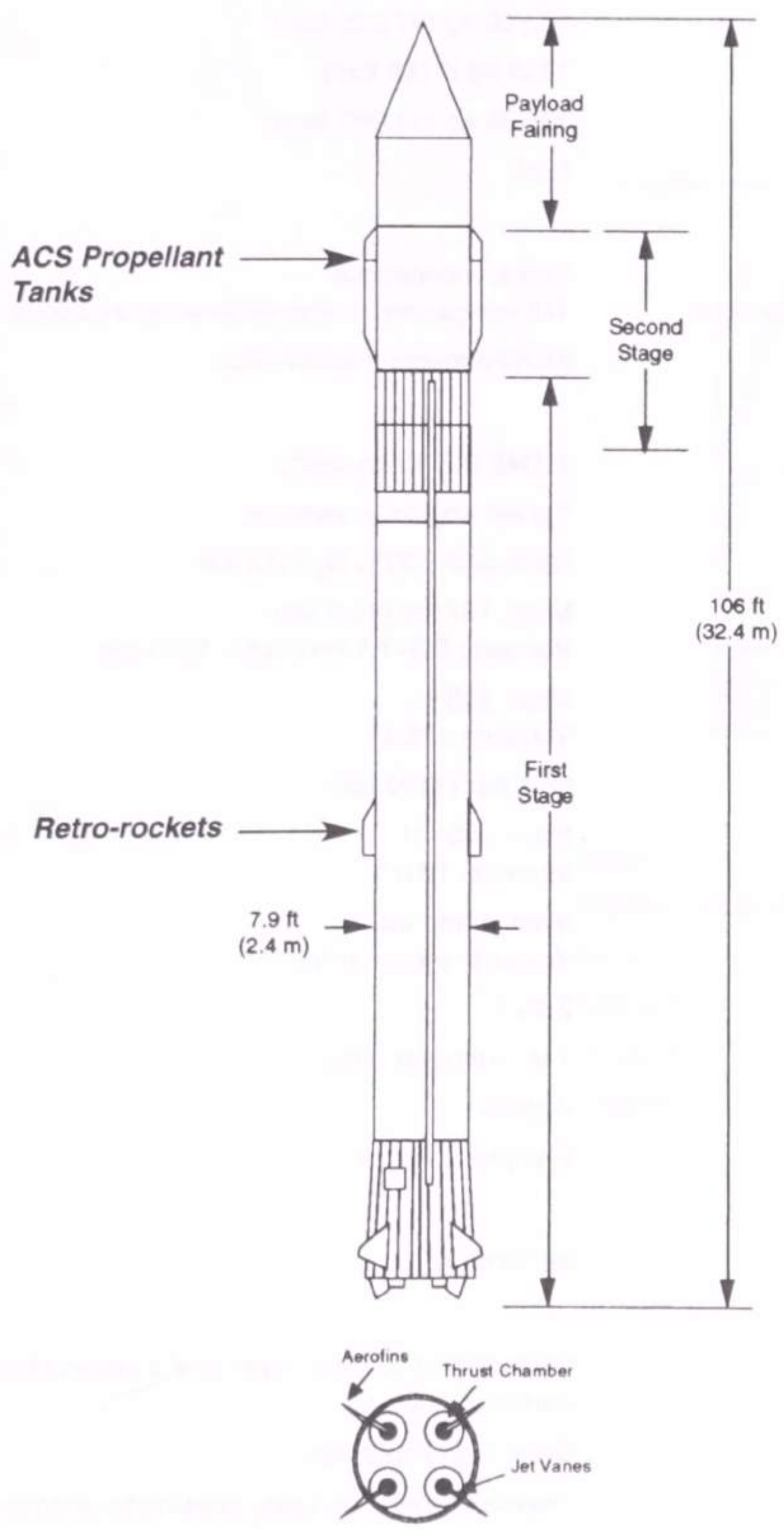
F	1964 Oct 23		1964 F12	Failure to orbit
F	1966 Nov 16		1966 F11	Failure to orbit
F	1967 Jun 26		1967 F07	Failure to orbit
F	1967 Sep 27		1967 F10	Vehicle exploded on launch pad.
F	1968 Jun 04		1968 F05	Failure to orbit
F	1968 Jun 15		1968 F06	Failure to orbit
F	1969 Dec 27		1969 F15	Failure to orbit
F	1970 Jun 27		1970 F05	Failure to orbit
F	1970 Dec 22	47117-106	1970 F10	First stage failed 1 s after launch.
F	1971 Jul 22	Yu149-32	1971 F07	Failure to orbit
F	1972 Oct 17	Yu149-40	1972 F06	Failure to orbit
F	1973 May 25	65024-109	1973 F03	Failure to orbit
F	1973 Jun 26	Yu47121-16		Exploded on pad before launch, killing nine people.
F	1975 Jun 03	53721-257	1975 F04	First stage failed 84 s after launch.
F	1975 Dec 19	53721-263	1975 F07	Failure to orbit
F	1977 Nov 29	53721-265	1977 F06	Failure to orbit
F	1978 Dec 20	47182-425	1978 119A	Second stage shut down during circularization burn, leaving satellite in lower than planned orbit.

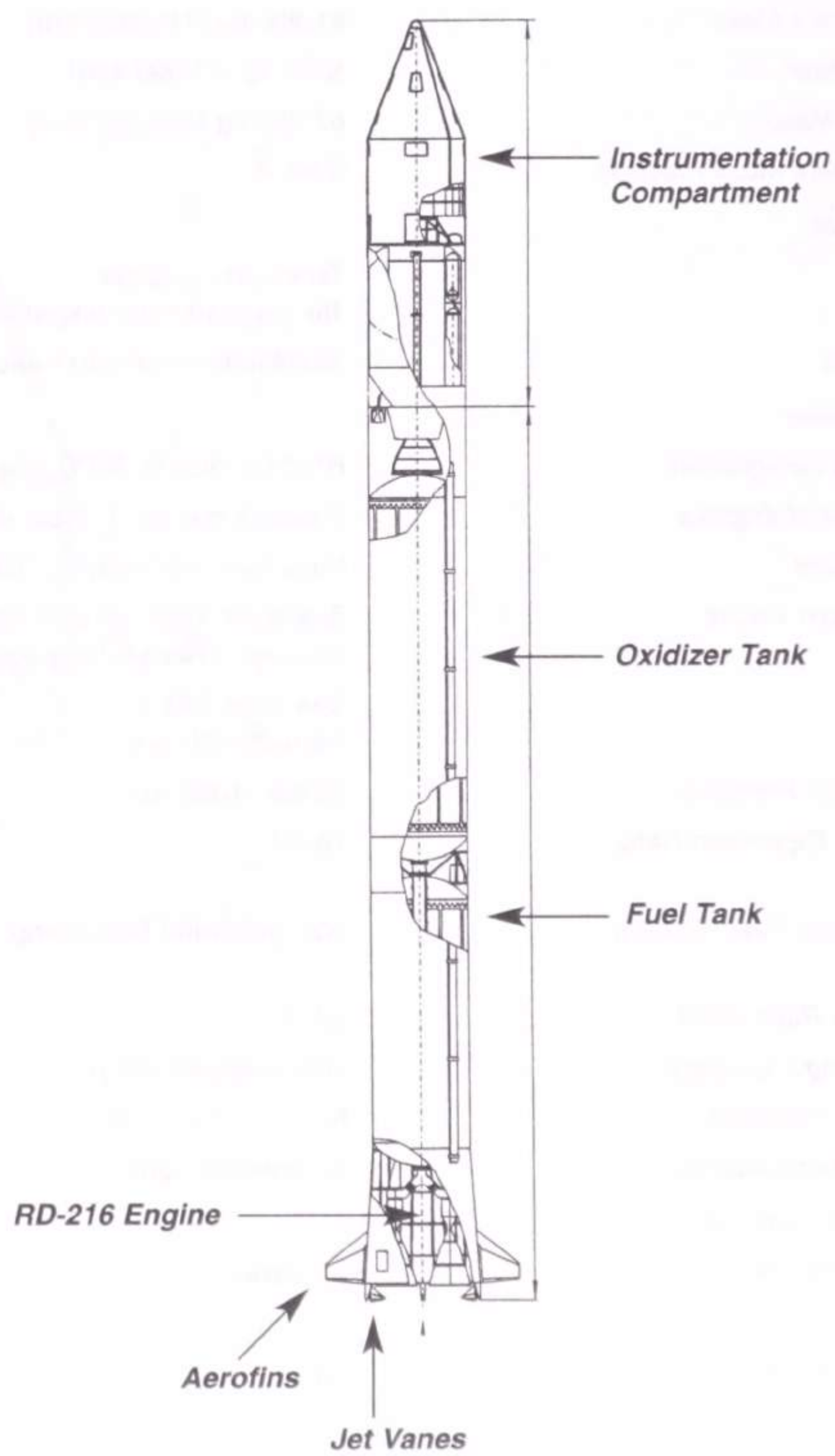
Flight History

F	1982 Mar 04	53739-530	1982 F01	Failure to orbit
F	1982 Jun 18	53783-460	1982 061A	Second stage malfunction during first burn resulted in low transfer orbit apogee.Satellite was deployed in lower than planned orbit and reentered after approximately nine days.
F	1982 Aug 30	65047-236	1982 F05	Failure to orbit
F	1982 Nov 24	47183-465	1982 F07	Failure to orbit
F	1983 Jan 25	47193-480	1983 F01	Failure to orbit
F	1985 Oct 23	65034-135	1985 F04	Failure to orbit
F	1991 Jun 25	65061-262	1991 F02	Vehicle did not reach orbit, possibly because of a second-stage failure.
P	1995 Oct 06		1995 052A	Fuel supply to second-stage engine was impeded, possibly the result of a blocked fuel valve. As a result of low performance during the circularization burn, the spacecraft was delivered to a lower than planned orbit, but reportedly became operational.
S	1999 Apr 28	ABRIXAS	1999 022A	At least 1 of 11 battery cells on the ABRIXAS satellite failed a few hours after launch causing power problems resulting in loss of communications on 1 May.
F	2000 Nov 20	47165-631	2000 074A	The spacecraft was delivered to an unstable orbit and reentered before contact could be established. The failure occurred sometime after the first burn of the second stage, but the cause could not be determined because of a lack of telemetry coverage. The launch vehicle had been in storage for 13 years.

Vehicle Design

Overall Vehicle





Kosmos 3M

Height	32.4 m (106 ft)
Gross Liftoff Mass	109 t (240 klbm)
Thrust at Liftoff	1485 kN (334 klbm)

Vehicle Design

Stages

Both stages of the Kosmos are cylindrical and carry storable liquid propellants. The first stage consists of an interstage, two propellant tanks, an intertank compartment, and a tail compartment that contains the propulsion system. The tanks are made of welded aluminum with a minimum thickness of 2.25 mm (0.09 in.). The oxidizer feed line passes through the middle of the fuel tank. The tail compartment is slightly conical, with a maximum diameter of 2.85 m (9.35 ft), at its base. Small aerodynamic fins bring the total span to 4.4 m (14.4 ft). The compartments are riveted aluminum skin-stringer construction, with skins 0.8–2.0 mm (0.03–0.08 in.) thick. The first-stage propulsion system is designated the RD-216 and consists of two sets of turbopumps and four thrust chambers. The intertank compartment contains some of the first-stage control and telemetry systems, as well as two solid retro-rockets for stage separation.

The second-stage configuration is similar to the first stage. A tail compartment contains the propulsion systems as well as a small solid motor that moves the second stage away after payload deployment. The propellant is housed in two cylindrical tanks with a common bulkhead, and the oxidizer feed line runs through the middle of the fuel tank. The cylindrical instrumentation compartment, which houses the control and telemetry systems, is located above the propellant tanks. An octagonal spacecraft mounting frame supports the payload. For missions to higher orbits a pair of long thin propellant tanks is attached to the outside of the second stage. These tanks carry the propellant for the ACS system and for the second burn of the main engine, which lasts only 2–8 s. The Kosmos second stage was the first in the Soviet Union capable of restarting for injection into high circular orbits. The spacecraft interface includes an unusual separation system. Instead of springs, separation is performed by pneumatic pistons, driven by 8–12 gas generators.

	Stage 1	Stage 2
Dimensions		
Length	22.4 m (73 ft) including interstage	6.0 m (19.7 ft)
Diameter	2.4 m (7.9 ft) primary diameter 4.4 m (14.4 ft) from fin-tip to fin-tip	2.4 m (7.9 ft)
Mass		
Propellant Mass	81,900 kg (180,560 lbm)	18,700 kg (41,225 lbm)
Inert Mass	5300 kg (11,690 lbm)	1435 kg (3165 lbm)
Gross Mass	87,200 kg (192,250 lbm)	20,135 kg (44,390 lbm)
Propellant Mass Fraction	0.94	0.93
Structure		
Type	Tanks: monocoque Tail compartment: ring-stiffened monocoque	Tanks: monocoque Tail compartment: ring-stiffened monocoque
Material	Aluminum–magnesium alloy	Aluminum–magnesium alloy
Propulsion		
Engine Designation	RD-216/11D614 (NPO Energomash)	11D49 (KB Khimmash)
Number of Engines	2 sets of pumps, 4 thrust chambers	1 main engine + verniers
Propellant	Nitric acid +27% N_2O_4 / UDMH	Nitric acid +27% N_2O_4/UDMH
Maximum Thrust	Sea level: 1485 kN (334 klbf) Vacuum: 1745 kN (392 klbf)	Main: 157 kN (35.3 klbf) Verniers: 5.6–7.1 kN (1250–1600 lbf)
Isp	Sea level: 248 s Vacuum: 291.3 s	Main: 303 s Verniers: 176 s
Chamber Pressure	75 bar (1088 psi)	98.1 bar (1420 psi)
Nozzle Expansion Ratio	18.9:1	Main: 103.4:1 Verniers: 12.0:1
Propellant Feed System	Gas-generator turbopump	Main: pump fed Verniers: pressure fed
Mixture Ratio (O/F)	2.5:1	2.65:1
Throttling Capability	Yes, unknown range	Yes, unknown range
Restart Capability	No	2 starts
Tank Pressurization	Compressed gas	Compressed gas
Attitude Control		
Pitch, Yaw, Roll	Jet vanes	Verniers, ACS
Staging		
Nominal Burn Time	130 s	Main: 325–335 s first burn, 2–8 s second burn. Verniers: 500 s
Shutdown Process	Burn to depletion	Command shutdown
Stage Separation	2 solid retro-rockets	Payload separation using pneumatic pistons

Vehicle Design

Attitude Control System

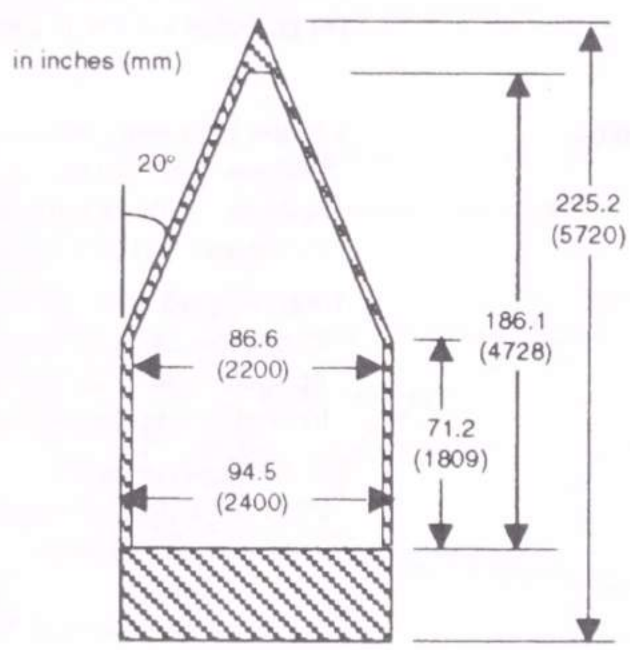
During first-stage flight, vehicle stability is enhanced by four small aerodynamic fins, and the attitude is controlled by a single jet vane in each nozzle. The second-stage propulsion system includes a set of verniers called orientation nozzles. These pressure-fed engines generate 5.6–7.1 kN (1250–1600 lbf) of thrust. They are turned on as the first-stage engine shuts down to maintain control while the second-stage engine develops full thrust, and they remain active for 5–20 s after second-stage shutdown. The second stage also includes an attitude control system that uses 100 N (22 lbf) thrusters to maintain three-axis control during the coast phase. These thrusters are also used for propellant settling before the orbit circularization burn.

Avionics

The Kosmos onboard control system uses an inertial guidance platform to follow preset altitude and velocity profiles. The system controls the engine throttling and gimbaling, generates all ignition, shutdown, and separation commands, and passes data into the telemetry system. The control system is powered by batteries with approximately 4 hours of lifetime. Telemetry is handled by a separate system on each stage with independent batteries. The telemetry system has the capability to store 53 s of basic performance data for periods when the vehicle is not within view of a tracking station. Tracking is facilitated by an onboard transceiver with independent power. In the event of a failure, the control system shuts down the engines for flight termination. No ordnance is used for flight termination.

Payload Fairing

The payload fairing has 14 access doors in the cylinder and conical sections with widths up to 522 mm (20.5 in.). While some of these are used for accessing the fairing separation system or providing conditioned air, the remaining doors provide access to the payload. The fairing is separated by cutting a clamp band and opening mechanical locks that join the two fairing halves. Springs in the base push the two halves open on hinges. The standard Kosmos fairing does not have acoustic blankets, but they can be added for commercial launches.



Length	5.72 m (18.8 ft)
Primary Diameter	2.4 m (7.9 ft)
Mass	348 kg (767 lbm)
Sections	2
Structure	Skin-stringer
Material	Aluminum

Payload Accommodations

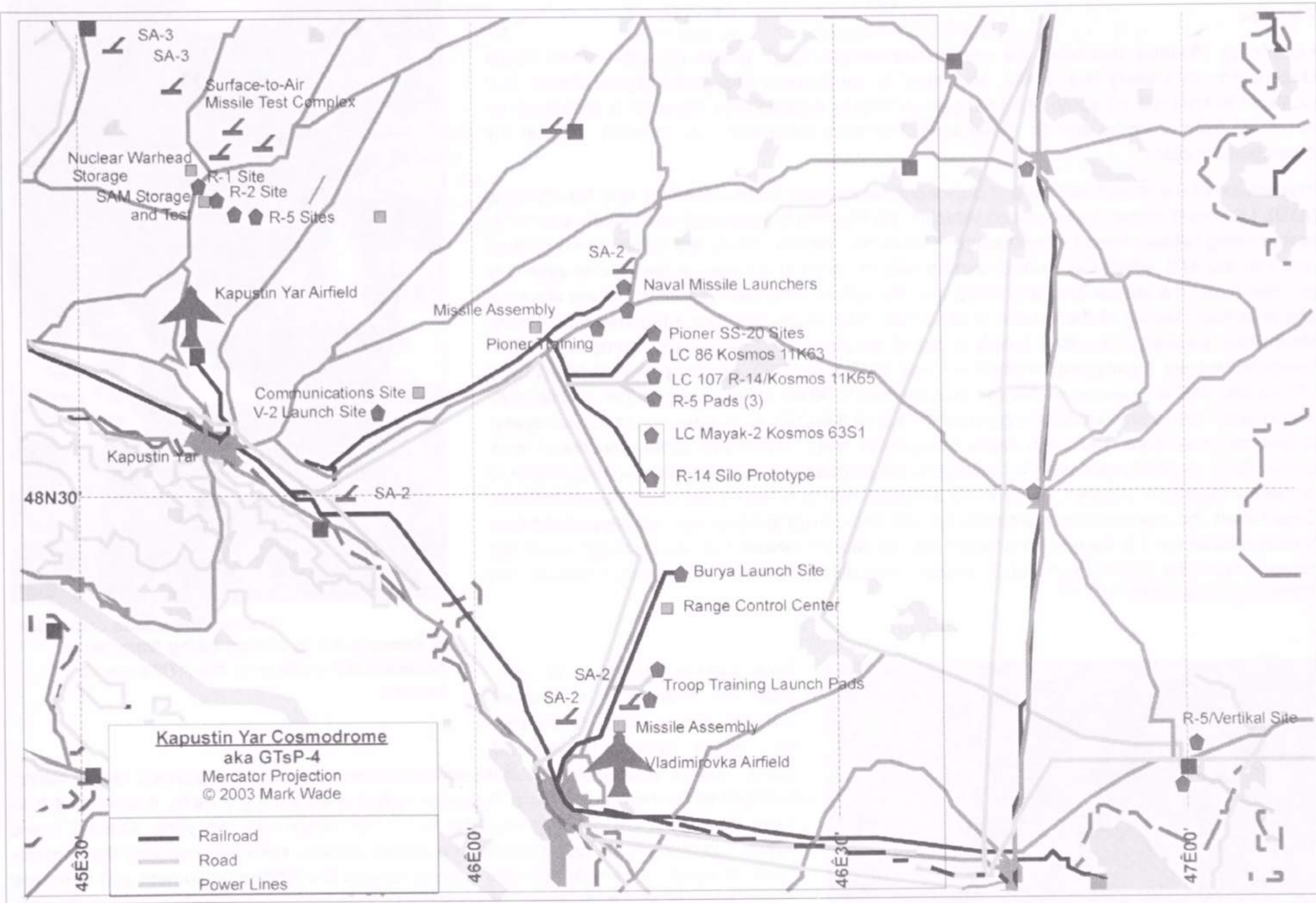
Payload Compartment	
Maximum Payload Diameter	2200 mm (86.6 in.)
Maximum Cylinder Length	1809 mm (71.2 in.)
Maximum Cone Length	2919 mm (114.9 in.)
Payload Adapter Interface Diameter	1060 mm (41.7 in.)
Payload Integration	
Nominal Mission Schedule Begins	T−12 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T−0
On-Pad Storage Capability	30 days for a fueled vehicle
Last Access to Payload	T−4 h through access doors
Environment	
Maximum Axial Load	+6.9 *g* axial
Maximum Lateral Load	±1.4 *g* lateral
Minimum Lateral/Longitudinal Payload Frequency	10 Hz / 10 Hz
Maximum Acoustic Level	128.0 dB at 100–200 Hz
Overall Sound Pressure Level	137.5 dB
Maximum Flight Shock	100–200 *g* at 500–2000 Hz
Maximum Dynamic Pressure on Fairing	42.8 kN (894 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	600 W/m^2
Maximum Pressure Change in Fairing	?
Cleanliness Level in Fairing	Class 10,000 or 100,000 for integration No particles > 5 μm in pad conditioned air
Payload Delivery	
Standard Orbit Injection Accuracy (99.3% probability)	470 km (254 nmi), 97.3 deg: Perigee: +6/−40 km (+3/ -21 nmi) Apogee: +30/−8 km (+16/ -4 nmi) Inclination: +0.03 /−0.06 deg 1000 km (540 nmi), 53 deg: Perigee: +34/ −55 km (+18/−29 nmi) Apogee: +36/ −27 km (+19/−14 nmi) Inclination: +0.04 / −0.08 deg
Attitude Accuracy (3 sigma)	For 800 kg spacecraft: Pitch: ±0.5 deg, ±3 deg/s Yaw: ±3 deg, ±3 deg/s Roll: ±0.5 deg/s Pointing is measured at ACS shutdown, tip-off rates are after separation.
Nominal Payload Separation Rate	Variable, typically 1.1–1.7 m/s (3.6-5.6 ft/s)
Deployment Rotation Rate Available	0 rpm
Loiter Duration in Orbit	3 h
Maneuvers (Thermal/Collision Avoidance)	Solid-rocket motor fires for collision avoidance.
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Kosmos routinely carries up to eight spacecraft, and can be used to deploy LEO constellations.
Auxiliary Payloads	Space is available inside the adapter below the separation plane, and extra performance capacity is available on many Kosmos missions. The volume inside the adapter is roughly the shape of a truncated cone 610 mm (24.0 in.) high, with an upper diameter of 1060 mm (41.7 in.) and a lower diameter of 620 mm (24.4 in.). Typical spacecraft mass for this space would be 50 kg (110 lbm). Larger auxiliary payloads can also be carried as load-bearing structures beneath the primary payload.

Production and Launch Operations

Production

Production of Kosmos has stopped, and all remaining flights are drawn from a stockpile of fully assembled vehicles. Kosmos 3M was manufactured by PO Polyot of Omsk, Russia. NPO Energomash provided the first-stage engine, while KB Khimmash provided the second-stage engine. Production could be restarted for an order of 10 or more vehicles.

Launch Facilities



Courtesy Mark Wade.

Kapustin Yar Cosmodrome

Production and Launch Operations

Launch Operations

Kosmos launches have been conducted from Baikonur, Plesetsk, and Kapustin Yar, making Kosmos the only launch vehicle to have operated from all three of the former Soviet launch sites. (Operations from the new Russian site at Svobodny are not planned.) Twelve test flights of the Kosmos 2 and 3 were conducted from Baikonur between August 1964 and June 1968. Operational Kosmos 3M launches are conducted from Plesetsk and Kapustin Yar. The majority of launches have been conducted from Plesetsk. No orbital launches were conducted from Kapustin Yar between 1987 and 1998, though suborbital Kosmos launches continued to occur infrequently. A commercial Kosmos launch was conducted from Kapustin Yar in the spring of 1999, and additional launches may occur in the future. The Russian military provides range and integration support for Kosmos launches at both Plesetsk and Kapustin Yar. Kosmos launches are performed with a degree of operational routine that is unmatched by any other launch system. Since 1980 there have been more than 200 Kosmos launches, and yet fewer than 2% of them took place on a weekend.

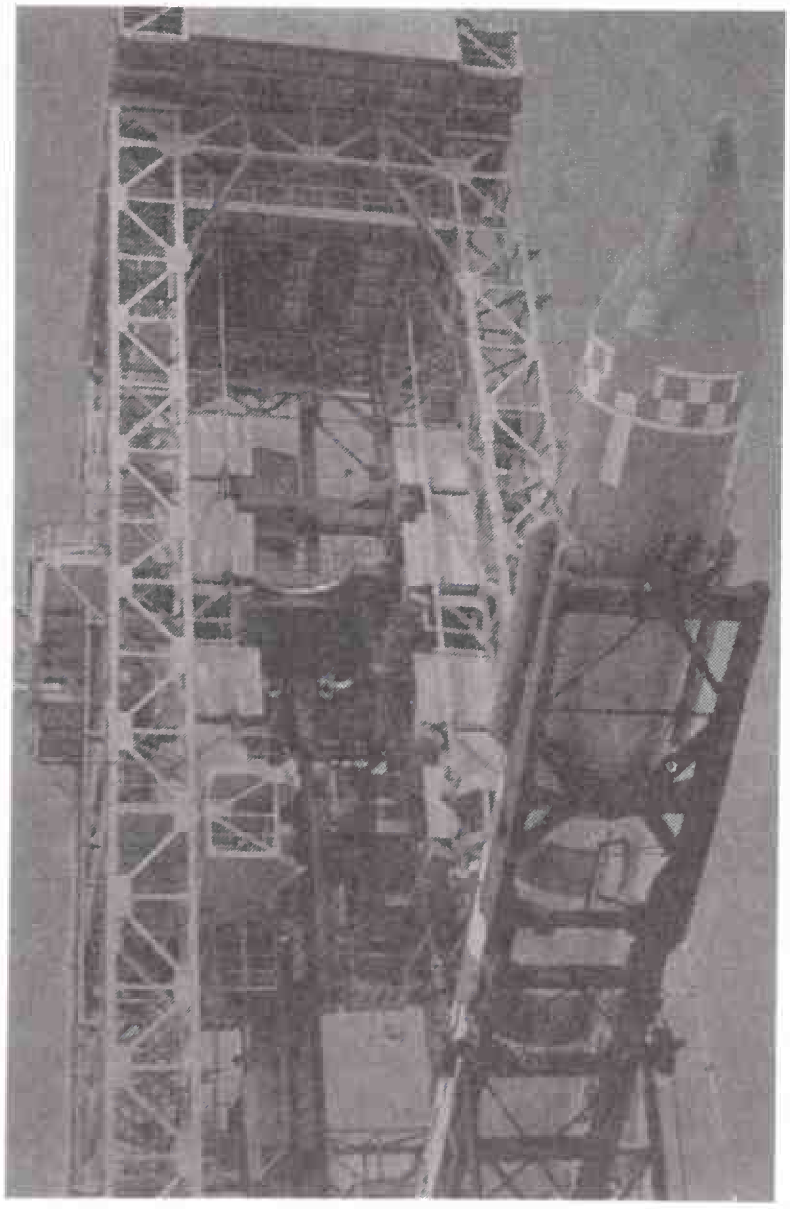
Plesetsk

Historically, Plesetsk has been the world's busiest spaceport. It was also the former Soviet Union's primary military launch site, equivalent to Vandenberg AFB in the United States. It is located 170 km (105 mi) south of Archangel in a heavily wooded area. Plesetsk is accessible by air or rail from Moscow and St. Petersburg. For more information on Plesetsk, consult the Spaceports chapter.

Preparation of the launch vehicle and payload(s) takes place in the Assembly and Test Building (ATB). Like most former Soviet launch vehicles, the Kosmos is integrated horizontally and transported using railroad-based infrastructure. The launch vehicle stages are delivered on railroad cars into the ATB, which has several parallel sets of tracks in the floor of the vehicle assembly and test area. The stages are transferred from the railcars onto testing trolleys, where electrical and pneumatic testing of the vehicle is performed. The stages are then integrated horizontally. Meanwhile, payload processing occurs in one of two spacecraft processing rooms in the ATB. Temperatures are maintained between 15° and 25°C (59–77°F). Facility power at 50 Hz, 220 VAC or 380 VAC is available, or this can be converted to 60 Hz 110 VAC power. Two mobile, soft-walled clean rooms are available to provide two 3.5×3.5 m (12×12 ft) class 100,000 clean areas, or one combined area. The clean rooms can provide class 10,000 with some reduction in work space. Once payload processing is completed, the payload is bagged to maintain cleanliness. If propellant loading is required, the payload is transported to a fueling cell in a different building. Once fueled, the spacecraft is returned to the ATB and moved to the launch vehicle/payload integration area where it is aligned horizontally with the launch vehicle. The mobile clean rooms are placed around the payload and launch vehicle. The payload is mated to the launch vehicle, and the fairing is installed.

Courtesy United Start Corporation.
A Kosmos 3M is shown being transferred between rail trolleys in the ATB prior to launch.

Once vehicle integration is complete, the Kosmos is transferred from the testing trolley to a transport trolley using an overhead crane. The transport trolley provides environmental control to the payload fairing during transport to the launch pad. At a maximum speed of 20 km/h (12 mph), it takes 2–4 h to travel the 40 km (25 mi) from the ATB to LC 132, which has two pads. About 7 h are required to prepare the pad to receive the launch vehicle, including preparing the communications systems, remote operations systems, and fire-fighting equipment and filling the fueling systems and compressed gas cylinders. Once the launch vehicle arrives at the pad, on-pad processing requires 120–135 people and 8–10 h. First, the transport trolley and launch vehicle are erected into the vertical position. Electrical and fueling lines are connected, and the vehicle control and telemetry systems are tested. The vehicle is then prepared for fueling. At T−4 h, all spacecraft operations are halted, and fueling begins. At T−1 h, the fairing air conditioning is disconnected, the mobile service tower is removed, and the launch vehicle is rotated to the flight azimuth on the launch mount. The prelaunch operations are completed by T−17 m, and at T−0, the launch conductor commands launch by pressing the launch button at the command center control desk.



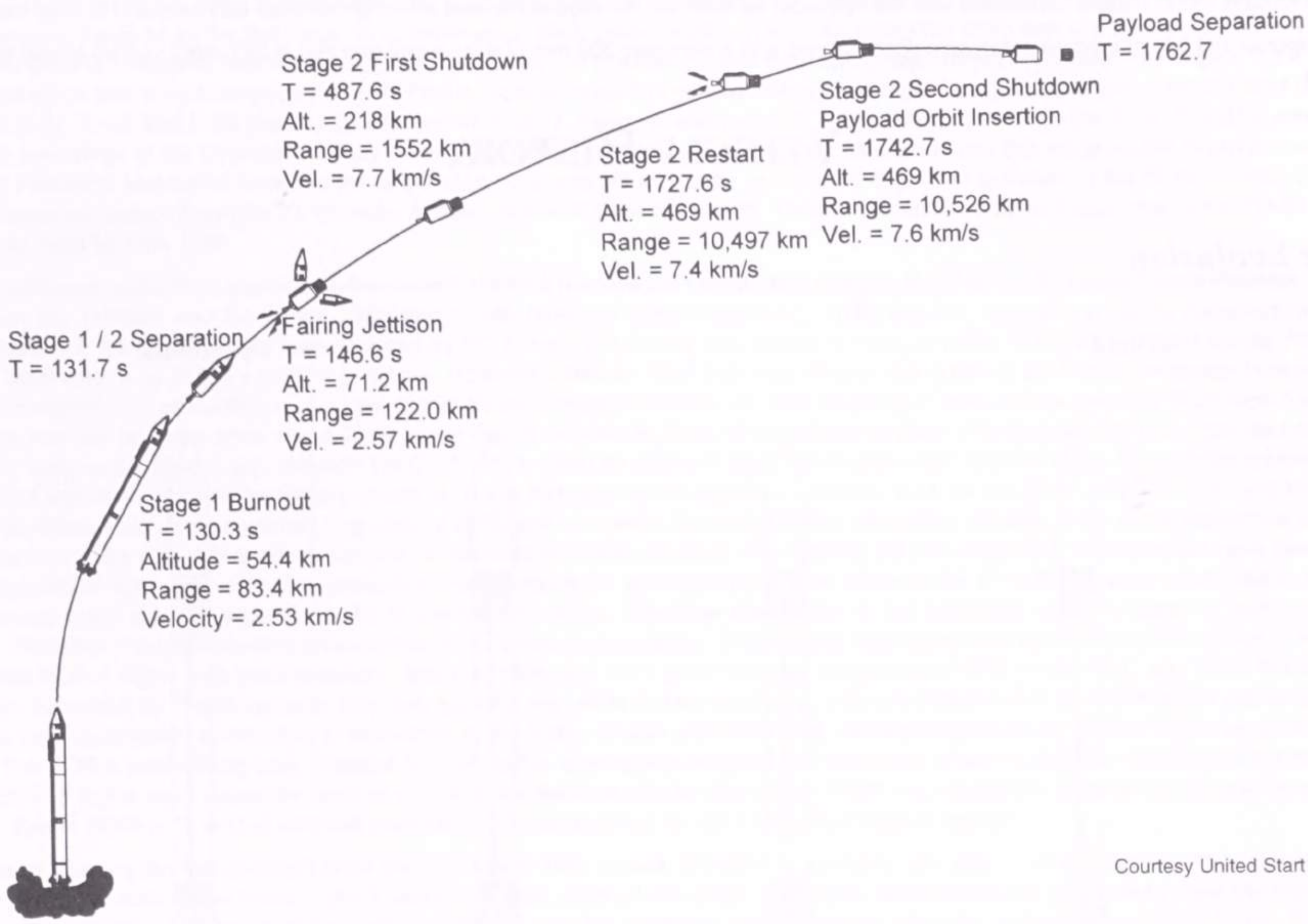
Courtesy United Start Corporation.
Kosmos vehicle being tilted into the launch pad services tower.

Kapustin Yar

Kapustin Yar, also known as GTsP-4 (State Test Range 4), was one of the first long distance missile test ranges in the former Soviet Union, and was later used for space launches of the Kosmos launch vehicle, as well as the 11K63/SL-7 launch vehicle. Kapustin Yar has been the least active of the three Soviet cosmodromes, and orbital launches stopped completely in 1987. In 1999, Kapustin Yar was reopened for space launches to support the commercial launch of the German ABRIXAS X-ray science satellite on Kosmos. Kapustin Yar has become more attractive because it allows Kosmos to reach inclinations of roughly 50 deg from a launch site on Russian soil. Launch operations at Kapustin Yar are similar to operations at Plesetsk, using one pad at LC 107.

Production and Launch Operations

Flight Sequence



Courtesy United Start Corporation.

Event sequence for 470 km (254 nmi) sun-synchronous orbit mission

Events	Time, s
Liftoff	0.0
First-stage shutdown preliminary command	130.3
Second-stage vernier engine ignition command	130.3
First-stage separation	131.7
Second-stage main-engine ignition command	134.7
Fairing jettison	146.6
Second-stage main-engine shutdown command	477.0
Second-stage vernier-engine shutdown command	487.6
Second-stage second vernier ignition command	1727.6
Second-stage second main-engine ignition command	1729.7
Second-stage second main-engine shutdown command	1732.4
Second-stage second vernier-engine shutdown command	1742.7
Spacecraft separation	1762.7

Vehicle Upgrade Plans

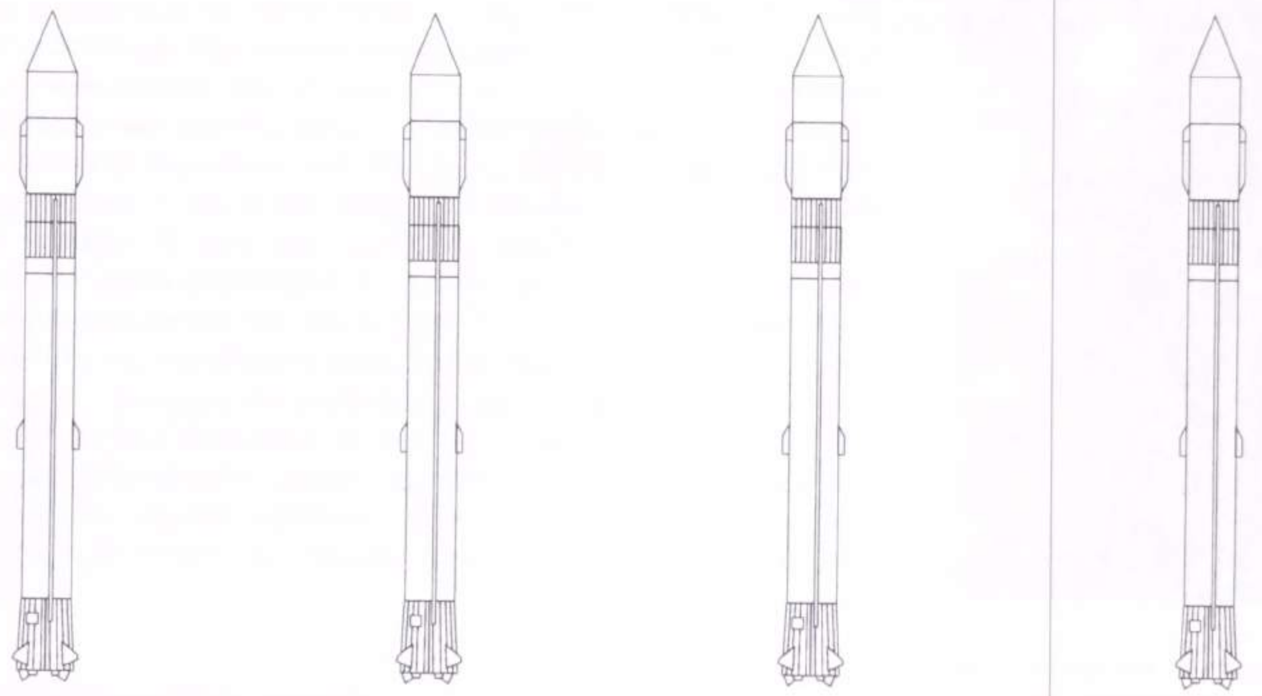
AKO Polyot has developed a proposal for an upgraded Kosmos designated Kosmos U or Kosmos 3M-U. The upgrade would include improved, modernized avionics and control systems, a reduction in the residual propellants in each stage, improvements to the first- and second-stage engines, a small increase in the length of the first stage, and upgrades to the ground support and launch facilities. The benefits would include a 150–180 kg (330–400 lbm) increase in performance, capability for yaw steering to reach other orbit inclinations, and improved orbit injection accuracy. The environmental impact of Kosmos launches would be reduced as a result of smaller impact zones and reduced propellant residuals in the jettisoned stage. However, there is no funding within Russia to proceed with the upgrade, so work will not proceed unless a foreign investor can be found to fund the project.

Cosmos International has offered an enlarged payload fairing with a diameter 200 mm (7.9 in) larger and that is 327 mm (12.9 in) longer than the existing fairings.

Vehicle History

Vehicle Evolution

Retired | **Operational**



Vehicle	**Kosmos 2**	**Kosmos 3**	**K65M-RB**	**Kosmos 3M**
Period of Service	1964–1965	1966–1968	1982–1984 ?	1967–Present
LEO Payload	1500 kg (3300 lbm) ?	1500 kg (3300 lbm) ?	(Suborbital)	1500 kg (3300 lbm)

Kosmos 2 and 3 configuration and performance are unknown, but assumed to be similar to Kosmos 3M.

Vehicle Description

Name	Article Number	Description
•Kosmos 2	65S3	Development version retired after eight flights.
•Kosmos 3	11K65	Development version retired after four flights.
?	K65M-RB	Variation of Kosmos 3M used for tests of BOR spaceplanes.
•Kosmos 3M	11K65M	Primary operational version used for space launch.

Vehicle History

Historical Summary

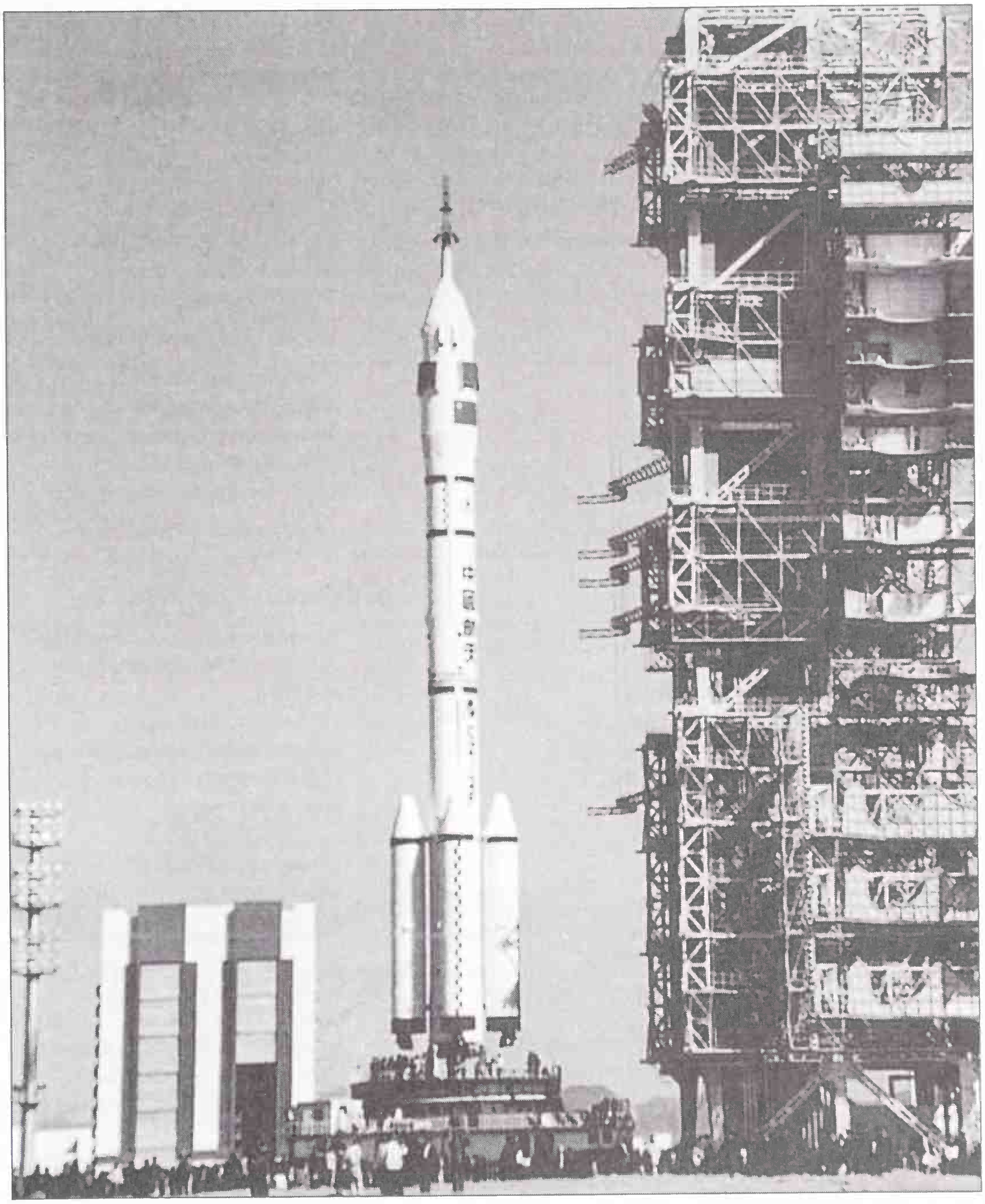
The history of the Kosmos began with the development of the R-14 IRBM (article 8K65, NATO designation SS-5 Skean) at Mikhail Yangel's Design Bureau starting in 1958. The bureau was created in Dnepropetrovsk, Ukraine, in 1954 with the designation SKB-586, and is now known as KB Yuzhnoye. It has specialized in producing strategic missiles. Static tests of the R-14 began on 28 March 1960, and flight tests from Kapustin Yar were conducted from June to December of that year. In April 1961, the R-14 was declared operational and development work began on the 65S3 Kosmos 2 satellite launch vehicle using the R-14 as the first stage. In approximately 1962, responsibility for manufacturing the 65S3 was apparently shifted from the Yangel Design Bureau to NPO Prikladnoi Mekhaniki ("Applied Mechanics"). It is assumed that the basic design remained largely unchanged, but it is possible that the second stage was in fact developed at NPO Prikladnoi Mekhaniki. In the early 1960s the Yangel Design Bureau proposed the S5M upper stage for use on the R-12, R-14, and R-36 missiles (predecessors to SL-7, Kosmos, and Cyclone, respectively), as well as the R-16. The S5M was eventually used as the third stage of the Cyclone 3. It may be that the Yangel Design Bureau originally planned to use this stage as the Kosmos second stage, and that NPO Prikladnoi Mekhaniki designed the larger stage that was actually used on Kosmos. Eight test launches of the 65S3 Kosmos 2 were conducted from Baikonur Launch Complex 41 between August 1964 and December 1965. These were followed by four launches of the 11K65 Kosmos 3 from November 1966 to June 1968.

The Kosmos achieved operational status with the current 11K65M Kosmos 3M version beginning in May 1967 with a launch from Plesetsk. The differences between the 11K65M and the earlier 65S3 and 11K65 have not been determined. While the first two versions were manufactured by NPO Prikladnoi Mekhaniki, the 11K65M was manufactured by PO Polyot. PO Polyot was formed in 1941 in Omsk to produce aircraft during World War II. Aircraft have continued to be Polyot's primary business. However, between 1954 and 1965, Polyot was a part of the Yuzhnoye design bureau, and thus was given responsibility for production of the operational Kosmos launch vehicles, as well as the SL-7 and various missiles. Yuzhnoye has not been involved in the Kosmos program since about 1965, when Polyot reverted to being an autonomous plant. The Kosmos 3M has a payload capability of 1500 kg (3300 lbm), which filled a gap between the SL-7, which could lift only 600 kg (1300 lbm) to orbit, and the larger Soyuz/Vostok/Molniya family. The Kosmos 3M was used primarily for military missions. These included communications systems, such as the Strela series of store and forward communications satellites, which were launched in groups of eight, and navigation systems. Civilian navigation systems, such as the Nadezhda and Tsikada navigation satellites were also launched on Kosmos. In addition to orbital missions, the Kosmos 3M and remaining R-14 missiles have been used for roughly 300 suborbital flights, generally for research in hypersonic flight and reentry systems. Many of the R-14 flights were conducted in the Vertikal program of collaborative science experiments by Soviet bloc countries. The most well known of the suborbital missions were the flights of the BOR (Becpilotniye Orbitalnie Raketoplan—unmanned orbital rocketplane) spaceplanes. These flights were performed by the K65M-RB version of the Kosmos 3M. Four orbital BOR-4 flights took place between 1982 and 1984, and were given Kosmos designations 1374, 1445, 1517, and 1614. These launches are considered suborbital by Polyot because they only lasted a few orbits before reentering, and are therefore not listed in Polyot's orbital launch logs. All four flights were successful, culminating in recoveries in the Indian Ocean and Black Sea. At least two suborbital BOR-4 flights were also launched on Kosmos. The BOR-4 was a lifting body adapted from an earlier spaceplane program and was used to test heatshield materials for the Buran space shuttle. The BOR-5 flights were suborbital tests of a 1/8th scale Buran space shuttle orbiter, which were launched to study the hypersonic entry profile of the Buran. Earlier BOR-1, -2, and -3 vehicles were lifting bodies launched by the 11K63/SL-7 launch vehicle.

The flight rate of Kosmos 3M has declined since the mid-1980s, from a peak of nearly 30 launches per year to an average of two per year. Many former Kosmos payloads, such as the Strela 1 and 2 series satellites, evolved into larger spacecraft, and subsequent generations were transferred to larger Cyclone launch vehicles. As a result of the reduced demand, and the decline in military funding after the collapse of the Soviet Union, Kosmos production ended by 1995. Since then launches have been performed using rockets taken out of storage. When the supply of existing rockets is exhausted, military payloads will be transferred to the Rockot launch vehicle. The end of the Kosmos may be hastened as a result of it regaining some of its old military payloads. In July 2002 Kosmos launched a pair of Strela 3 satellites, which are normally launched six at a time on Cyclone 3 vehicles. Strela launches are being transferred back to Kosmos because of a pair of Cyclone 3 failures and the limited number of available Cyclones.

Commercial launches have become more important to the Kosmos program as Russian use has decreased. The first launch of a U.S. satellite on a Russian space launch vehicle was performed by a Kosmos 3M in January 1995. Along with the Tsikada primary payload, the Kosmos booster carried the U.S. satellite FAISAT-1, a demonstrator for a store and forward LEO communications constellation planned by Final Analysis, Inc. The launch also carried the Astrid scientific microsatellite for Sweden. In recent years roughly half of Kosmos launches have carried commercial spacecraft, primarily European scientific and engineering satellies. Commercial Kosmos launches are marketed by Cosmos International and by Puskovie Uslugi, and its U.S. partner Assured Space Access through their joint venture United Start Corporation.

LONG MARCH



Courtesy China Great Wall Industry Corporation.

The Long March family of space launch systems is used to launch all of China's satellites. It includes large and small launch vehicles for LEO or GTO missions. China's newest vehicle, the Long March 2F (above), is capable of carrying humans to orbit.

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General Description

Long March 2C

Long March 2C

Long March 2C/SD or 2C/CTS

Summary

The Long March 2C is a direct descendent of China's DF-5 (CSS-4) ICBM, adapted for space launch missions. The LM-2C has been in use since 1975 and has been launched more often and more reliably than any other Long March configuration. In the mid 1990s the LM-2C was modified into the LM-2C/SD with a stretched second stage and a new third stage called the Smart Dispenser to launch pairs of Iridium satellites. The LM-2C is currently offered with a modified version of this stage called the CTS.

Status

LM-2C: Operational. First launch in 1975.

LM-2C/SD: Operational. First launch in 1997.

Origin

China

Key Organizations

Marketing Organization	China Great Wall Industry Corporation (CGWIC)
Launch Service Provider	China Satellite Launch and Tracking Control General (CLTC)
Prime Contractor	China Academy of Launch Vehicle Technology (CALT)

Primary Missions

Small payloads to LEO

Estimated Launch Price

$20–25 million (FAA)

Spaceports

Launch Site	Taiyuan Satellite Launch Center (TSLC)
Location	37.8° N, 111.5° E
Available Inclinations	96–98 deg
Launch Site	Jiuquan Satellite Launch Center (JSLC)
Location	41.3° N, 100.3° E
Available Inclinations	56.9–69 deg
Launch Site	Xichang Satellite Launch Center (XSLC)
Location	28.2° N, 102.2° E
Available Inclinations	27.5–31.1 deg

Performance Summary

200 km (108 nmi), 28.5 deg	LM-2C: 4000 kg (8800 lbm)
200 km (108 nmi), 90 deg	LM-2C: 2200 kg (4850 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability (inclination not available)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	LM-2C/CTS: 1600 kg (3525 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 28.5 deg	1400 kg (3100 lbm) with CPKM kick motor
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	22
Launch Vehicle Successes	22
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–1 per year

General Description

Long March 2E and 2F

Long March 2E

Long March 2F

Summary

The LM-2E is a larger version of the LM-2C, with four additional strap-on boosters. It can carry medium payloads to LEO or to GTO using an additional kick stage. The LM-2E was used for many of China's early commercial launches, and it is still marketed, although it has not been launched since 1995. The LM-2E serves as the basis for a new derivative, the LM-2F, which is used in China's human spaceflight program. A series of unmanned test launches began in 1999, and the first launch with a crew occurred in October 2003. The LM-2F is outwardly similar to the LM-2E, with the addition of a Shenzhou space capsule and escape tower on top. The internal systems have been enhanced to increase reliability.

Status

LM-2E: Operational or retired. First launch in 1990.

LM-2F: Operational. First launch in 1999.

Origin

China

Key Organizations

Marketing Organization	China Great Wall Industry Corporation (CGWIC)
Launch Service Provider	China Satellite Launch and Tracking Control General (CLTC)
Prime Contractor	China Academy of Launch Vehicle Technology (CALT)

Primary Missions

Human spaceflight

Estimated Launch Price

?

Spaceports

	LM-2E
Launch Site	Xichang Satellite Launch Center (XSLC) Pad 2
Location	28.2° N, 102.0° E
Available Inclinations	27.5–31.1 deg
	LM-2E and 2F
Launch Site	Jiuquan Satellite Launch Center (JSLC), South Launch Complex
Location	41.3° N, 100.3° E
Available Inclinations	56.9–69 deg

Performance Summary

200 km (108 nmi), 28.5 deg	LM-2E: 9500 kg (20,950 lbm)
200 km (108 nmi), 90 deg	LM-2E: 6800 kg (15,000 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	LM-2E/ETS: 7300 kg (16,100 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	?
GTO: 185×35,786 km (100×19,323 nmi), 28.5 deg	LM-2E: 3500 kg (7710 lbm) with EPKM kick motor
Geostationary Orbit	No capability

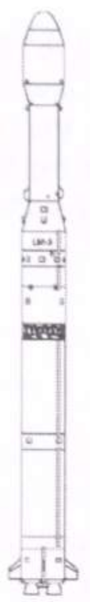
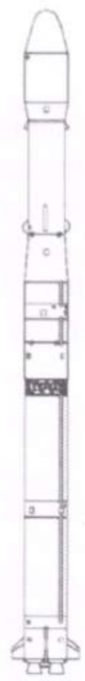
Flight Record (through 31 December 2003)

	LM-2E	LM-2F
Total Orbital Flights	7	5
Launch Vehicle Successes	4	5
Launch Vehicle Partial Failures	1	0
Launch Vehicle Failures	2	0

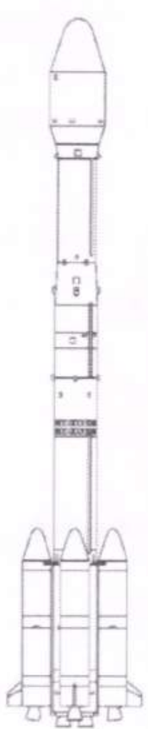
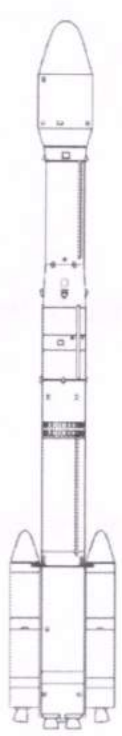
Flight Rate

LM-2E	LM-2F
0 per year	0–2 per year

GENERAL DESCRIPTION

Long March 3 | Long March 3A

Long March 3B | Long March 3C

LONG MARCH 3, 3A, 3B, AND 3C

Summary

The LM-3 series of vehicles includes a cryogenic upper stage to lift spacecraft to GTO. When the LM-3 first flew in 1984, China became only the third nation to use a LOX/hydrogen upper stage. The LM-3 performed China's first commercial launch in 1990. The more recent LM-3A has a larger upper stage. The LM-3B is now China's primary commercial launch vehicle for GTO payloads. It is similar to the LM-3A with four strap-on boosters added for increased performance. The LM-3C is a planned derivative of the LM-3B with only two strap-on boosters.

Status

LM-3: Operational. First launch in 1984. LM-3A: Operational. First launch in 1994. LM-3B: Operational. First launch in 1996.
LM-3C: In development. First launch unknown.

Origin

China

Key Organizations

Marketing Organization	China Great Wall Industry Corporation (CGWIC)
Launch Service Provider	China Satellite Launch and Tracking Control General (CLTC)
Prime Contractors	China Academy of Launch Vehicle Technology (CALT) Shanghai Academy of Spaceflight Technology (SAST)

Primary Missions

GTO

Estimated Launch Price

LM-3: $35–40 million (FAA)
LM-3A: $45–55 million (FAA)
LM-3B: $50–70 million (FAA)
LM-3C: ?

Spaceport

Launch Site	Xichang Satellite Launch Center (XSLC) LM-3: Pad 1 LM-3A, 3B, and 3C: Pad 2
Location	28.2° N, 102.0° E
Available Inclinations	27.5–31.1 deg

Performance Summary

200 km (108 nmi), 28.5 deg	LM-3A: 6000 kg (13,225 lbm) LM-3B: 11,200 kg (24,700 lbm) LM-3C: 9100 kg (20,000 lbm)
200 km (108 nmi), 90 deg	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	LM-3B: 6000 kg (1325 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 28.5 deg	LM-3: 1500 kg (3300 lbm) LM-3A: 2600 kg (5700 lbm) LM-3B: 5100 kg (11,250 lbm) LM-3C: 3800 kg (8400 lbm)
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

	Combined LM-3 series
Total Orbital Flights	26
Launch Vehicle Successes	21
Launch Vehicle Partial Failures	2
Launch Vehicle Failures	3

Flight Rate

0–2 per year

GENERAL DESCRIPTION

LONG MARCH 4B

Long March 4B

Summary

The LM-4B is based on the first two stages of the LM-3 with a small storable propellant third stage to launch spacecraft into sun-synchronous orbits. Unlike other vehicles in the Long March family, the LM-4 series are built by Shanghai Academy of Spaceflight Technology.

Status

Operational. First launch in 1999.

Origin

China

Key Organizations

Marketing Organization	China Great Wall Industry Corporation (CGWIC)
Launch Service Provider	China Satellite Launch and Tracking Control General (CLTC)
Prime Contractor	Shanghai Academy of Spaceflight Technology (SAST)

Primary Missions

Smalll SSO satellites

Estimated Launch Price

\$25–35 million (FAA)

Spaceports

Launch Site	Taiyuan Satellite Launch Center (TSLC)
Location	37.8° N, 111.5° E
Available Inclinations	96–98 deg
Launch Site	Jiuquan Satellite Launch Center (JSLC)
Location	41.3° N, 100.3° E
Available Inclinations	56.9–69 deg

Performance Summary

200 km (108 nmi), 28.5 deg	No capability
200 km (108 nmi), 90 deg	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 900 km (486 nmi)	2800 kg (6170 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 28.5 deg	?
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	6
Launch Vehicle Successes	6
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–2 per year

Nomenclature

China's primary launch vehicle family is named Long March, in reference to the famous march of Mao Tse-tung's Communists across China in 1935. The transliteration from the Chinese is Chang Zheng. As a result, a vehicle such as the Long March 2C may be labeled either LM-2C or CZ-2C. When China began marketing its launch vehicles internationally, the LM designation became the more common of the two abbreviations, but both are still in use. The CZ designation is commonly used for domestic missions.

The Long March family consists of approximately 6–8 active launch vehicle types. The Long March vehicles are divided into four series, LM-1, LM-2, LM-3, and LM-4, with letter designations indicating variations within each series. The small LM-1 performed China's first two orbital launches in 1970 and 1971, but has not been used for orbital launches since. The LM-2 series includes both medium and large launch vehicles for LEO missions. The LM-3 series is distinguished by the use of a cryogenic upper stage for GTO missions. The LM-4 series is used for sun-synchronous missions. The nomenclature of Long March vehicles appears to be determined more by the intended function than by common engineering heritage. For example, the name of the LM-2D would appear to indicate that it is a follow-on to the LM-2C, but in fact, its hardware is closely related to the LM-4 instead. Similarly, the LM-3B includes stages from the LM-2E and LM-4 families, adapted for launching GTO payloads. The function-based naming system is further confused by the fact that some vehicles can be used for multiple purposes. The LM-2E has been used for GTO launches with the addition of a solid upper stage, and the LM-3B has been marketed for LEO and SSO missions.

The LM-4 is also known as the LM-4A. This naming method is not used with the LM-3 series. That is, the LM-3 and LM-3A are distinctly different vehicles.

Cost

China Great Wall has not released official pricing information. Only one contract price has been publicly disclosed: the Korea Aerospace Research Institute reportedly agreed in 2001 to pay $16.9 million to launch Kompsat 2 on a Long March 2C/CTS. However, Kompsat 2 was later shifted to a Rockot launch. In 2002 the FAA estimated the following typical price ranges for Long March launch services using publicly available data. These estimates are uncertain because of the limited number of Long March contracts.

Launch Vehicle	Estimated Price
LM-2C	$20–25 million
LM-2E	?
LM-2F	?
LM-3	$35–40 million
LM-3A	$45–55 million
LM-3B	$50–70 million
LM-3C	?
LM-4B	?

Chinese launch prices have been lower than in Western countries because of radically lower labor costs. In 2000 the chief designer at the China Academy of Launch Vehicle Technology had a monthly salary of about 3000 CNY ($360). In recent years the economic development of China has made higher paying jobs available for workers with technical skills, putting upward pressure on salaries in the space program.

Availability

Long March launch services are marketed internationally by China Great Wall Industry Corporation (CGWIC). However, the use of Long March for commercial launches has effectively been stopped by the United States. In the late 1990s, the U.S. government became concerned that China's long-range missile program was benefiting from technology transfer from commercial launch services agreements with U.S. customers. Although there is not a formal ban on Chinese launches, the United States has not granted export licenses to China for satellites built in the United States or having U.S. components. CGWIC has been awarded several launch contracts in recent years, but the satellites, such as AtlanticBird 1 and APR-3, have had to shift to other vehicles, so no commercial Long March launches have actually taken place since 1999.

The Long March family includes six to eight active configurations, down from about a dozen in the mid 1990s. After a string of failures, China concluded that there were too many variations of Long March, making it difficult to achieve the affordability and reliability that can result from higher flight rates. In some cases it is unclear which configurations have been retired, since years can go by between flights of some configurations, and some continue to be advertised long after they last flew. In recent years China has primarily used three configurations: the LM-4B for sun-synchronous satellites, the LM-3A for domestic geosynchronous satellites, and the LM-2F for its nascent human spaceflight program. The LM-3B and LM-2C were heavily used in the late 1990s for commercial launches. Although neither has launched recently, both have been chosen for recent foreign launch contracts so they are known to be available. The LM-3C has never been launched, but was selected for AtlanticBird 1 before that satellite shifted to Ariane. Given the similarity to the LM-3B, the LM-3C could presumably be launched whenever an appropriate payload becomes available. The LM-2E has not been launched since 1995 and may have been retired. However, it is still marketed and is similar to the LM-2F for which little direct data are available. The basic LM-3 has been explicitly dropped from the commercial market in favor of its more powerful descendents, however it may still have a few more flights scheduled for government missions.

Performance

Long March launch vehicles are launched from three launch complexes in China. Because all three sites are inland, near populated areas, only certain launch azimuths are available. For example, GTO launches from Xichang use a 97-deg launch azimuth. The more modern Long March configurations are capable of performing yaw steering maneuvers to reach nonstandard inclinations at the cost of reduced performance.

LM-2C

The basic two-stage LM-2C can only reach circular orbits below 400–500 km (215–270 nmi) because the second stage is not restartable. To reach higher orbits, a new third stage, the CTS, is used. The first two stages deliver the CTS and payload to an elliptical orbit, and the CTS fires a small solid motor at apogee to circularize the orbit. Performance with and without the CTS is shown. When the third stage was added in the mid 1990s for Iridium deployment missions (it was then called the Smart Dispenser), the LM-2C was upgraded with stretched propellant tanks. Performance for earlier LM-2C vehicles was lower than shown here.

LM-2E

Like the LM-2C, the LM-2E can reach circular orbits only at low altitudes. Performance capabilities of the LM-2F are not known with certainty but are similar to the LM-2E. A third stage has been proposed for the LM-2E using a similar design to the CTS, but with a larger motor. Performance in LEO for the two-stage and three-stage versions are shown. A kick stage called the EPKM has also been used to deliver payloads to GTO. Performance is shown for a standard GTO orbit, a raised apogee GTO orbit, and a standard orbit with a reduced probability of command shutdown during injection into the LEO parking orbit.

LM-3B

The LM-3B is used primarily for GTO launches, but its cryogenic upper stage also provides high performance for LEO orbits as well. The standard 5100 kg (11,250 lbm) GTO performance level assumes that the spacecraft is encapsulated in the payload fairing on the launch pad. In this case, the LM-3B avionics are mounted on a vehicle equipment bay (VEB), which consists of a flat plate encircling the payload adapter inside the payload fairing. If the spacecraft is encapsulated offline in a payload processing facility, the avionics cannot be mounted around the payload adapter. Instead the VEB configuration is a load-bearing cylindrical structure, which is placed above the third stage and below the encapsulated payload adapter/fairing. The additional structural mass of this configuration reduces performance of this configuration by 100 kg (220 lbm). The same is true for the LM-3C.

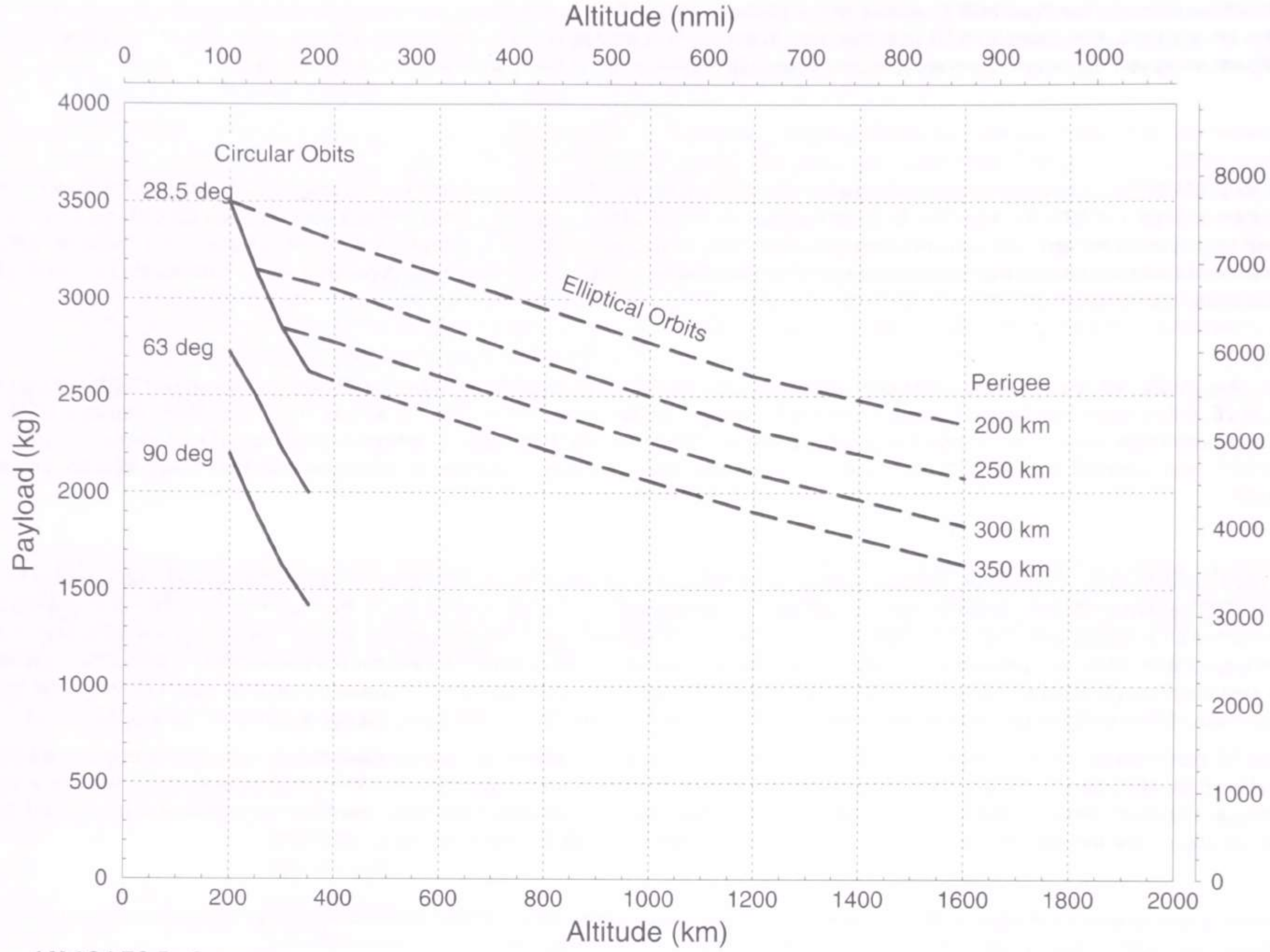
LM-3A, 3B, and 3C performance can be customized in two different ways. The launch vehicle can deliver spacecraft to subsynchronous or supersynchronous transfer orbits. Alternatively, the flight performance reserve can be reduced from the standard 99.73% (3 sigma) level, to a lower value or to a minimum residual shutdown (MRS). In the case of the MRS, the third stage burns to propellant depletion, resulting in a higher performance but also larger possible variation in the transfer orbit apogee. The MRS option increases GTO performance by 100 kg (220 lbm)

Long March vehicles cannot achieve 90-deg-inclination polar orbits and 51.6-deg space station orbits from existing launch sites unless dogleg or plane change maneuvers are used. Contact CGWIC for more information on these types of missions.

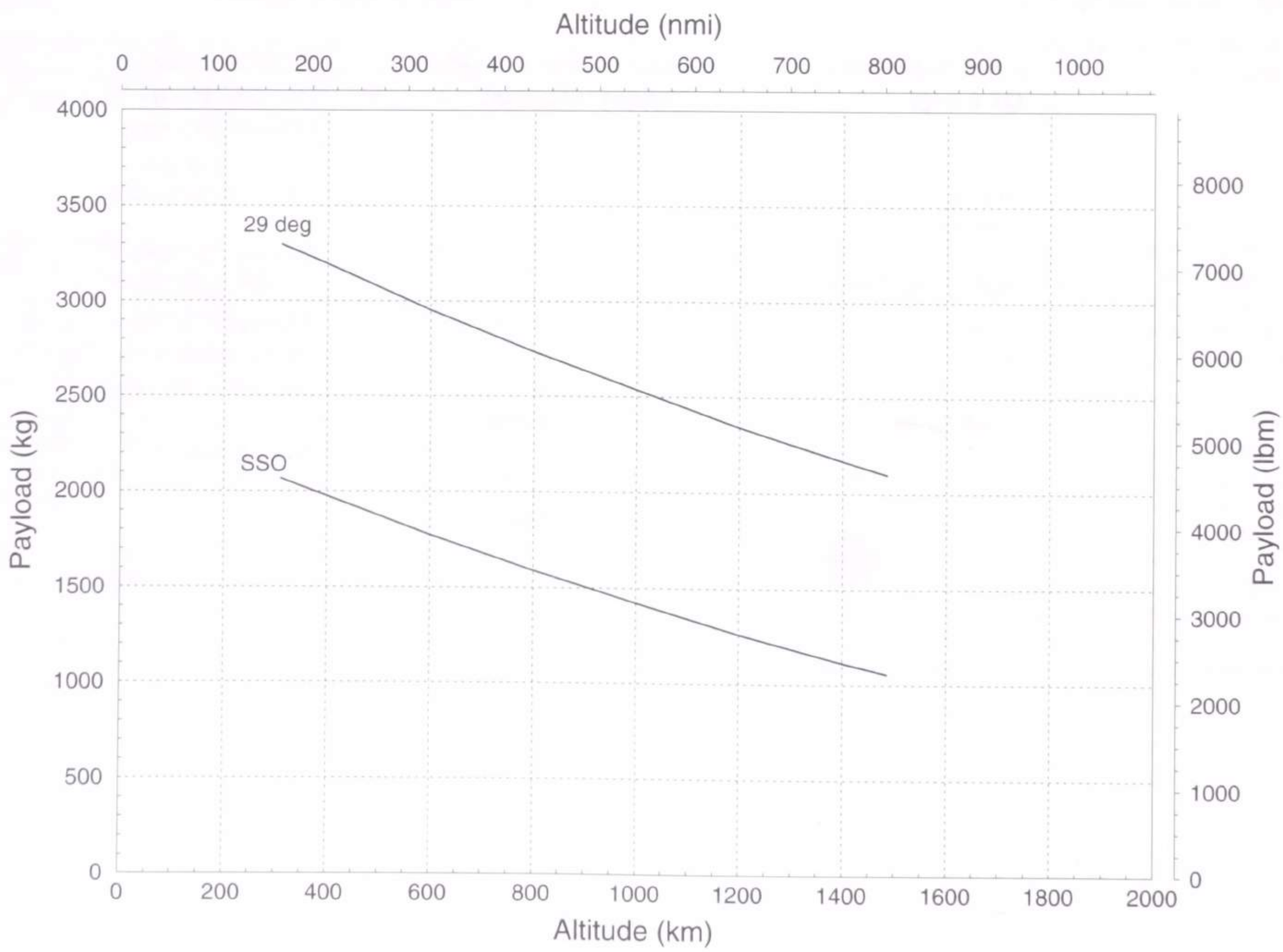
LM-3B performance assumes on-pad encapsulation, guidance command shutdown, and use of a 4.0-m (13.1-ft) fairing. Payload adapters are not accounted for in performance estimates. Flight performance reserve for 99.7% probability of command shutdown is assumed.

Vehicle	**200 km (108 nmi) (28.5 deg)**	**Sun-Synchronous orbit (Various altitudes)**	**GTO: 28.5 deg 200×35,786 km (108×19,323 nmi)**
LM-2C	3900 kg (8600 lbm)	?	1400 kg (3100 lbm) with CPKM kick motor
LM-2E	9500 kg (20950 lbm)	?	3500 kg (7710 lbm) with EPKM kick motor
LM-3	?	?	1500 kg (3300 lbm)
LM-3A	?	?	2600 kg (5700 lbm)
LM-3B	11,200 kg (24,700 lbm))	800 km (432 nmi): 6000 kg (13,250 lbm)	5100 kg (11,250 lbm)
LM-3C	?	?	3800 kg (8400 lbm)
LM-4B	?	900 km (486 nmi) 2800 kg (6170 lbm)	?

PERFORMANCE

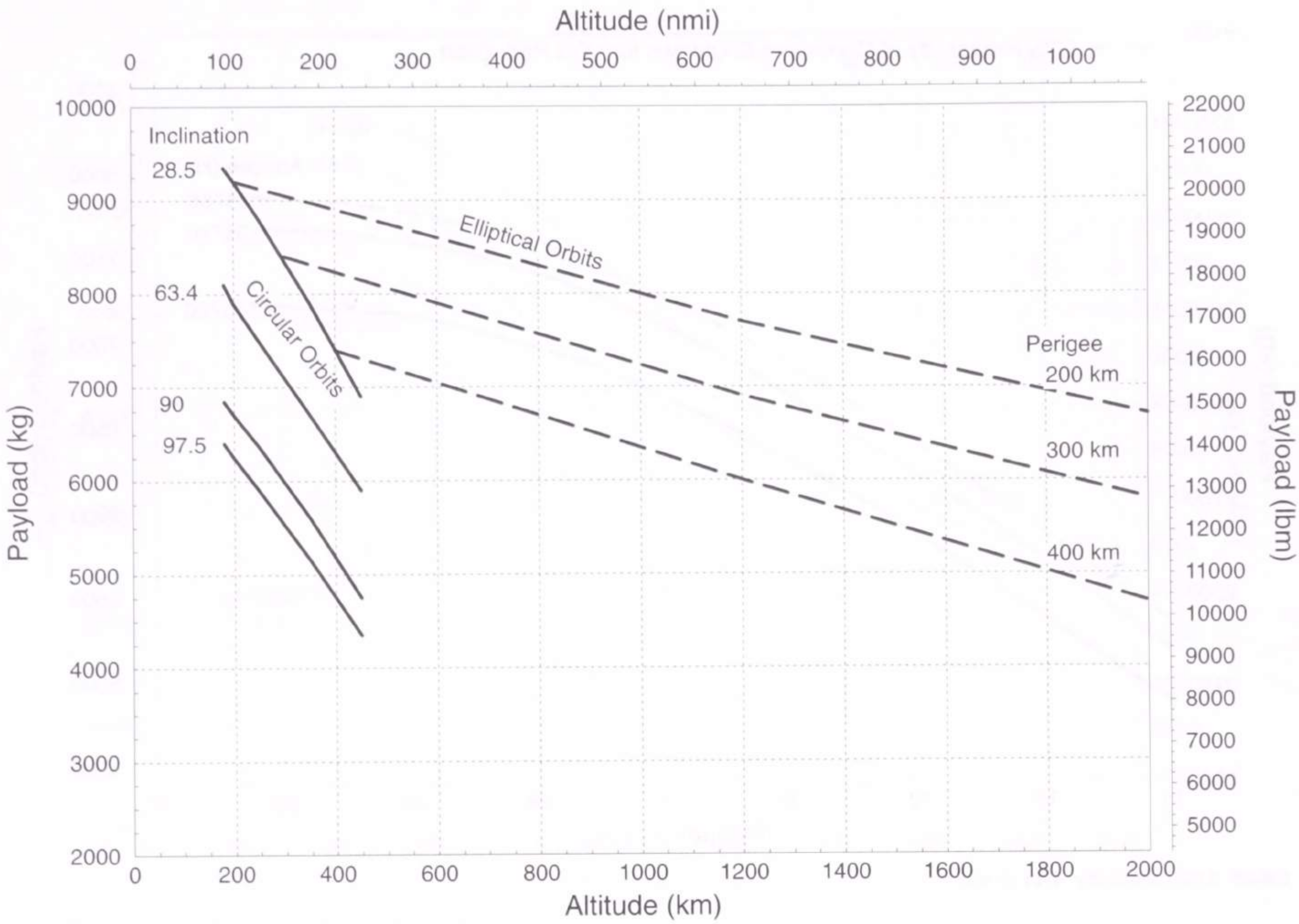


LM-2C LEO Performance

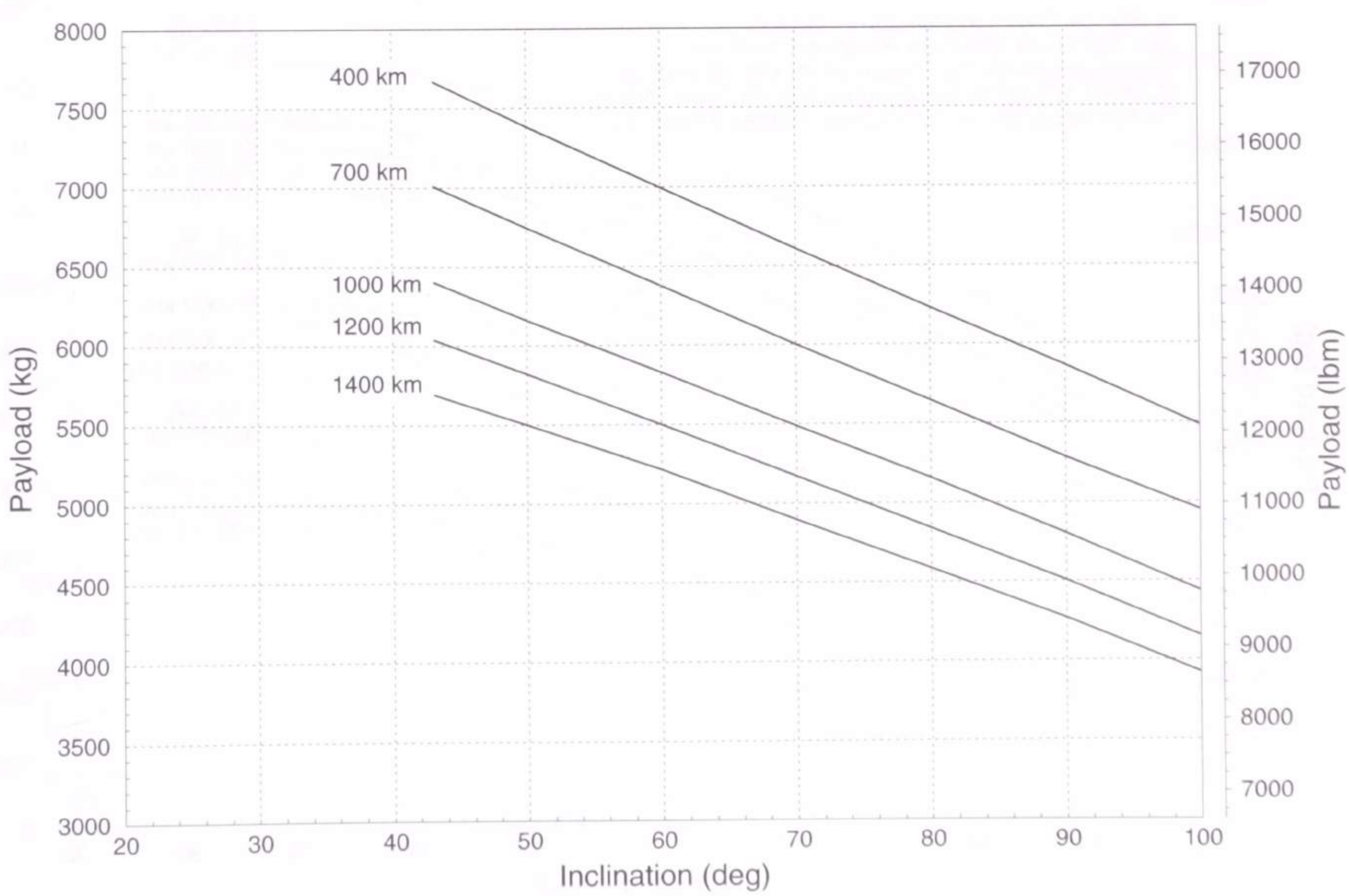


LM-2C/CTS Performance in LEO

PERFORMANCE

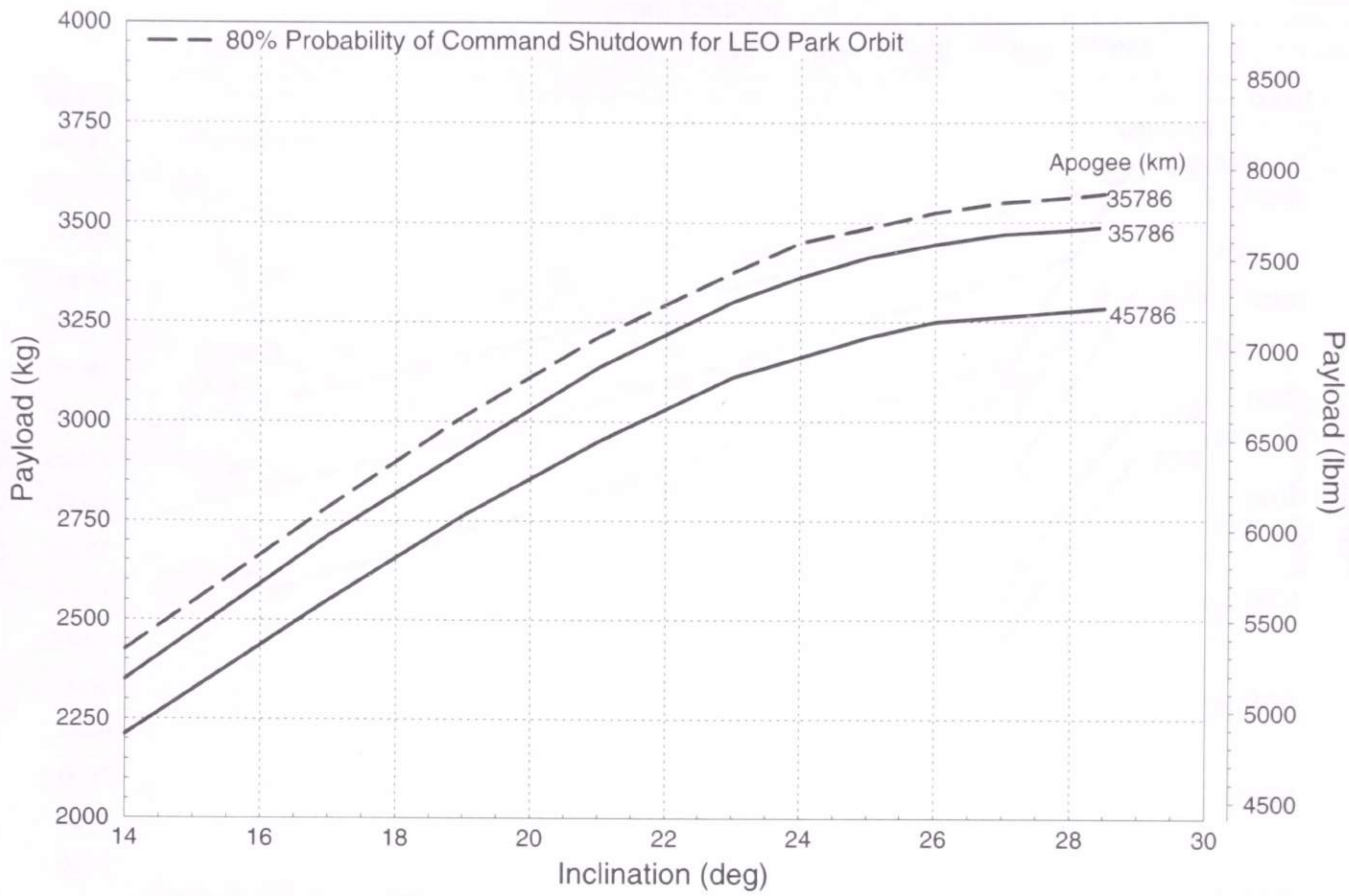


LM-2E LEO Performance

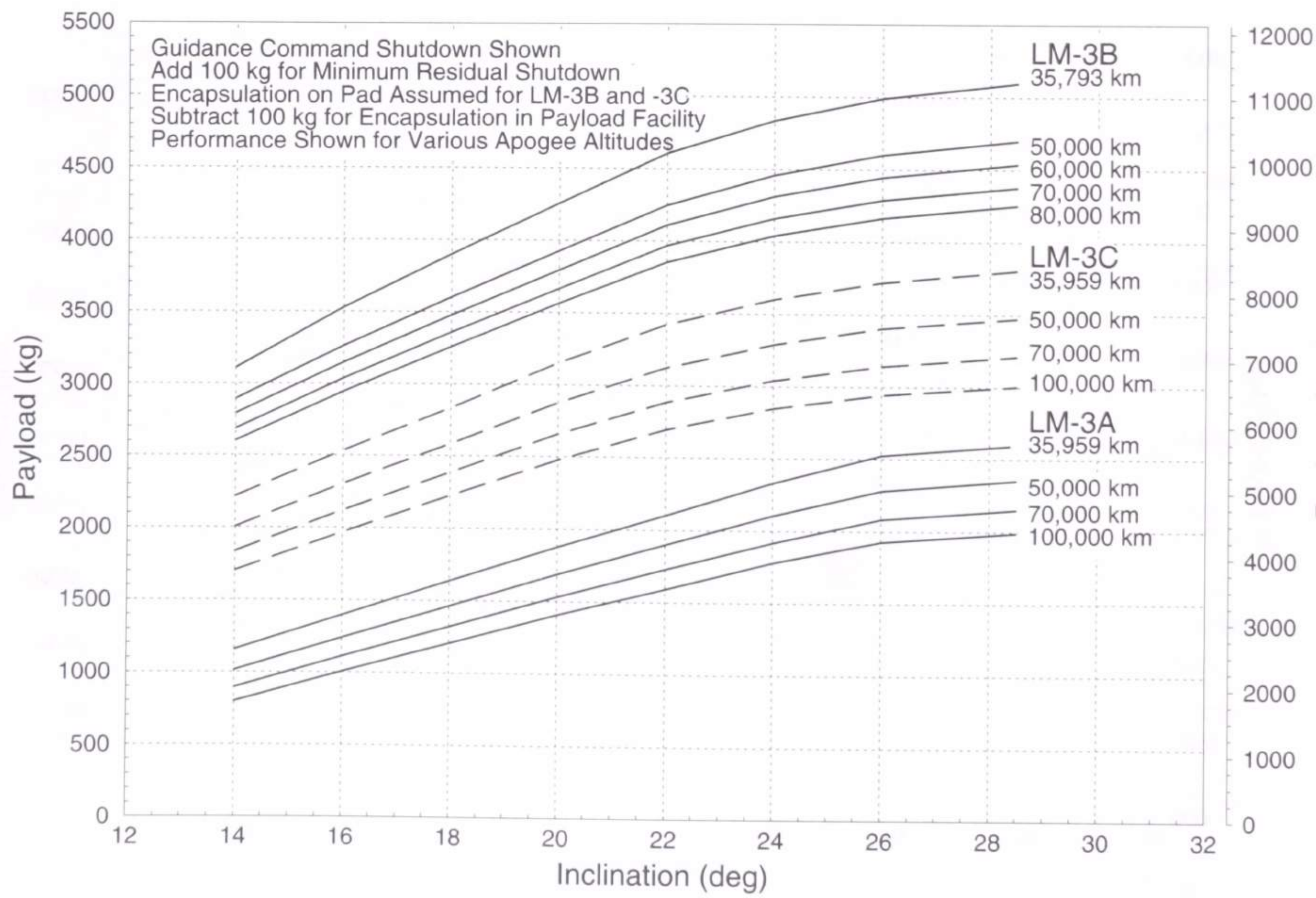


LM-2E/ETS Performance in LEO

Performance

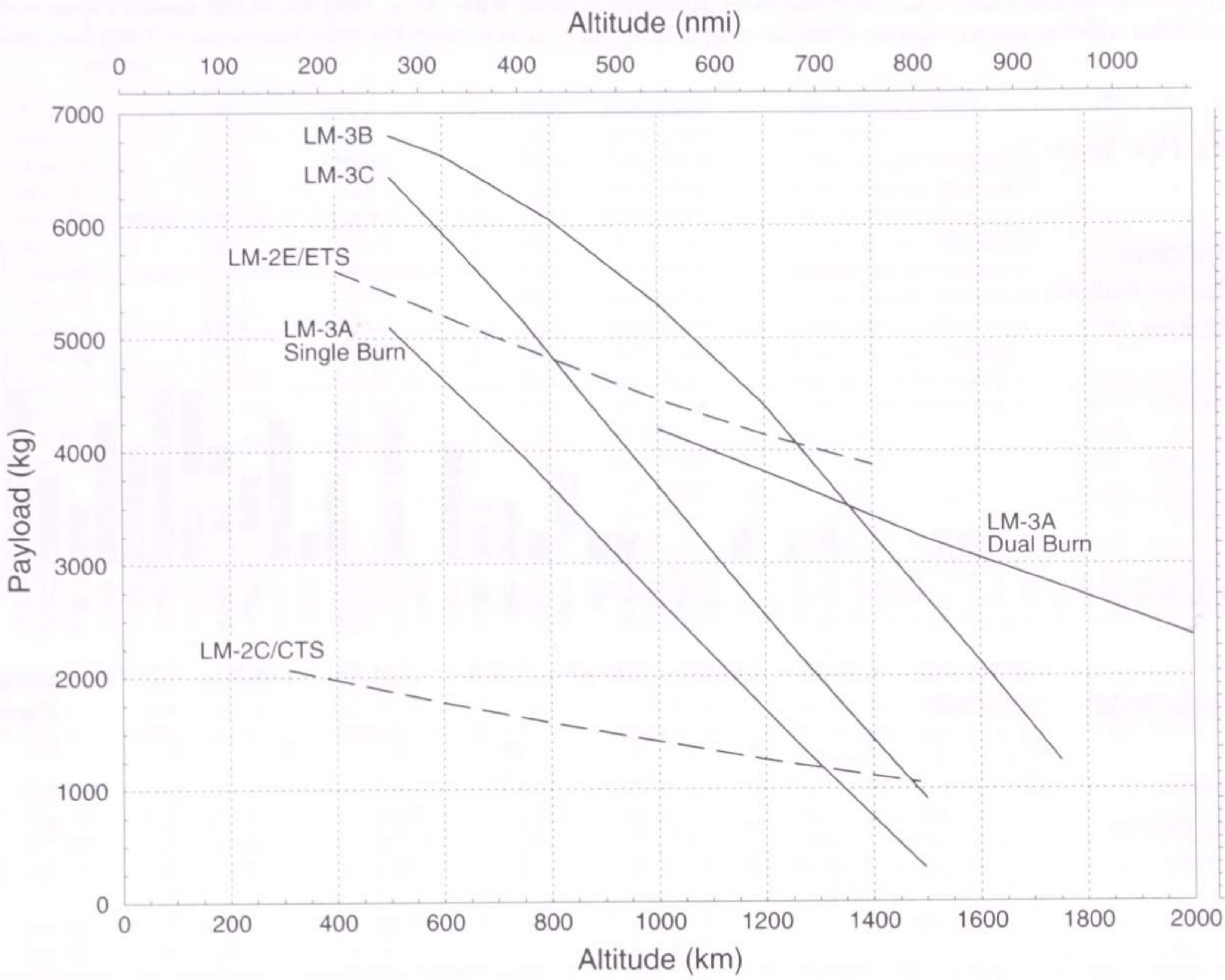


LM-2E GTO Capability With EPKM

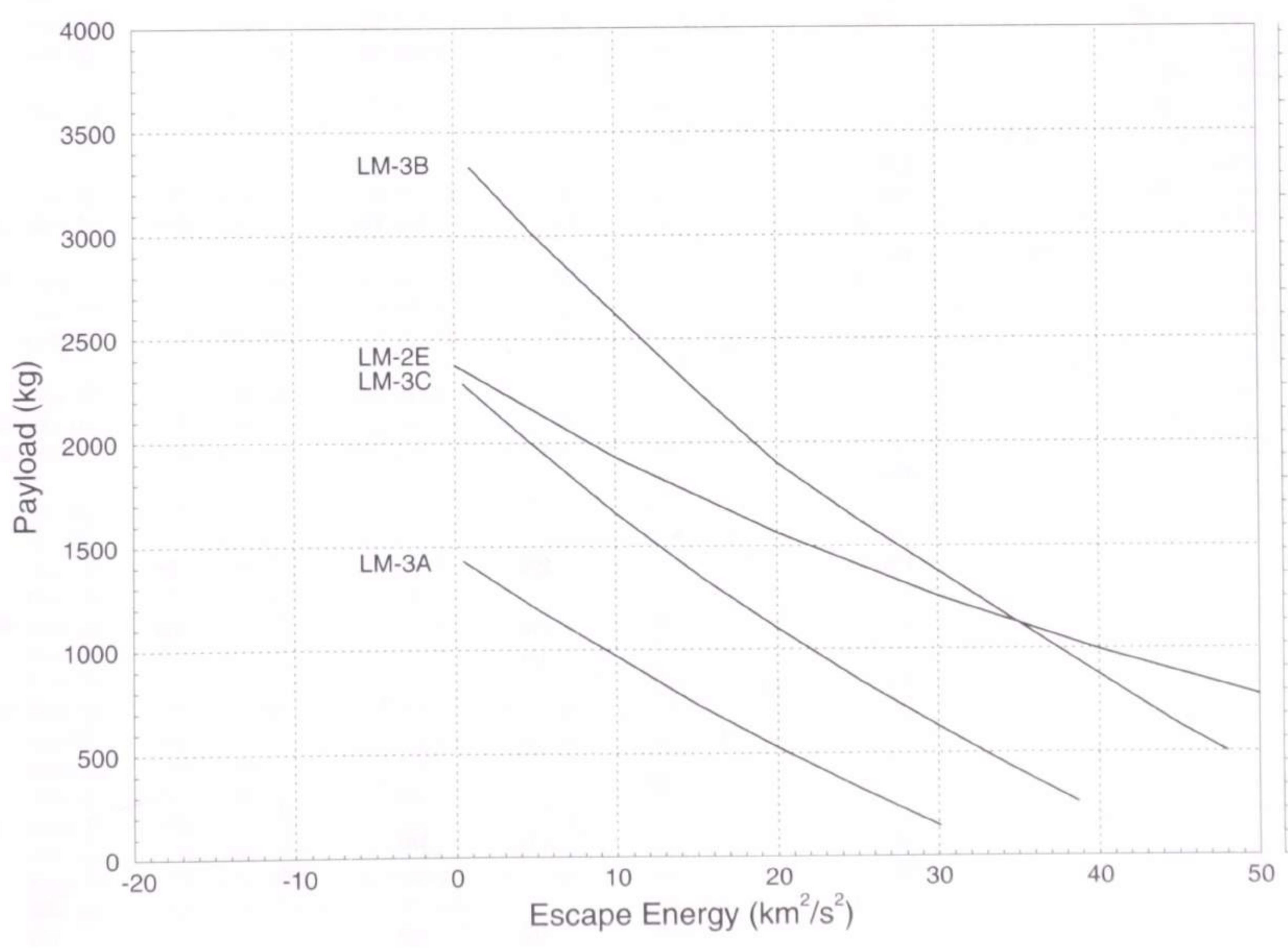


LM-3 Series Performance for Various GTO Orbits

PERFORMANCE



Long March Capability for Sun-Synchronous Orbits

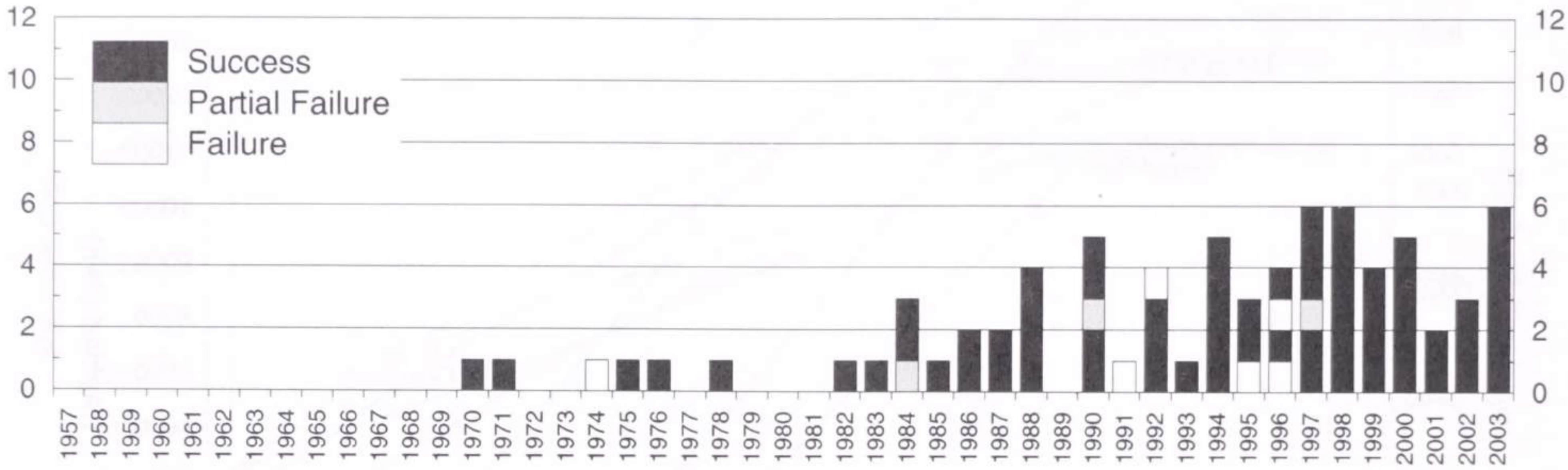


Long March Family Capabilities for Earth Escape Missions

Flight History

In addition to the Long March series, China developed the related Feng Bao launch vehicles in the 1970s. Feng Bao launch dates are listed in the table below for reference, but these launches are not included in the bar chart or Long March launch totals. For more information on Feng Bao, please see the history section.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	**LM-2C and LM-2C/SD**	**LM-2E**	**LM-2D**	**LM-2F**	**LM-3**	**LM-3A**	**LM-3B**	**LM-4B**	**Long March Family Total**
Total Orbital Flights	22	7	4	5	13	8	5	6	75
Launch Vehicle Successes	22	4	4	5	10	8	3	6	66
Launch Vehicle Partial Failures	0	1	0	0	1	0	1	0	3
Launch Vehicle Failures	0	2	0	0	2	0	1	0	6

	Total	Failures or Partial Failures	LM 1 T/F	LM 2A T/F	LM 2C T/F	LM 2D T/F	LM 2E T/F	LM 3 T/F	LM 3A T/F	LM 3B T/F	LM 4 T/F	LM 4B T/F	LM 2F T/F
Total	75	9	2/0	1/1	22/0	4/0	7/3	13/3	8/0	5/2	2/0	6/0	5/0
1970	1	0	1/0										
1971	1	0	1/0										
1972	0	0											
1973	0	0											
1974	1	1		1/1									
1975	1	0			1/0								
1976	1	0			1/0								
1977	0	0											
1978	1	0			1/0								
1979	0	0											
1980	0	0											
1981	0	0											
1982	1	0			1/0								
1983	1	0			1/0								
1984	3	1			1/0			2/1					
1985	1	0			1/0								
1986	2	0			1/0			1/0					
1987	2	0			2/0								
1988	5	0			1/0			2/0			1/0		
1989	0	0											
1990	5	1			1/0		1/1	2/0			1/0		
1991	1	1						1/1					
1992	4	1			1/0	1/0	2/1						
1993	1	0			1/0								
1994	5	0				1/0	1/0	1/0	2/0				
1995	3	1					3/1						
1996	4	2				1/0		2/1		1/1			
1997	6	1			2/0			1/0	1/0	2/1			
1998	6	0			4/0					2/0			
1999	4	0			1/0							2/0	1/0
2000	5	0						1/0	3/0			1/0	1/0
2001	1	0											
2002	4	0										2/0	2/0
2003	6	0			1/0	1/0			2/0			1/0	1/0

T = Total Launch Attempts; F = Failures and Partial Failures

FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1	1970 Apr 24	—	LM 1		J	NLC	1970 034A		Dong Fang Hong 1	173	LEO (68.4)	CIV	China
	2	1971 Mar 03	313	LM 1		J	NLC	1971 018A		Shi Jian 1	221	LEO (69.9)	CIV	China
F	—	**1973 Sep 18**		**FB 1**	**FB1-2**	**J**	NLC	1973 F07A		Ji Shu Shiyan Weixing 1			MIL	China
F	—	**1974 Jul 14**		**FB 1**	**FB1-3**	**J**	NLC	1974 F05A		Ji Shu Shiyan Weixing 2			MIL	China
F	**3**	**Nov 05**	**1343**	**LM 2A**	**CZ 2-1**	**J**	NLC	1974 F07A		Fanhui Shi Weixing			CIV	China
	—	1975 Jul 26		FB 1	FB1-4	J	NLC	1975 070A		Ji Shu Shiyan Weixing 3	1107	LEO (69)	MIL	China
	4	Nov 26	386	LM 2C	CZ 2C-1	J	NLC	1975 111A		Fanhui Shi Weixing-0 1	2500	LEO (62.9)	CIV	China
	—	Dec 16		FB 1	FB1-5	J	NLC	1975 119A		Ji Shu Shiyan Weixing 4	1109	LEO (68.9)	MIL	China
	—	1976 Aug 30		FB 1	FB1-6	J	NLC	1976 087A		Ji Shu Shiyan Weixing 5	1108	LEO (69)	MIL	China
F	—	**Nov 10**		**FB 1**	**FB1-7**	**J**	NLC	1976 F03A		Ji Shu Shiyan Weixing 6			MIL	China
	5	Dec 07	377	LM 2C	CZ 2C-2	J	NLC	1976 117A		Fanhui Shi Weixing-0 2	1812	LEO (59.4)	CIV	China
	6	1978 Jan 26	415	LM 2C	CZ 2C-3	J	NLC	1978 011A		Fanhui Shi Weixing-0 3	1810	LEO (57)	CIV	China
F	—	**1979 Jul 28**		**FB 1**	**FB-10**	**J**	NLC	1979 F02A		Shi Jian	770		CIV	China
	—	1981 Sep 19		FB 1	FB-11	J	NLC	1981 093A	M	Shi Jian 2	257	LEO (59.4)	CIV	China
								1981 093B	M	Shi Jian 2A	28	LEO (59.4)	MIL	China
								1981 093C	M	Shi Jian 2B	483	LEO (59.5)	MIL	China
	7	1982 Sep 09	1687	LM 2C	CZ 2C-4	J	NLC	1982 090A		Fanhui Shi Weixing-0 4	1783	LEO (62.9)	CIV	China
	8	1983 Aug 19	344	LM 2C	CZ 2C-5	J	NLC	1983 086A		Fanhui Shi Weixing-0 5	1842	LEO (63.3)	CIV	China
P	**9**	**1984 Jan 29**	**163**	**LM 3**	**CZ 3-1**	**X**	LC 1	1984 008A		Shiyan Weixing 1	900	GTO	CIV	China
	10	Apr 08	70	LM 3	CZ 3-2	X	LC 1	1984 035A		Shiyong Tongbu Tongxin Weixing X	915	GTO	CIV	China
	11	Sep 12	157	LM 2C	CZ 2C-6	J	NLC	1984 098A		Fanhui Shi Weixing-0 6	1809	LEO (67.9)	CIV	China
	12	1985 Oct 21	404	LM 2C	CZ 2C-7	J	NLC	1985 096A		Fanhui Shi Weixing-0 7	1820	LEO (62.9)	CIV	China
	13	1986 Feb 01	103	LM 3	CZ 3-3	X	LC 1	1986 010A		Shiyong Tongbu Tongxin Weixing 1 (Chinasat 1)	915	GTO	CIV	China
	14	Oct 06	247	LM 2C	CZ 2C-8	J	NLC	1986 076A		Fanhui Shi Weixing-0 8	1820	LEO (56.9)	CIV	China
	15	1987 Aug 05	303	LM 2C	CZ 2C-9	J	NLC	1987 067A		Fanhui Shi Weixing-0 9	1820	LEO (63)	CIV	China
	16	Sep 09	35	LM 2C	CZ 2C-10	J	NLC	1987 075A		Fanhui Shi Weixing-1 1	2100	LEO (63)	CIV	China
	17	1988 Mar 07	180	LM 3	CZ 3-4	X	LC 1	1988 014A		Shiyong Tongbu Tongxin Weixing 2 (Chinasat 2)	1024	GTO	CIV	China
	18	Aug 05	151	LM 2C	CZ 2C-11	J	NLC	1988 067A		Fanhui Shi Weixing-1 2	1850	LEO (63)	CIV	China
	19	Sep 06	32	LM 4	CZ 4-1	T		1988 080A		Feng Yun 1-1	750	SSO	CIV	China
	20	Dec 22	107	LM 3	CZ 3-5	X	LC 1	1988 111A		Shiyong Tongbu Tongxin Weixing 3 (Chinasat 3)	1024	GTO	CIV	China
	21	1990 Feb 04	409	LM 3	CZ 3-6	X	LC 1	1990 011A		Shiyong Tongbu Tongxin Weixing 4 (Chinasat 4)	1024	GTO	CIV	China
	22	Apr 07	62	LM 3	CZ 3-7	X	LC 1	1990 030A		AsiaSat 1 (Westar 6)	1442	GTO	CML	China
P	**23**	**Jul 16**	**100**	**LM 2E**	**CZ 2E-1**	**X**	**LC 2**	none		HS 601 Mass demonstrator		LEO (28.5)	CML	China
								1990 059A	A	Badr A	52	LEO (28.4)	CIV	Pakistan
	24	Sep 03	49	LM 4	CZ 4-2	T		1990 081A		Feng Yun 1-2	881	SSO	CIV	China
								1990 081B	A	Da Qi 1	4	SSO	CIV	China
								1990 081C	A	Da Qi 2	4	SSO	CIV	China
	25	Oct 05	32	LM 2C	CZ 2C-12	J	NLC	1990 089A		Fanhui Shi Weixing-1 3	2080	LEO (56.9)	CIV	China
F	**26**	**1991 Dec 28**	**449**	**LM 3**	**CZ 3-8**	**X**	**LC 1**	1991 088A		Shiyong Tongbu Tongxin Weixing 5 (Chinasat 5)	1025	GTO	CML	China
	27	1992 Aug 09	225	LM 2D	CZ 2D-1	J	NLC	1992 051A		Fanhui Shi Weixing-2 1	2500	LEO (63)	CIV	China
	28	Aug 13	4	LM 2E	CZ 2E-2	X	LC 2	1992 054A		Optus B1 (Aussat B1)	2940	GTO	CML	Australia
	29	Oct 06	54	LM 2C	CZ 2C-13	J	NLC	1992 064A	A	Freja	259	LEO (63)	CIV	Sweden
								1992 064B		Fanhui Shi Weixing-1 4	2100	LEO (63)	CIV	China
F	**30**	**Dec 21**	**76**	**LM 2E**	**CZ 2E-3**	**X**	**LC 2**	1992 090A		Optus B2 (Aussat B2)	2940	GTO	CML	Australia
	31	1993 Oct 08	291	LM 2C	CZ 2C-14	J	NLC	1993 063A		Fanhui Shi Weixing-1 5	2099	LEO (56.9)	CIV	China
	32	1994 Feb 08	123	LM 3A	CZ 3A-1	X	LC 2	1994 010A		Shi Jian 4	410	EEO (28.5)	CIV	China
								1994 010B		KF 1 (DFH 3 mockup)	1600	EEO (28.5)	CIV	China
	33	Jul 03	145	LM 2D	CZ 2D-2	J	NLC	1994 037A		Fanhui Shi Weixing-2 2	2600	LEO (62.9)	CIV	China
	34	Jul 21	18	LM 3	CZ 3-9	X	LC 1	1994 043A		Apstar 1	1383	GTO	CML	China
	35	Aug 27	37	LM 2E	CZ 2E-4	X	LC 2	1994 055A		Optus B3	2940	GTO	CML	Australia
S	36	Nov 29	94	LM 3A	CZ 3A-2	X	LC 2	1994 080A		Dong Fang Hong 3	2230	GTO	CIV	China
F	**37**	**1995 Jan 25**	**57**	**LM 2E**	**CZ 2E-5**	**X**	**LC 2**	1995 F01A		Apstar 2	3000	GTO	CML	China
	38	Nov 28	307	LM 2E	CZ 2E-6	X	LC 2	1995 064A		AsiaSat 2	3485	GTO	CML	China
	39	Dec 28	30	LM 2E	CZ 2E-7	X	LC 2	1995 073A		Echostar 1	3315	GTO	CML	USA
F	**40**	**1996 Feb 14**	**48**	**LM 3B**	**CZ 3B-1**	**X**	**LC 2**	1996 F01A		Intelsat 708	4567	GTO	CML	Int'l
	41	Jul 03	140	LM 3	CZ 3-10	X	LC 1	1996 039A		Apstar 1A	1400	GTO	CML	China
F	**42**	**Aug 18**	**46**	**LM 3**	**CZ 3-11**	**X**	**LC 1**	1996 048A		Zhongxing 7 (Chinasat 7)	1350	GTO	CML	China
	43	Oct 24	67	LM 2D	CZ 2D-3	J	NLC	1996 059A		Fanhui Shi Weixing-2 3	2600	LEO (63)	CIV	China
	44	1997 May 11	199	LM 3A	CZ 3A-3	X	LC 2	1997 021A		Zhongxing 6 (Chinasat 6)	2232	GTO	CML	China
	45	Jun 10	30	LM 3	CZ 3-12	X	LC 1	1997 029A		Feng Yun 2A	1380	GTO	CIV	China
P	**46**	**Aug 19**	**70**	**LM 3B**	**CZ 3B-2**	**X**	**LC 2**	1997 042A		Agila 2	3770	GTO +	CML	Phillipines
	47	Sep 01	13	LM 2C/SD	CZ 2C-15	T		1997 048A	M	Iridium MFS 1	690	LEO (87)	CML	USA
								1997 048B	M	Iridium MFS 2	690	LEO (87)	CML	USA
	48	Oct 16	45	LM 3B	CZ 3B-3	X	LC 2	1997 062A		Apstar 2R	3747	GTO +	CML	China
	49	Dec 08	53	LM 2C/SD	CZ 2C-16	T		1997 077A	M	Iridium 42	690	LEO (87)	CML	USA
								1997 077B	M	Iridium 44	690	LEO (87)	CML	USA

T = Taiyuan Satellite Launch Center; X = Xichang Satellite Launch Center; J = Jiuquan Satellite Launch Center

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Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

FLIGHT HISTORY

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	50	1998	Mar 25	107	LM 2C/SD	CZ 2C-17	T		1998 018A	M	Iridium 51	690	LEO (87)	CML	USA
									1998 018B	M	Iridium 61	690	LEO (87)	CML	USA
	51		May 02	38	LM 2C/SD	CZ 2C-18	T		1998 026A	M	Iridium 69	690	LEO (87)	CML	USA
									1998 026B	M	Iridium 71	690	LEO (87)	CML	USA
	52		May 30	28	LM 3B	CZ 3B-4	X	LC 2	1998 033A		Zhongwei 1 (Chinastar 1)	3000	GTO	CML	China
	53		Jul 18	49	LM 3B	CZ 3B-5	X	LC 2	1998 044A		Sinosat 1	3492	GTO +	CML	China
	54		Aug 19	32	LM 2C/SD	CZ 2C-19	T		1998 048A	M	Iridium 3	690	LEO (87)	CML	USA
									1998 048B	M	Iridium 76	690	LEO (87)	CML	USA
	55		Dec 19	122	LM 2C/SD	CZ 2C-20	T		1998 074A	M	Iridium 88	690	LEO (87)	CML	USA
									1998 074B	M	Iridium 89	690	LEO (87)	CML	USA
	56	1999	May 10	142	LM 4B	CZ 4-3	T		1999 025A		Feng Yun 1-3		SSO	CIV	China
									1999 025B	A	Shi Jian 5		SSO	CIV	China
	57		Jun 11	32	LM 2C/SD	CZ 2C-21	T		1999 033A	M	Iridium 14A	690	LEO (87)	CML	USA
									1999 033B	M	Iridium 21A	690	LEO (87)	CML	USA
	58		Oct 14	125	LM 4B	CZ4-4	T		1999 057A		Zi Yuan 1-1 (CBERS 1)	1450	SSO	CIV	Brazil
S									1999 057B	A	SACI 1	60	SSO	CIV	Brazil
	59		Nov 19	36	LM 2F	CZ 2F-1	J	SLC	1999 061A		Shenzhou 1		LEO (42.6)	CIV	China
	60	2000	Jan 25	67	LM 3A	CZ 3A-4	X	LC 2	2000 003A		Zhongxing 22	2300	GTO	CML	China
	61		Jun 25	152	LM 3	CZ 3-13	X	LC 1	2000 032A		Feng Yun 2B	1400	GTO	CIV	China
	62		Sep 01	68	LM 4B	CZ 4-5	T		2000 050A		Zi Yuan 2-1		SSO	CIV	China
	63		Oct 30	59	LM 3A	CZ 3A-5	X	LC 1	2000 069A		Beidou 1A	2200	GTO	CIV	China
	64		Dec 20	51	LM 3A	CZ 3A-6	X	LC 1	2000 082A		Beidou 2	2200	GTO	CIV	China
	65	2001	Jan 09	20	LM 2F	CZ 2F-2	J	SLC	2001 001A		Shenzhou 2	7800	LEO (42.6)	CIV	China
	66	2002	Mar 25	440	LM 2F	CZ 2F-3	J	SLC	2002 014A		Shenzhou 3	7800	LEO (42.4)	CIV	China
	67		May 15	51	LM 4B	CZ 4-6	T		2002 024B	C	Feng Yun 1-4	958	SSO	CIV	China
									2002 024A	C	Haiyang 1	360	SSO	MIL	China
	68		Oct 27	165	LM 4B	CZ 4-7	T		2002 049A		Zi Yuan 2-2		LEO (97.4)	CIV	China
	69		Dec 29	63	LM 2F	CZ 2F-4	J	SLC	2002 061A		Shenzhou 4	7600	LEO (42.4)	CIV	China
	70	2003	May 24	146	LM 3A	CZ 3A-7	X	LC 3	2003 021A		Beidou 1C	2200	GTO	MIL	China
	71		Oct 15	144	LM 2F	CZ 2F-5	J	LC 5	2003 045A		Shenzhou 5	7790	LEO (42.4)	CIV	China
	72		Oct 21	6	LM 4B	CZ 4B-6	T		2003 049A		Zi Yuan 1-2 (CBERS 2)	1550	SSO	CIV	Brazil/China
									2003 049B	A	Chuang Xin 1	100	SSO	CIV	China
	73		Nov 03	13	LM 2D	CZ 2D-4	J	LC 4	2003 051A		Fanhui Shi Weixing-3 1	3000	LEO (62)	CIV	China
	74		Nov 14	11	LM 3A	CZ 3A-8	X	LC 3	2003 052A		Zhongxing 20 (Chinastar 20)	2300	GTO	CML	China
	75		Dec 29	45	LM 2C/SM	CZ 2C-22	X	LC 3	2003 061A		Tan Ce 1	350	EEO (28.5)	CIV	China

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Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

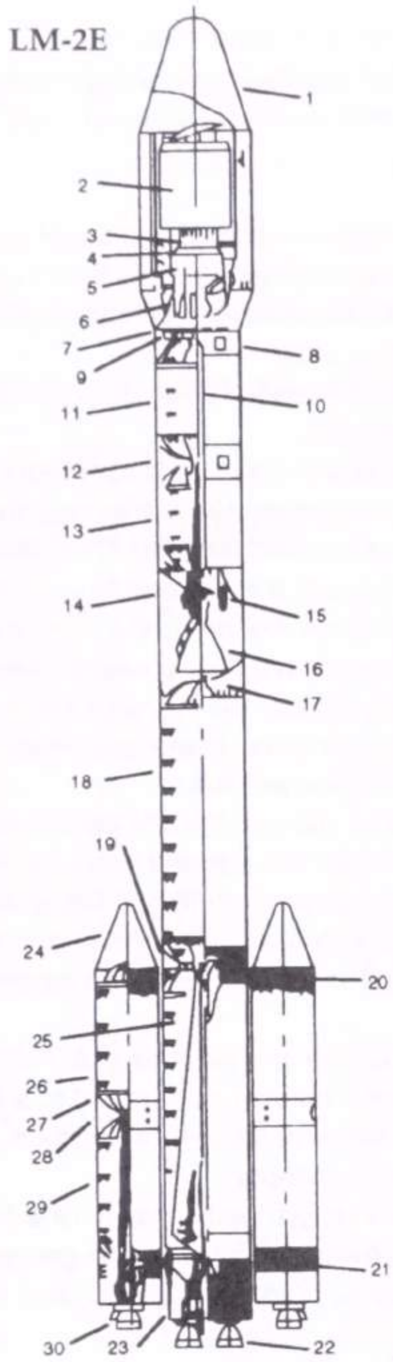
Flight History

Failure Descriptions:

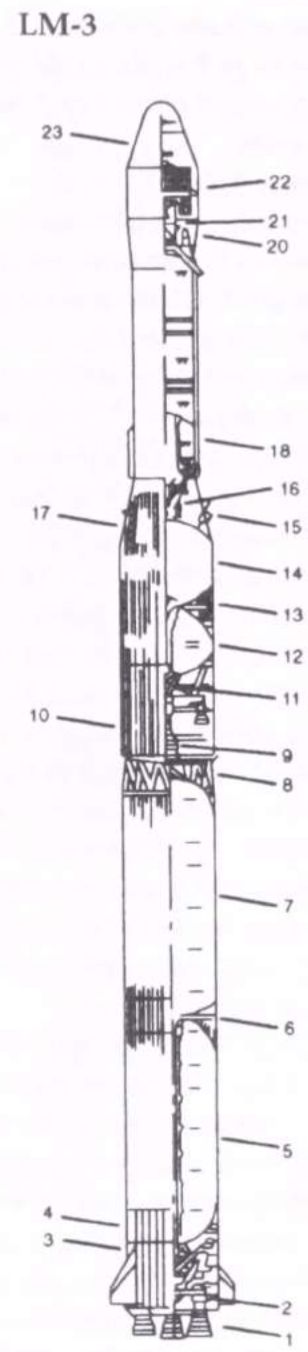
F	1973 Sep 18	FB1-2	1973 F07	Failure to orbit.
F	1974 Jul 14	FB1-3	1974 F05	Vehicle lost attitude stability and was destroyed by onboard destruct system.
F	1974 Nov 05	CZ 2-1	1974 F07	A broken wire in the pitch rate gyro circuit caused pitch oscillations starting shortly after launch resulting in loss of control and destruction by the onboard destruct system at T+20 s.
F	1976 Nov 10	FB1-7	1976 F03	Failure to orbit.
F	1979 Jul 28	FB1-10	1979 F02	Second-stage failure.
P	1984 Jan 29	CZ 3-1	1984 008	Third stage failed 4 s after restart for GTO insertion of the satellite, because of incorrect mixture ratio in the engine gas generator, which caused high temperatures and burned out the turbine shell. However, many planned tests on the experimental communications spacecraft were still carried out in the resulting elliptical orbit.
P	1990 Jul 16	CZ 2E-1	1990 059	Second stage provided incorrect orientation for payload release. Mass demonstrator payload failed to achieve GTO; Badr A did successfully deploy.
F	1991 Dec 28	CZ 3-8	1991 088	The third-stage engine suffered a loss of turbine speed and combustion pressure 58 s after reigniting for the second burn for GTO insertion and shut down completely 135 s after reignition. Loss of pressure in the high-pressure helium supply used for engine control had reduced the propellant flow.
F	1992 Dec 21	CZ 2E-3	1992 090	At T+48 s at an altitude of 7000 m an explosion occurred in the payload fairing, destroying both the payload and fairing. The payload was scattered downrange, but the rest of the launch vehicle went on to achieve orbit with the intended accuracy. CGWIC concluded the spacecraft had failed and that the launch vehicle was not at fault. However, it is widely believed that the fairing probably collapsed because of transonic buffeting during this period of high dynamic pressure, causing destruction of the spacecraft.
S	1994 Nov 29	Dong Fang Hong 3	1994 080A	Satellite problems resulted in insufficient propellant to enter service.
F	1995 Jan 25	CZ 2E-5	1995 F01	At T+51 s an explosion destroyed the forward portion of the vehicle. Six people were killed by falling debris. Hughes concluded that the fairing longitudinal split line opened because of high aerodynamic loads from buffeting and wind shear. As the fairing collapsed, it damaged the spacecraft propellant tanks, causing the fire that destroyed the spacecraft and forward end of the launch vehicle. CGWIC disagreed, concluding that a failure of the interface between the rocket and satellite caused the explosion of the spacecraft.
F	1996 Feb 14	CZ 3B-1	1996 F01	The vehicle pitched over immediately after liftoff, impacting and exploding at T+22 s near a village close to the launch site. At least six people were killed. The fault was traced to a lack of output from the power module for the servo-loop in the follow-up frame of the inertial platform. This caused a faulty inertial reference, which made the launch vehicle steer incorrectly.
F	1996 Aug 18	CZ 3-11	1996 048	The third-stage engine shut down roughly 40 s earlier than planned because of a fire in the LH_2 injector of the gas generator. Insufficient purging had permitted oxygen to freeze in the gas generator during flight.
P	1997 Aug 19	CZ 3B-2	1997 042	Spacecraft had to use onboard propellant to reach correct orbit because of poor injection accuracy on the part of the launch vehicle.
S	1999 Oct 14	SACI-1	1999 057B	Brazil lost contact with the satellite shortly after launch.

Vehicle Design

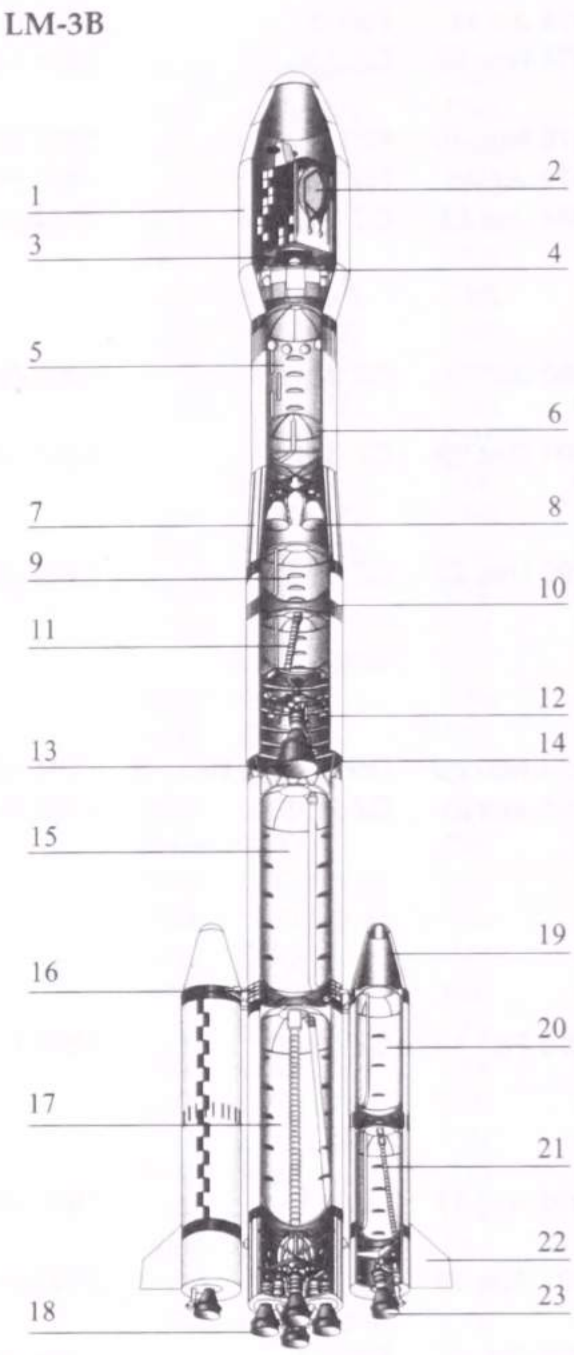
Overall Vehicle



1. Fairing
2. Spacecraft
3. Spacecraft attach fitting
4. Payload clampband & sep. sys.
5. PKM
6. Payload adapter (4 pieces)
7. Fairing clampband
8. Vehicle equipment bay
9. Equipments
10. Electrical external ducting
11. Second stage oxidizer tank
12. Second stage intertank section
13. Second stage fuel tank
14. Interstage section
15. Second stage vernier engine
16. Second stage main engine
17. Exhaust window
18. First stage oxidizer tank
19. First stage intertank section
20. Forward mounting system
21. Rear mounting system
22. First stage engine
23. Tail section
24. Conical head
25. First stage fuel tank
26. Liquid boosters oxidizer tank
27. Separation rockets (sixteen)
28. Intertank skirt
29. Liquid boosters fuel tank
30. Liquid boosters engine



1. First stage engine
2. First stage servomechanism
3. First stage tail section
4. Aft skirt
5. Fuel tank
6. Intertank section
7. Oxidizer tank
8. Interstage truss
9. Second stage engine
10. Interstage section
11. Second stage servomechanism
12. Fuel tank
13. Interstage section
14. Oxidizer tank
15. Retro-rockets
16. Third stage engine
17. Interstage section
18. LOX tank
19. LH tank
20. Vehicle equipment bay
21. Payload / CZ-3 adapter
22. Payload
23. Fairing (Model A)



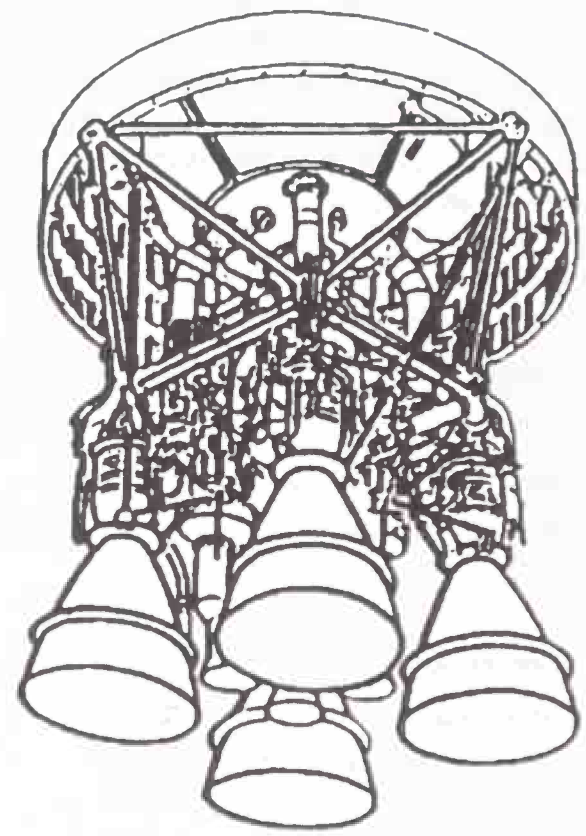
1. Fairing
2. SC
3. Payload Adapter
4. Vehicle Equipment Bay
5. LH_2 Tank
6. LOX Tank
7. Inter-stage Section
8. Third Stage Engine
9. Second Stage Oxidizer Tank
10. Inter-tank Section
11. Second Stage Fuel Tank
12. Second Stage Vernier Engine
13. Second Stage Main Engine
14. Inter-stage Section
15. First Stage Oxidizer Tank
16. Inter-tank Section
17. First Stage Fuel Tank
18. First Stage Engine
19. Booster's Nose
20. Booster's Oxidizer tank
21. Booster's Fuel Tank
22. Stabilizer
23. Booster's Engine

	LM 2C, 2C/CTS	LM-2E	LM-2F	LM-3	LM-3A	LM-3B, 3C	LM-4B
Height	42 m (138 ft)	49.7 m (163 ft)	58.3 m (191.3 ft)	44.6 m (146 ft)	52.5 m (172 ft)	54.8 m (180 ft)	45.8 m (150 ft)
Gross Liftoff Mass	233 t (514 klbm)	460 t (1015 klbm)	479.7 t (1057 klbm)	204 t (450 klbm)	241 t (531 klbm)	3B: 425.8 t (939 klbm) 3C: 345 t (761 klbm)	249.2 t (549 klbm)
Thrust at Liftoff	2962 kN (665.8 klbf)	5923 kN (1332 klbf)	5923 kN (1332 klbf)	2962 kN (665.8 klbf)	2962 kN (665.8 klbf)	3B: 5923 kN (1332 klbf) 3C: 4443 kN (999 klbf)	2962 kN (665.8 klbf)

Vehicle Design

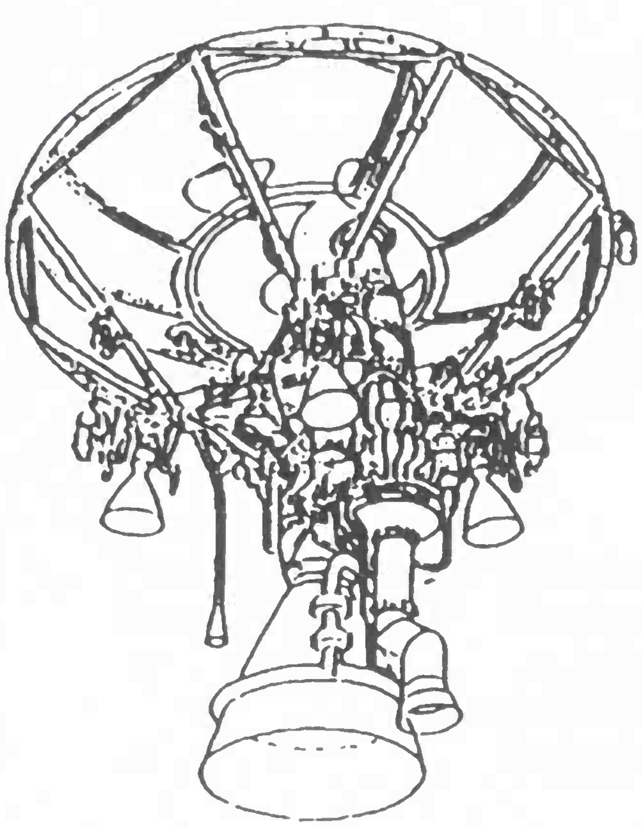
Stages

Most of the stages of different Long March vehicles share a few basic features resulting from their common ballistic missile heritage. The first Long March 1 was derived from the DF-4 (CSS-3) ballistic missile and had a diameter of 2.25 m (7.4 ft). This structural design was carried over to the strap-on boosters used on the LM-2E and LM-3B, and the basic tooling was used to create tankage of the same diameter for the first Chinese cryogenic stage on the LM-3. The Long March 2 and subsequent vehicles are derived from the DF-5 (CSS-4) ICBM. The DF-5 had two stages of 3.35-m (11-ft) diameter with storable N_2O_4/UDMH propellants. As a result, the first two stages of all currently operational Long March versions still have this diameter, the same propellants, and closely related propulsion systems. The stage lengths are varied and different upper stages are used on the different Long March types. The propellant tanks are independent cylindrical structures joined by an intertank structure. The oxidizer tank is above the fuel tank, and oxidizer is fed to the engines by a propellant line running through the middle of the lower tank. The engines are contained in the tail section.



First Stage YF-20 Engine

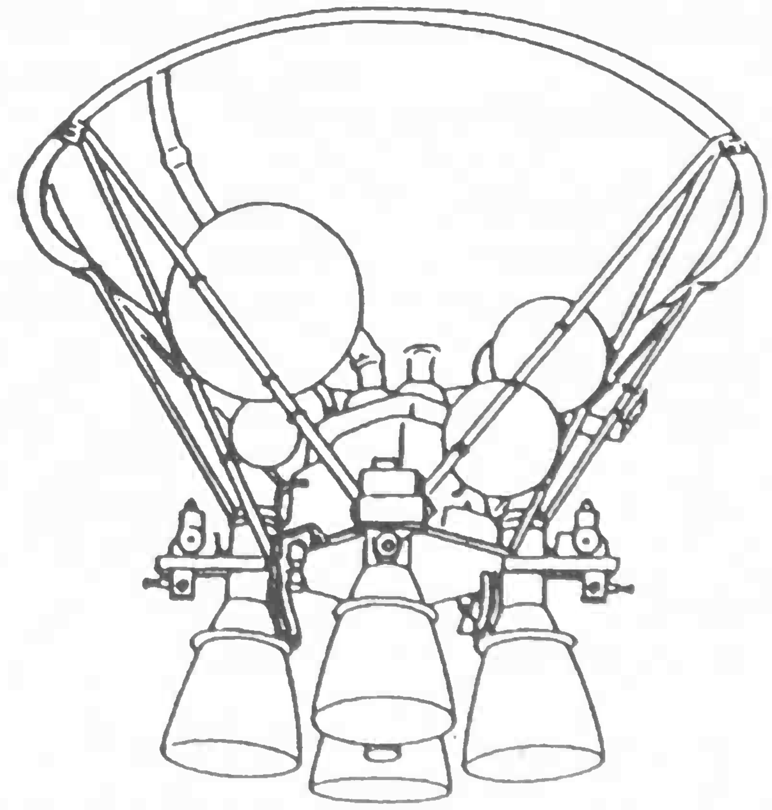
The first two stages of Long March vehicles are all powered by variations of the YF-20 engine. YF stands for yei-ti fa-dong-ji (liquid engine). The YF-20 is driven by a single-shaft gas generator turbopump burning N_2O_4 and UDMH. Different variations and clusters of the YF-20 have different names. A single engine, designated the DaFY5-1, powers the strap-on boosters used on LM-2E, 2F, 3B and 3C vehicles. A cluster of four YF-20B engines powers the first stage of all Long March vehicles. In some vehicles this combined first-stage propulsion system is called the YF-21B or DaYF-21. In other vehicles it is given the designation DaFY6-2. The differences, if any, between these engines are unknown but are assumed to be minor. The second stage uses a single engine plus four verniers. In some vehicles the core engine is called the YF-22, the verniers are the YF-23, and the combined system is the YF-24. In other vehicles the main engine is designated the DaFY-20-1, with the vernier designated DaFY-21-1.



YF-22/YF-23 Engine

The cryogenic upper stage used on the LM-3 series is of a different design from other Long March stages and has a smaller tank diameter. The original LM-3 third stage was only 2.25 m (7. 4 ft) in diameter, while the LM-3A, B, and C stage is 3.0 m (9.9 ft) in diameter. The tanks share a common bulkhead, and the oxygen tank is on the bottom. The original LM-3 uses the four chamber YF-73 engine. The YF-73 has a restart capability, using two tanks of pressurized nitrogen to spin up the turbopumps. A heat exchanger is built into the engine to warm oxygen and hydrogen gases for pressurization of the propellant tanks. The LM-3A, B, and C use a pair of single-chamber YF-75 engines, which are derived from the YF-73. The two engines gimbal independently for attitude control.

Hot separation is used for the separation between the first and second stages. Just before separation, the second stage engine ignites. When the thrust of the first-stage engine reduces to a lower level after the engine shutoff, the explosive bolts connecting the first and second stages release the stages. The first stage is pushed away by the second-stage engine jet flow acting on the upper surface of the first stage forward tank. The interstage is constructed either as a truss structure, or as a cylinder with exhaust vents on the lower half. Both designs allow exhaust gas to escape, preventing high pressure from building up in the interstage during separation.



YF-73 Engine

Vehicle Design

LM-2C

The LM-2C is a two-stage launch vehicle closely based on the DF-5 (CSS-4) ICBM. In the mid 1990s, both stages were stretched to increase performance, but the vehicle did not receive a new designation. An optional upper stage was developed at the same time to reach circular orbits above 500 km (270 nmi), initially in support of seven Iridum deployment missions. Originally called the Smart Dispenser for Iridium, the generic version is called the CTS (LM-2C Top Stage). It consists of a single, small solid motor embedded in a disk structure whose top surface supports the satellites and stage electronics. An ACS system provides three-axis control.

	Stage 1	Stage 2	CTS
Dimensions			
Length	25.7 m (84.3 ft)	7.7 m (25.3 ft)	1.5 m (4.9 ft)
Diameter	3.35 m (11 ft)	3.35 m (11 ft)	2.7 m (8.9 ft)
Mass			
Propellant Mass	162.7 t (358.7 klbm)	54,667 kg (120.5 klbm)	125 kg solid + 50 kg hydrazine (275 + 110 lbm)
Inert Mass	?	?	?
Gross Mass	?	?	?
Propellant Mass Fraction	?	?	?
Structure			
Type	Skin-stringer	Skin-stringer	?
Material	Aluminum	Aluminum	?
Propulsion			
Engine Designation	DaFY6-2	DaFY20-1 main engine with DaFY21-1 vernier	FG-47/SpaB-54
Number of Engines	1 system, comprising 4 engines	1 main engine + 1 vernier with 4 thrust chambers	1
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	HTPB
Average Thrust	Sea level: 2961.6 kN (665.7 klbf)	Main engine: 742 kN + Vernier: 4×11.8 kN (167 klbf + 4×2.65 klbf)	10.78 kN (2425 klbf)
Isp	Sea level: 260.7 s	Main engine: 298.0 s Vernier: 296.8 s	286 s
Chamber Pressure	?	?	?
Nozzle Expansion Ratio	?	?	?
Propellant Feed System	Gas-generator turbopump	Main engine: gas-generator turbopump Vernier: gas-generator turbopump	—
Mixture Ratio (O/F)	2.1:1	2.1:1 main engine 1.57 verniers	—
Throttling Capability	100% only	100% only	—
Restart Capability	None	None	None
Tank Pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	—
Attitude Control			
Pitch, Yaw	Hydraulic gimbaling ±10 deg in single plane	Hydraulic gimbaling of vernier nozzles ±60 deg	Hydrazine ACS
Roll	Hydraulic gimbaling ±10 deg in single plane	Hydraulic gimbaling of vernier nozzles ±60 deg	ACS
Staging			
Nominal Burn Time	122 s	Main engine: 109 s Verniers: 347 s	35 s
Shutdown Process	Burn to depletion	Command shutdown	Burn to depletion
Stage Separation	Explosive bolts, fire-in-the-hole hot separation	Explosive bolts, retro-rockets	

Vehicle Design

LM-2E and LM-2F

The LM-2E uses two core stages plus four strap-on boosters to carry heavy payloads to low LEO orbits. Two upper stages are available. The EPKM (E Perigee Kick Motor) is a spin-stabilized solid motor that has been used to deliver medium satellites to GTO. The ETS (E Top Stage) is a larger version of the LM-2C Smart Dispenser CTS stage. It is a squat disc with a solid motor to circularize higher LEO orbits plus ACS and control system. The ETS stage has not been used to date. The LM-2F is a more recent variation of the LM-2E. Its design is quite similar, with improvements in systems reliability and redundancy to carry humans on Shenzhou spacecraft. One addition to the LM-2F is the emergency escape tower. Similar to systems on Soyuz or Apollo, the Shenzhou emergency escape system is a solid-propellant rocket that can pull the crew module away from a failed vehicle up to 160 s after launch (when the fairing and tower separate). The first functional escape system was installed on the SZ-3 flight.

	Strap-on Boosters	Stage 1	Stage 2	EPKM (Optional Stage 3)
Dimensions				
Length	15.33 m (50.3 ft)	23.7 m (77.8 ft)	15.5 m (50.9 ft)	2.79 m (9.15 ft)
Diameter	2.25 m (7.4 ft)	3.35 m (11.0 ft)	3.35 m (11.0 ft)	1.7 m (5.576 ft)
Mass				
Propellant Mass	37.75 t (83.22 klbm) each	187 t (412 klbm)	86 t (190 klbm)	5444 kg (12,002 lbm)
Inert Mass	3.2 t (7.1 klbm) each	9.5 t (21 klbm)	5.5 t (12 klbm)	541kg (1192 lbm)
Gross Mass	41 t (90.4 klbm) each	196.5 t (433 klbm)	91.5 t (202 klbm)	5985 kg (13,194 lbm)
Propellant Mass Fraction	0.92	0.95	0.94	0.91
Structure				
Type	Skin-stringer	Skin-stringer	Skin-stringer	Filament-wound monocoque
Material	Aluminum	Aluminum	Aluminum	Glass fiber
Propulsion				
Engine Designation	DaFY5-1	DaFY6-2	DaFY20-1 main engine, DaFY21-1 vernier	FG-46/SpaB-17
Number of Engines	1 per booster, 4 boosters	One YF-21 system, consisting of four YF-20 engines	One YF-24 system consisting of one YF-22 main engine + one YF 23 vernier with 4 thrust chambers	1
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	HTPB
Average Thrust	Sea level: 740.4 kN (166.5 klbf)	Sea level: 2961.6 kN (665.7 klbf)	Main engine: 742 kN + Vernier: 4×11.8 kN	177 kN (39.8 klbf) (167 + 4×2.7 klbf)
Isp	Sea level: 260.7 s	Sea level: 260.7 s	Main engine: 298.0 s	292 s Vernier: 296.8
Chamber Pressure	?	?	?	?
Nozzle Expansion Ratio	?	?	?	?
Propellant Feed System	Gas-generator turbopump	Gas-generator turbopump	Main engine: gas-generator turbopump Vernier: gas-generator turbopump	—
Mixture Ratio (O/F)	2.1:1	2.1:1	2.1:1 main engine 1.57 verniers	—
Throttling Capability	100% only	100% only	100% only	None
Restart Capability	None	None	None	None
Tank Pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	—
Attitude Control				
Pitch, Yaw	Fixed cant, stage 1 provides control	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Spin-stabilized
Roll	Fixed cant, stage 1 provides control	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Spin-stabilized
Staging				
Nominal Burn Time	128 s	166 s	Main engine: 295 s Verniers: 410 s	87 s
Shutdown Process	Burn to depletion	Burn to depletion	Command shutdown	Burn to depletion
Stage Separation	Pyrotechnic release mechanism plus 4 retro-rockets each	Explosive bolts, fire-in-the-hole hot separation	Explosive bolts, retro-rockets	

Vehicle Design

LM-3

The LM-3 differs from the Long March 2 and 4 series by the addition of China's first cryogenic upper stage. It is powered by a four-chamber YF-73 engine.

	Stage 1	Stage 2	Stage 3
Dimensions			
Length	20.22 m (66.3 ft)	9.71 m (31.8 ft)	7.48 m (24.6 ft)
Diameter	3.35 m (11.0 ft)	3.35 m (11.0 ft)	2.25 m (7.4 ft)
Mass			
Propellant Mass	144 t (317 klbm)	35 t (77 klbm)	8.5 t (18.7 klbm)
Inert Mass	9 t (20 klbm)	4 t (9 klbm)	2 t (4.4 klbm)
Gross Mass	151 t (333 klbm)	39 t (86 klbm)	10.5 t (23.1 klbm)
Propellant Mass Fraction	0.95	0.90	0.81
Structure			
Type	Skin-stringer	Skin-stringer	Skin-stringer?
Material	Aluminum	Aluminum	Aluminum ?
Propulsion			
Engine Designation	YF-21	YF-22 main engine, YF-23 vernier	YF-73
Number of Engines	1 YF-21 system, consisting of 4 YF-20 engines	1 YF-24 system consisting of 1 YF-22 main engine + 1 YF 23 vernier with 4 thrust chambers	1 engine with 4 chambers
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	LOX/LH_2
Average Thrust	Sea level: 2785 kN (626 klbf)	Main engine: 720 kN + Vernier: 4×11.5 kN (162 klbf + 4×2.6 klbf)	44.147 kN (9.92 klbf)
Isp	Sea level: 259 s	Main engine: 289 s Vernier: 281.7 s	425 s
Chamber Pressure	?	?	26.3 bar (382 psi)
Nozzle Expansion Ratio	?	?	40:1
Propellant Feed System	Gas-generator turbopump	Main engine: gas-generator turbopump Vernier: gas-generator turbopump	Gas-generator turbopump
Mixture Ratio (O/F)	2.1:1	2.1:1 main engine 1.57 verniers	5.0:1
Throttling Capability	100% only	100% only	100% only
Restart Capability	None	None	1 restart standard
Tank Pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	Warm oxygen and hydrogen
Attitude Control			
Pitch, Yaw	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Engine gimbal ±24 deg
Roll	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Engine gimbal ±24 deg
Staging			
Nominal Burn Time	132 s	180 s	500 s + 300 s
Shutdown Process	Burn to depletion	Command shutdown	Command shutdown
Stage Separation	Explosive bolts, fire-in-the-hole hot separation	Explosive bolts, retro-rockets	Spring jettison of payload

Vehicle Design

LM-3A, 3B, 3C

The LM-3A, B, and C are related launch vehicles that evolved from the basic LM-3. They differ from the LM-3 in the use of stretched first and second stages and a different cryogenic upper stage. It is wider and is powered by a pair of YF-75 engines, rather than the older YF-73. The LM-3B and 3C have four or two strap-on boosters respectively to increase performance over the basic LM-3A.

	LM-3B, 3C Strap-on Boosters	Stage 1	Stage 2	Stage 3
Dimensions				
Length	15.33 m (50.3 ft)	23.27 m (76.32 ft)	9.94 m (32.6 ft)	12.375 m (40.6 ft)
Diameter	2.25 m (7.4 ft)	3.35 m (11.0 ft)	3.35 m (11.0 ft)	3.0 m (9.84 ft)
Mass				
Propellant Mass	37.75 t (83.22 klbm) each	171.78 t (378.7 klbm)	LM-3A: 29.6 t (65.3 klbm) LM-3B, 3C: 49.60 t (109.4 klbm)	18.19 t (40.1 klbm)
Inert Mass	3.2 t (7.1 klbm) each	?	LM-3A: 4 t (8.8 klbm) LM-3B, 3C: ?	?
Gross Mass	41 t (90.4 klbm) each	Roughly 180 t (397 klbm)	LM-3A: 33.6 t (74.1 klbm) LM-3B, 3C: ?	Roughly 21 t (46.3 klbm)
Propellant Mass Fraction	0.92	0.95	LM-3A: 0.88 LM-3B, 3C: ?	0.86
Structure				
Type	Skin-stringer	Skin-stringer	Skin-stringer	Skin-stringer?
Material	Aluminum	Aluminum	Aluminum	Aluminum ?
Propulsion				
Engine Designation	DaFY5-1	DaFY6-2	DaFY20-1 main engine, DaFY21-1 vernier	FG-46/SpaB-17
Number of Engines	1 per booster LM-3A: 0 boosters LM-3B: 4 boosters LM-3C: 2 boosters	1 YF-21 system, consisting of 4 YF-20 engines	1 YF-24 system consisting of 1 YF-22 main engine + 1 YF 23 vernier with 4 thrust chambers	2
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	LOX/LH_2
Average Thrust	Sea level: 740.4 kN (166.5 klbf)	Sea level: 2961.6 kN (665.6 klbf)	Main engine: 742 kN + Vernier: 4×11.8 kN (167 + 4×2.7 klbf)	2×78.5 kN (17.6 klbf)
Isp	Sea level: 260.7 s	Sea level: 260.7 s	Main engine: 298.0 s Vernier: 296.8	440 s
Chamber Pressure	?	?	?	37.6 bar (546 psi)
Nozzle Expansion Ratio	?	?	?	80:1
Propellant Feed System	Gas-generator turbopump	Gas-generator turbopump	Main engine: gas-generator turbopump Vernier: gas-generator turbopump	Gas-generator turbopump
Mixture Ratio (O/F)	2.1:1	2.1:1	2.1:1 main engine	5:1 1.57 verniers
Throttling Capability	100% only	100% only	100% only	100% only?
Restart Capability	None	None	None	1 restart standard
Tank Pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	Helium bottle and self-pressurization
Attitude Control				
Pitch, Yaw	Fixed cant, stage 1 provides control	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Engine gimbal
Roll	Fixed cant, stage 1 provides control	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Engine gimbal
Staging				
Nominal Burn Time	128 s	145 s	180 s	470 s
Shutdown Process	Burn to depletion	Burn to depletion burn to depletion	Command shutdown	Command shutdown or
Stage Separation	Pyrotechnic release mechanism plus 4 retro-rockets each	14 explosive bolts, fire-in-the-hole hot separation	Explosive bolts, retro-rockets	Spring jettison of payload

Vehicle Design

LM-4B

The LM-4B vehicle has two conventional lower stages plus a small third stage. The third-stage tanks have a common bulkhead and carry UDMH/N_2O_4 propellants. The LM-4B has an improved third stage and larger payload fairing.

	Stage 1	Stage 2	Stage 3
Dimensions			
Length	24.66 m (80.9 ft)	10.41 m (34.2 ft)	1.92 m (6.3 ft)
Diameter	3.35 m (11.0 ft)	3.35 m (11.0 ft)	2.9 m (9.5 ft)
Mass			
Propellant Mass	183.2 t (403.9 klbm)	35.6 t (78.5 klbm)	14.2 t (31.3 klbm)
Inert Mass	9.5 t (20.9 klbm)	4 t (8.8 klbm)	1 t (2.2 klbm)
Gross Mass	192.7 t (424.8 klbm)	39.6 t (87.3 klbm)	15.2 t (33.5 klbm)
Propellant Mass Fraction	0.95	0.90	0.93
Structure			
Type	Skin stringer	Skin stringer	Skin stringer?
Material	Aluminum	Aluminum	Aluminum?
Propulsion			
Engine Designation	YF-21B	YF-22B main engine and YF-23B vernier	YF-40
Number of Engines	1 YF-21 system consisting of 4 YF-20 engines	1 main engine + 1 vernier with 4 thrust chambers	2
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Average Thrust	Sea level: 2961.6 kN (665.6 klbf)	Main engine: 742 kN + Vernier: 4×11.8 kN (167 + 4×2.7 klbf)	98.1 kN (22.1 klbf)
Isp	Sea level: 260.7 s	Main engine: 298.0 s Vernier: 296.8 s	303 s
Chamber Pressure	?	?	?
Nozzle Expansion Ratio	?	?	?
Propellant Feed System	Gas-generator turbopump	Main engine: gas-generator turbopump Vernier: gas-generator turbopump	Gas-generator turbopump
Mixture Ratio (O/F)	2.1:1	2.1:1 main engine 1.57 verniers	?
Throttling Capability	100% only	100% only	100% only
Restart Capability	None	None	None
Tank Pressurization	Nitrogen bottle and self-pressurization	Nitrogen bottle and self-pressurization	?
Attitude Control			
Pitch, Yaw	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Hydrazine thrusters
Roll	Hydraulic gimbaling ±10 deg	Hydraulic gimbaling of vernier nozzles ±60 deg	Hydrazine thrusters
Staging			
Nominal Burn Time	170 s	Main engine:126.8 s Verniers: 137 s	303 s
Shutdown Process	Burn to depletion	Command shutdown	Command shutdown
Stage Separation	14 explosive bolts, fire-in-the-hole hot separation	Explosive bolts, retro-rockets	Spring ejection of payload

Vehicle Design

Attitude Control System

The first-stage propulsion system is composed of four mutually independent engines connected with an engine frame. Each single engine can swing in the tangential plane to provide three-axis attitude control of the vehicle. The engines of the liquid strap-on boosters are fixed. Several Long March configurations include fixed aerodynamic fins on the first stage or boosters for added stability.

The second stage is controlled by the four thrust chambers of the YF-23 vernier engine. Each vernier engine thrust chamber is connected to a servomechanism and can swing tangentially for attitude control of the second stage.

The third stage is controlled by the YF-73 for LM-3 and YF-75 for LM-3A, 3B, and 3C. On the LM-3, each of four YF-73 combustion chambers can gimbal in the tangential plane to provide attitude control for the flight of the third stage. On the LM-3A, B, and C, third-stage attitude control is provided by gimbaling the two YF-75 engines. Unlike the lower-stage engines, the YF-75 engines have two-axis gimbal capability. A reorientation control system is used for attitude control and propellant management after second-stage shutoff and for payload orientation after third-stage shutoff. This hydrazine monopropellant system is pressure-fed by helium. On the LM-3B this system uses two 300-N (67-lbf) and two 45-N (10-lbf) thrusters for propellant settling and twelve 70-N or 40-N (16- or 9-lbf) thrusters for three-axis attitude control.

Avionics

The control system includes the guidance system, attitude control system, a programmable sequencer, and the power distribution system. The guidance system uses a four-axis flexible inertial platform and digital computer. The programmed distribution system adopts a scheme that subsystems and large power equipment are supplied through the main and auxiliary distributors, respectively. The system conducts timing control over all airborne systems during flight.

The airborne telemetry system consists of sensors, encoders, transmitting devices, and recording and signal transmitting devices all over the vehicle. The tracking system consists of the ground continuous-wave tracking radars, single-pulse tracking radars, two airborne responders for the continuous-wave radars, one responder for the single-pulse radars, and the antennas of these responders. The vehicle flight condition is monitored by the vehicle telemetry system and ground receiving and real-time transmission equipment to provide initial injection information and range safety data. There are two types of destruction for range safety: radio telecommanded destruction and airborne independent self-destruction for when the attitude angle exceeds the allowable value. The telecommanded destruction subsystem consists of the ground telecontrol radars, a safety command receiver, and three linearly polarized antennas. These three antennas are installed circularly on the vehicle surface to form an all-around antenna array. The above responders and receiver are all mounted inside the second-stage intertank section, while the antennas are mounted on the outside surface of the same intertank section.

The LM-3B and 3C avionics can be mounted in one of two types of vehicle equipment bays (VEBs). The standard VEB mounts the avionics in a ring around the payload adapter inside the fairing. However, this requires that the spacecraft be encapsulated in the payload fairing on the launch pad. If offline encapsulation is preferred, a cylindrical VEB structure is mounted between the third stage and payload adapter–fairing assembly. This load-bearing structure is heavier, and therefore reduces performance.

Payload Fairing

The payload fairing design is a two half-shell structure with a longitudinal separation system. The nose dome is made of phenolic-resin glass cloth. The bicone is made of glass honeycomb. The faces and the core of the honeycomb are made of barium phenolic resin and glass cloth. For most of the fairings, noncontaminative explosive cords are used for longitudinal (i.e., between two half-shells) separation. Expansion of the gas in the capsule springs open the fairing. Lateral (i.e., between fairing and the lower stage) unlocking is initiated by explosive bolts unlocking the clampband, and the clampband is pulled to the fairing side by a traction spring. For some fairings, separation is realized through the exploding bolts releasing springs. Each explosive bolt is put in a box with a baffle on the inner side to avoid contamination of the payload when the bolt explodes. The fairing is generally jettisoned above 110 km (59 mi) altitude. Access doors may be placed on the cylindrical section to facilitate payload operations. Radio-transparent windows may be placed on the forward cone and cylindrical section at the user's request. In general, radio transparency is not less than 85%.

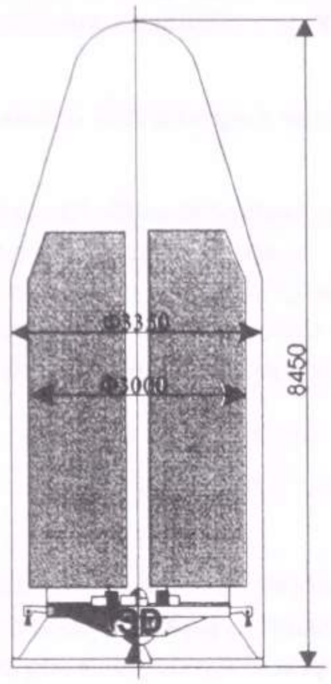
The LM-2C can also launch payloads without a fairing. The payload (typically a recoverable spacecraft) is attached directly to the forward frame of the vehicle equipment bay by four explosive bolts.

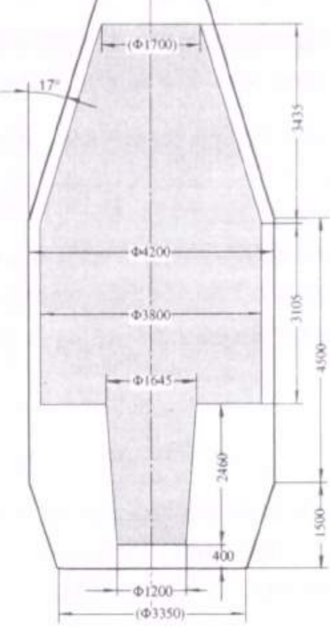
The LM-3B and -3C can use one of four fairing types. The fairings are designated by their diameter in millimeters, either 4000 mm or 4200 mm. A suffix indicates whether they are designed to be encapsulated on the pad, with the vehicle avionics mounted inside the fairing (type F), or encapsulated offline, with the avionics in a vehicle equipment bay below the fairing (type Z). The fairings are covered with cork for thermal insulation. Longitudinal separation is performed by redundant explosive cords; lateral separation is provided by 12 explosive bolts. The fairing halves are driven apart by six springs and rotate on hinges to clear the vehicle smoothly.

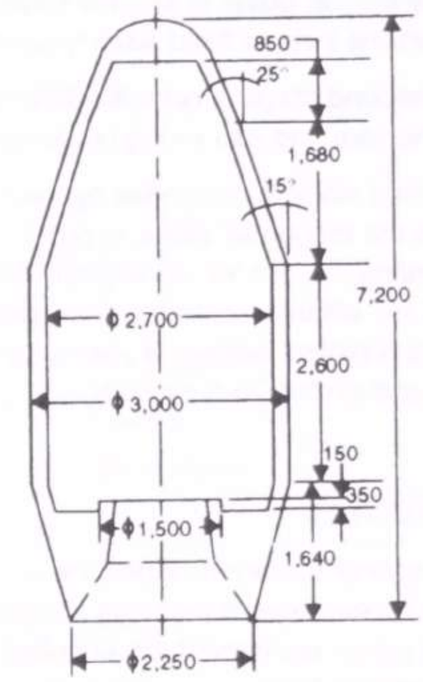
The LM-2F has a new payload fairing design that encapsulates a crewed spacecraft. The fairing is equipped with escape rockets mounted on a tower at the top of the fairing, which can pull the spacecraft to safety in the event of a launch vehicle failure.

Vehicle Design

Note: Fairing diagrams are not to scale.

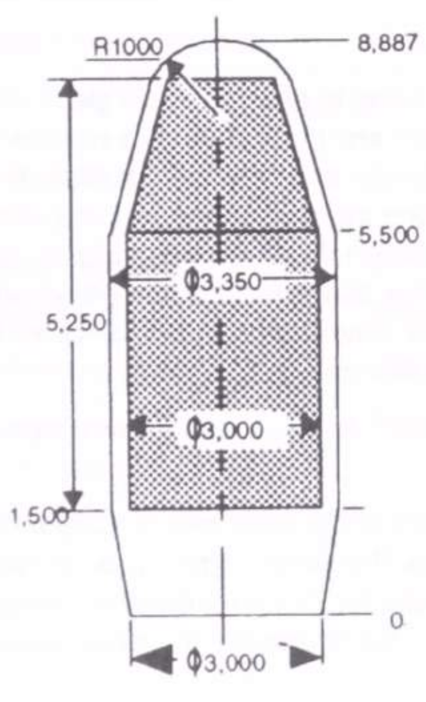


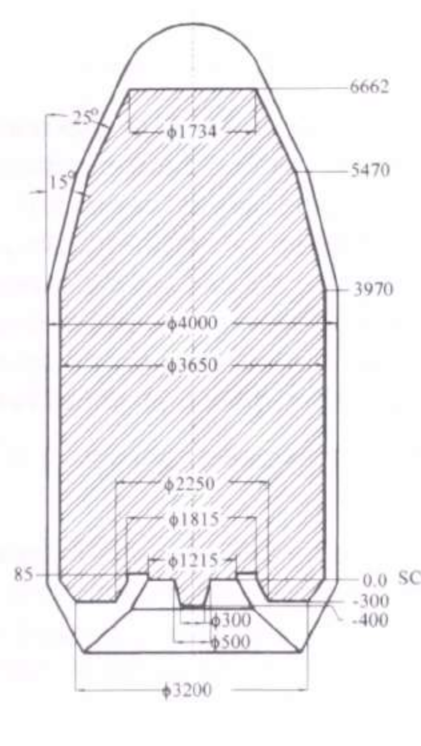


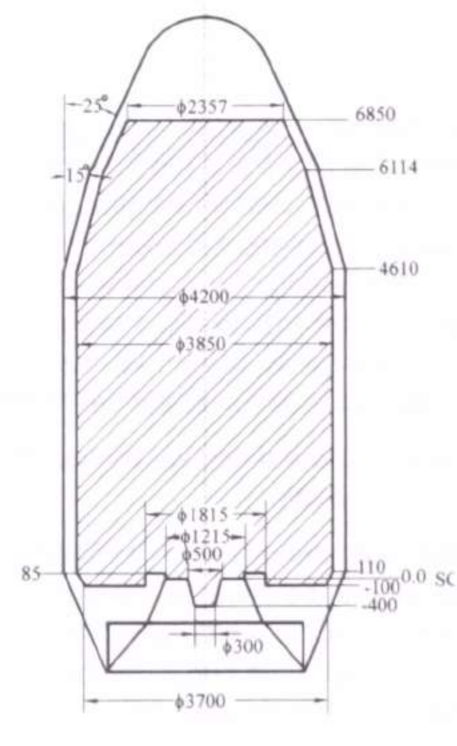


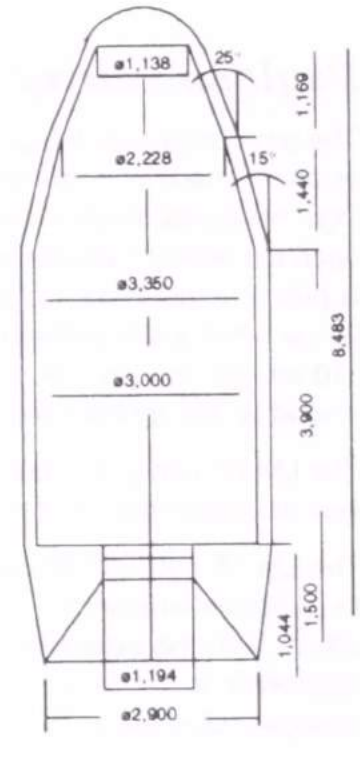
	LM-2C	LM-2E	LM-3 Model B
Length	LM-2C Type B: 7.12 m (23.4 ft) LM-2C/SD: 8.45 m (27.7 ft)	10.45 m (34.3 ft)	7.27 m (23.9 ft)
Primary Diameter	3.35 m (11.0 ft)	4.2 (13.8 ft)	3.0 m (9.8 ft)
Mass	?	?	?
Sections	2	2	2
Structure	Composite sandwich	Composite sandwich	Composite sandwich
Material	Glass fiber over honeycomb	Glass fiber over honeycomb	Glass fiber over honeycomb

Courtesy China Academy of Launch Vehicle Technology









	LM-3A	LM-3B Model	LM-3B Model 4000Z/F	LM-4B 4200Z/F
Length	8.89 m (29.2 ft)	4000Z: 8.58 m (28.1 f) 4000F: 9.56 m (31.4 ft)	4200Z: 9.38 m (30.8 ft) 4200F:9.78 m (32.1 ft)	8.48 m (27.8 ft)
Primary Diameter	3.35 m (11.0 ft)	4.0 m (13.1 ft)	4.2 m (13.8 ft)	3.35 m (11.0 ft)
Mass	?	?	?	?
Sections	2	2	2	2
Structure	Composite sandwich	Composite sandwich	Composite sandwich	Composite sandwich
Material	Glass fiber over honeycomb	Glass fiber over honeycomb	Glass fiber over honeycomb	Glass fiber over honeycomb

Payload Accommodations

	LM-2C	LM-2E	LM-3B
Payload Compartment			
Maximum Payload Diameter	3350 mm (131.9 in.)	3800 mm (149.6 in.)	3650 mm or 3850 mm (143.7 or 151.6 in.)
Maximum Cylinder Length	2000 mm (78.7 in.)	3105 mm (122.3 in.)	4610 mm (181.5 in.)
Maximum Cone Length	3000 mm (118.1 in.)	3435 mm (135.2 in.)	2240 mm (88.2 in.)
Payload Adapter Interface Diameter	937 mm (36.7 in.) 1194 mm (47.0 in.)	1627 mm (64.1 in.) 1728 mm (68.0 in.) 3114 mm (122.6 in.) 2306 mm (90.8 in.) 2002 mm (78.8 in.)	937 mm (36.7 in.) 1194 mm (47.0 in.) 1666 mm (65.6 in.)
Payload Integration			
Nominal Mission Schedule Begins	?	24 months	24 months
Launch Window			
Last Countdown Hold Not Requiring Recycling	?	?	?
On-Pad Storage Capability	?	?	?
Last Access to Payload	?	?	?
Environment			
Maximum Axial Load	7.9 *g*	5.6 *g*	6.1 *g*
Maximum Lateral Load	1.0 *g*	1.2 *g*	1.5 *g*
Minimum Lateral/Longitudinal Payload Frequency	?	10/26 Hz	10/30 Hz
Maximum Acoustic Level	137 dB at 500 Hz	136 dB at 500 Hz	136 dB at 250 Hz
Overall Sound Pressure Level	140 dB	140 dB	141 dB
Maximum Flight Shock	1500 *g* above 700 Hz	1200 *g* above 1500 Hz	4000 *g* at 1500-5000 Hz
Maximum Dynamic Pressure on Fairing	?		?
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1BTU/ft^2/s)	1135 W/m^2 (0.1BTU/ft^2/s)	1135 W/m^2 (0.1BTU/ft^2/s)
Maximum Pressure Change in Fairing	?	?	6.9 kPa/s (1.0 psi/s)
Cleanliness Level in Fairing	Class 100,000	Class 100,000	Class 100,000
Payload Delivery			
Standard Orbit Injection Accuracy (3 sigma)	LEO: Semimajor axis: 3 km (1.6 nmi) Inclination: ±0.15 deg	200×400 km (108×216 nmi) 63.4 deg LEO Perigee: ±6 km (3.3 nmi) Eccentricity: ±0.00066 Inclination: ±0.15 deg	Standard GTO: Semimajor axis: 120 km (64.8 nmi) Inclination: ±0.21 deg Perigee altitude: ±30 km (16.2 nmi)
Attitude Accuracy (3 sigma)	Pitch and yaw: ±1.4 deg, ±0.35 deg/s Roll: ±0.5 deg, ±0.25 deg/s	Spinning: ±4.5 deg, ±0.9 deg/s Nonspinning: Pitch and yaw: ±1.5 deg, ±0.6 deg/s Roll: ±0.6 deg, ±0.45 deg/s	<1 deg
Nominal Payload Separation Rate	>0.7 m/s (2.3 ft/s)	1.0 m/s (3.3 ft/s)	>0.5 m/s (1.6 ft/s)
Deployment Rotation Rate Available	Spin-up requires nonstandard equipment	5–8 rpm	up to 10 rpm
Loiter Duration in Orbit	?	?	?
Maneuvers (Thermal/Collision Aviodance)	Yes	Yes	Yes
Multiple/Auxiliary Payloads			
Multiple or Comanifest	Comanifest payloads have not been carried in the past. However, multiple payloads for a single customer can be carried. For example, pairs of Iridium spacecraft have been launched on the LM-2C/SD.	No comanifest capability has been reported. Multispacecraft capability has been advertised for LEO constellations, using mission specific adapters for either bottom or side mounting.	No comanifest capability has been reported. The LM-3B will likely be capable of carrying multiple LEO spacecraft for a single customer on mission-specific dispensers.
Auxiliary Payloads	No standard auxiliary payload interface has been reported, but one auxiliary payload, Sweden's Freja spacecraft, was deployed on LM-2C. Contact CGWIC for further information.	No standard auxiliary payload interface has been reported, but one auxiliary payload, Pakistan's Badr spacecraft, was deployed on LM-2E. Contact CGWIC for further information.	The LM-3 series has not carried auxilliary spacecraft in the past, but flight opportunities may be available. Contact CGWIC for more more information.

Production and Launch Operations

Production

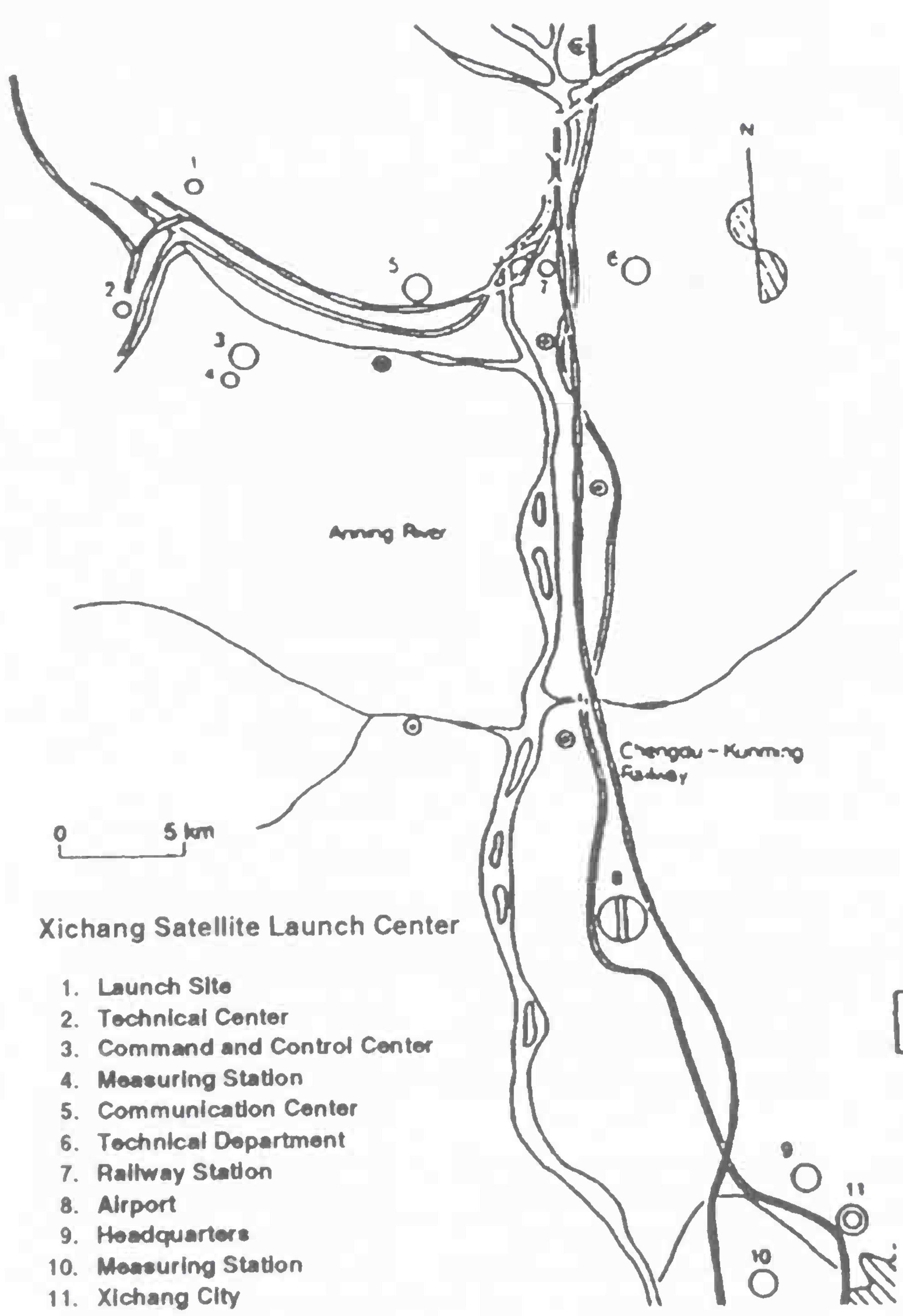
Chinese space industry was restructured in the late 1990s. In 1999 the newly formed China Aerospace Science and Technology Corporation (CASC) took over more than 130 organizations from the predecessor China Aerospace Industry Corporation (also CASC). CASC is a large government-owned organization employing more than 110,000 people. It operates most of China's major space research and manufacturing facilities, including the China Academy of Launch Technology (CALT), Shanghai Academy of Space Technology (SAST), and China Satellite Launch and Tracking Control General (CLTC), as well as the international trade company, China Great Wall Industry Corporation (CGWIC).

CALT is the lead organization for development, production, and mission integration for most of the Long March variants. It is located south of Beijing and employs roughly 25,000 people. Its partner and occasional rival, SAST (once known as the Shanghai Bureau of Astronautics) is located in Shanghai and is responsible for parts of the LM-3 series vehicles as well as the LM-4 and LM-2D. It was also the creator of the Feng Bao, a competing variation of the LM-2C in the 1970s. CLTC runs China's three launch centers, the satellite control center in Xi'an, and a global tracking, telemetry, and control network that includes a fleet of tracking ships. CLTC performs the launch operations required to launch Long March vehicles from each of its three launch sites.

Manufacturers responsible for Long March stages

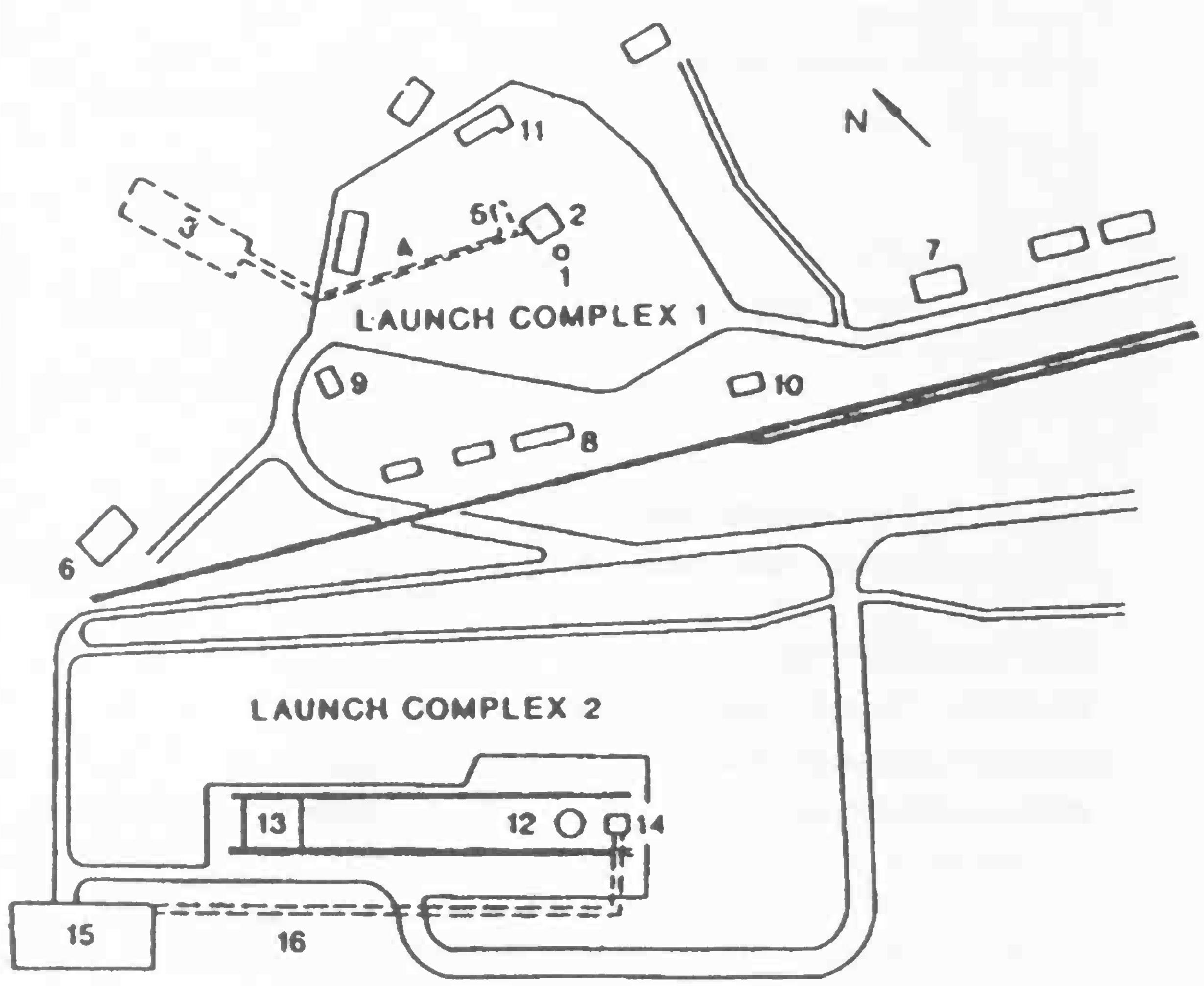
	Stage 1	Stage 2	Stage 3
LM-2C	CALT	CALT	CALT?
LM-2E	CALT	CALT	Hexi Company
LM-3, 3A, 3B, 3C	SAST	SAST	CALT
LLM-4B	SAST	SAST	SAST

Launch Facilities



Xichang Satellite Launch Center

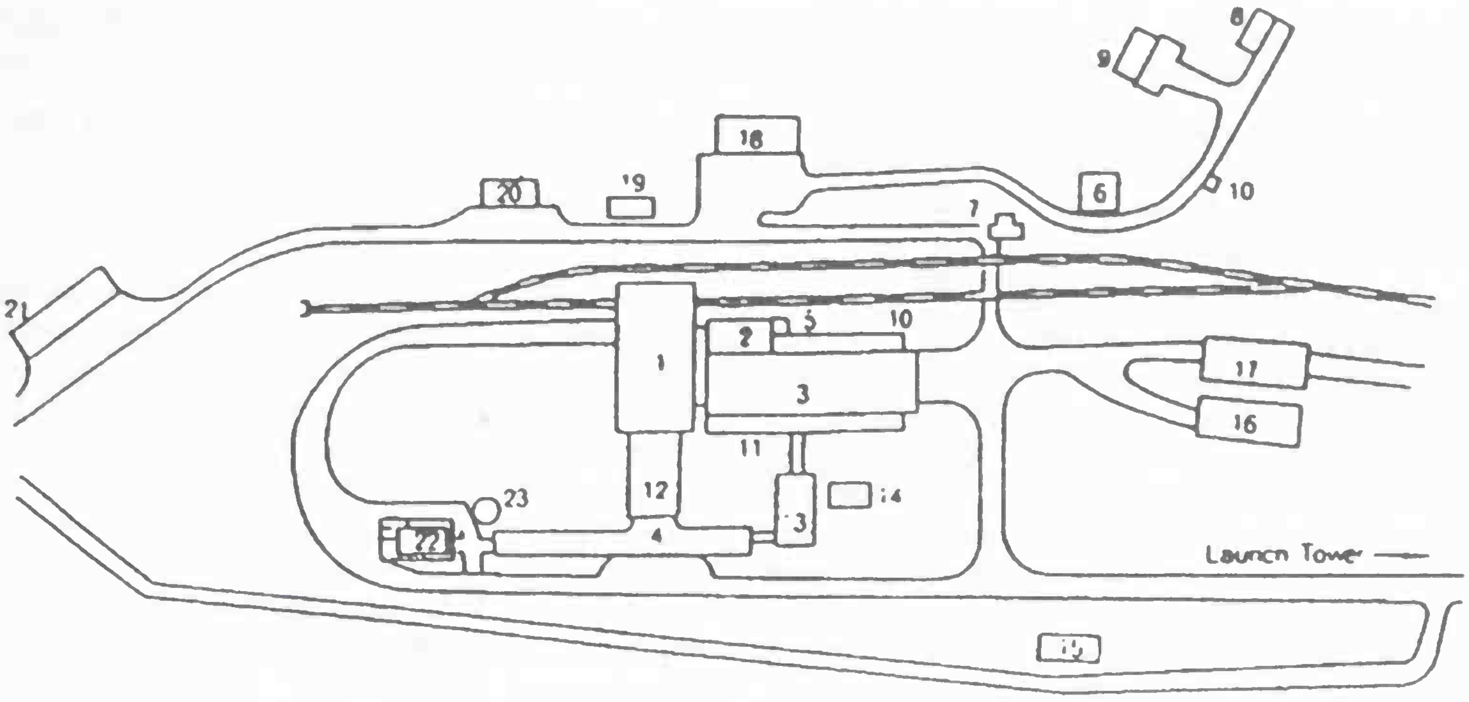
1. Launch Site
2. Technical Center
3. Command and Control Center
4. Measuring Station
5. Communication Center
6. Technical Department
7. Railway Station
8. Airport
9. Headquarters
10. Measuring Station
11. Xichang City



XSLC Launch Area

1. Launch Pad (No. 1)
2. Launch Tower (No. 1)
3. Launch Control Room
4. Cable Ditch
5. Battery Room
6. Fuel Store
7. Oxidizer Store
8. Gas Vessel Store
9. Aiming Room
10. LH Filling Room
11. LOX Filling Room
12. Launch Pad (No. 2)
13. Movable Tower
14. Launch Tower (No. 2)
15. Launch Control Room
16. Cable Ditch

Production and Launch Operations



XSLC Technical Center

1. Transit Hall
2. Payload Hangar BS1
3. Launch Vehicle Checkout Hangar
4. Component Checkout Hangar
5. Hydrazine Filling Hangar
6. Apogee Engine Hangar
7. Leakage Inspection Hangar
8. Pyrotechnics Store
9. Pyrotechnics Checkout Room

10-12. Working Rooms

13. Air Conditioning Equipment
14. Transformer Station
15. Boiler Room

16-21. Stores

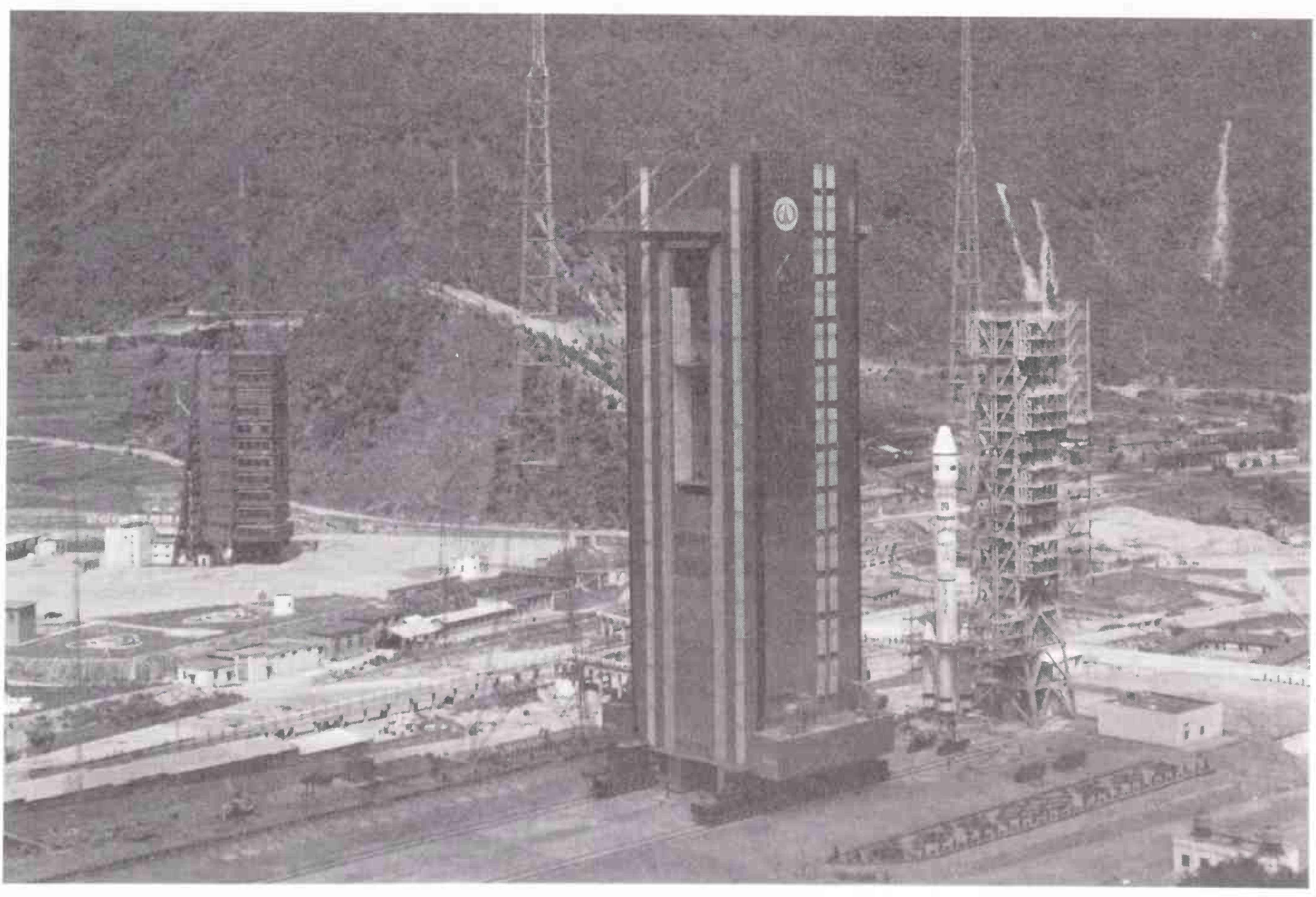
22. Payload Hangar BS2, BS3
23. Antenna Tower

Launch Operations

China has three launch sites for the Long March family. The Xichang Satellite Launch Center (XSLC) supports the LM-2C, LM-3, LM-3A, LM-2E, and LM-3B and is generally used for low-inclination LEO and GTO launches. The Jiuquan Satellite Launch Center (JSLC) supports the LM-2C and is used for mid-inclination LEO launches. Jiuquan is being expanded to support the LM-2F. The Taiyuan Satellite Launch Center (TSLC) launches the LM-2C, LM-2C/SD, and the LM-4 series and is used for launches to polar or sun-synchronous orbits. All of China's launch sites are landlocked. Designated launch azimuths are used to control the impact of launch vehicle stages much like the Russian approach. However, China is much more densely populated than Siberia or Kazakhstan, and it is not uncommon for a launch failure to cause casualties or property damage. Even successful launches can be dangerous. In October 2002 debris from the launch of a LM-4B struck the village of Yanghe near a designated impact zone, injuring one person.

Xichang Satellite Launch Center

XSLC became operational in 1984 for launching Chinese domestic communications satellites on the LM-3. XSLC is in a mountainous area 64 km (40 mi) northwest of Xichang City in Sichuan Province. It is approximately 1800 m (5900 ft) above sea level. The geographical coordinates of the launch site are 28.2° N, 102.0° E. At the site of XSLC the climate is subtropical, with a long frost-free period and clearly divided dry and rainy sea-



View of Launch Complex 1 (left background) and Complex 2 at Xichang Satellite Launch Center. An LM-2E is stacked on Pad 2.

Production and Launch Operations

sons. The rainy season is the period from June to September. The average annual temperature is 16°C (61°F), with the average highest temperature at 25°C (77°F) in summer, and the average lowest temperature at 2°C (36°F) in winter. Wind speed at the launch site is very low all year round.

The Xichang Airport, located in the northern suburbs of Xichang City, has been upgraded to accommodate 747 and C-130 type aircraft. Xichang Airport is 13.5 km (8.4 mi) away from Xichang City and 50 km (31 mi) away from the launch site. The Chengdu–Kunming Railway and the western part of the Sichuan–Yunnan highway pass through XSLC. There is a dedicated railway branch and a dedicated highway branch leading straight to the launch site. The maximum slope of the railway is 3%. The maximum slope of the highway is 5.6%.

The technical center, which is used for launch vehicle and payload test and checkout, consists of a number of facilities such as the launch vehicle test building, payload preparation building, and hazardous processing building. Long March stages are shipped by rail to the launch site. A vehicle will spend about five weeks at the launch vehicle checkout hangar in a horizontal mated position for checkout before being disassembled and trucked in stages about 2.2 km (1.3 mi) north to the launch area on a 8-m (26-ft) wide cement road. Integrated checkout of the vehicle and payload is done only at the pad, not in the center. The payload preparation building is for nonhazardous assembly, integration, and testing operations for spacecraft and, if required by the mission, upper stages. It is large enough to accommodate at least two spacecraft in tandem in the assembly and test hall in Class 100,000 clean-room conditions. In addition, there are Class 10,000 clean rooms for assembly and test operations on equipment requiring a particularly clean environment. The hazardous processing building is for hazardous assembly operations, spacecraft propellant fueling and pressurization, solid-motor integration, installation of electro-explosive devices, spin-balancing of the payload, payload and/or upper stage weighing, and hazardous testing. Encapsulation of the spacecraft in the launch vehicle fairing is performed here for the LM-2E, LM-3A, and optionally for the LM-3B. For the LM-3 or the LM-3B, encapsulations can take place on the launch tower inside the environmental enclosure.

After leaving the facilities in the technical center either inside an environmentally controlled payload container or inside the fairing, the spacecraft is taken to the launch area. The launch area is the site for performing the tasks of mating, testing and checkout, direction orientation, propellant filling, and launching of the Long March. There are two launch complexes at XSLC, about 0.8 km (0.5 mi) apart. Pad 1 is used to launch the LM-2C and LM-3C. Pad 2 supports the LM-2E, LM-3A, and LM-3B. It took 15 months to construct the new pad for the first launch of LM-2E.

Pad 1 has a 77-m (252-ft) tall LM-3 launch tower fixed next to the pad that includes a rotating column crane on top to stack stages and the payload. The spacecraft is integrated with, and tested on the launch vehicle, inside an environmental enclosure at the top of the launch tower. The enclosure has a cleanliness level of Class 100,000. During launch, an area of 5 km^2 (2 mi^2) around the pad is evacuated and local residents are allowed to return about 10 min after launch.

Pad 2 includes a launch pad, mobile service tower, fixed umbilical tower, launch control center, fueling facilities, and lightning towers. The mobile services tower is 90.6 m (297 ft) high with two cranes with lifting capacities of 20 t (44 klbm) and 10 t (22 klbm) for assembly of the launch vehicle. The tower is topped by a Class 100,000 clean room for encapsulation of the payload. A rotating crane with a capacity of 8 t (17.6 klbm) can be used to lift the payload. The umbilical tower has swinging arms that provide work platforms, cryogenic fueling, power, and air conditioning.

The primary purposes of the command and control center are to control operations, to receive, process and disseminate telemetry data from the launch vehicle, and to monitor and control the safety of the launch vehicle before and during flight. Downrange measuring stations under the management of XSLC are distributed in Xichang and Yibin of Sichuan Province and Guiyang of Guizhou Province.

There is a network of telemetry, tracking, and control (TTC) stations, including two tracking ships, which are used for TTC operations during the liftoff and flight. XSLC has a weather station that is connected to the national weather network. Other facilities include a chemical analysis laboratory, workshop facilities, and comprehensive medical facilities at the launch site, as well as a hospital at Xichang City. To alleviate concerns over technology transfer, many of the services that would normally be performed by the launch service provider, such as photographic and videographic documentation of preparation processes or fuel testing, can be performed by the launch customer instead.

Jiuquan Satellite Launch Center

JSLC, previously known in the West as Shuang Cheng Tzu, is China's original launch site. JSLC is located about 1600 km (1000 mi) west of Beijing, to the north, at the tail of the Great Wall, on the edge of the Gobi Desert in Gansu Province in northwestern China. It is approximately 1000 m (3300 ft) above sea level. Historically, orbital missions are launched southeast to avoid overflying Mongolia and Russia and enter LEO in a narrow band between 56.9- and 69.9-deg inclination. This also allows most of the trajectory to be monitored from tracking stations within China. Although XSLC is now used for GEO launches, JSLC continues to be used for recoverable reconnaissance and Earth resources satellites as well as China's human spaceflight program.

Jiuquan is accessible by air through the Dingxin airport 80 km (50 mi) to the southwest, by rail on the Gansu–Xinjiang railway, or by highway. The space center is divided into a central headquarters with a hotel, tracking facilities, and the command center; a North Launch Site for LM-2C and LM-2D; and a new South Launch Site for the LM-2E and LM-2F. Each launch site is divided into a technical center, which includes integration and processing facilities for the spacecraft and launch vehicles, and a launch center, where the launch pads are located. The North Launch Site is the older of the two and is set up in the traditional style with fixed launch pads and umbilical towers, which can be enclosed in a 55 m (180 ft) mobile service tower for vehicle assembly. There are two pads—Pad 1 and 2.

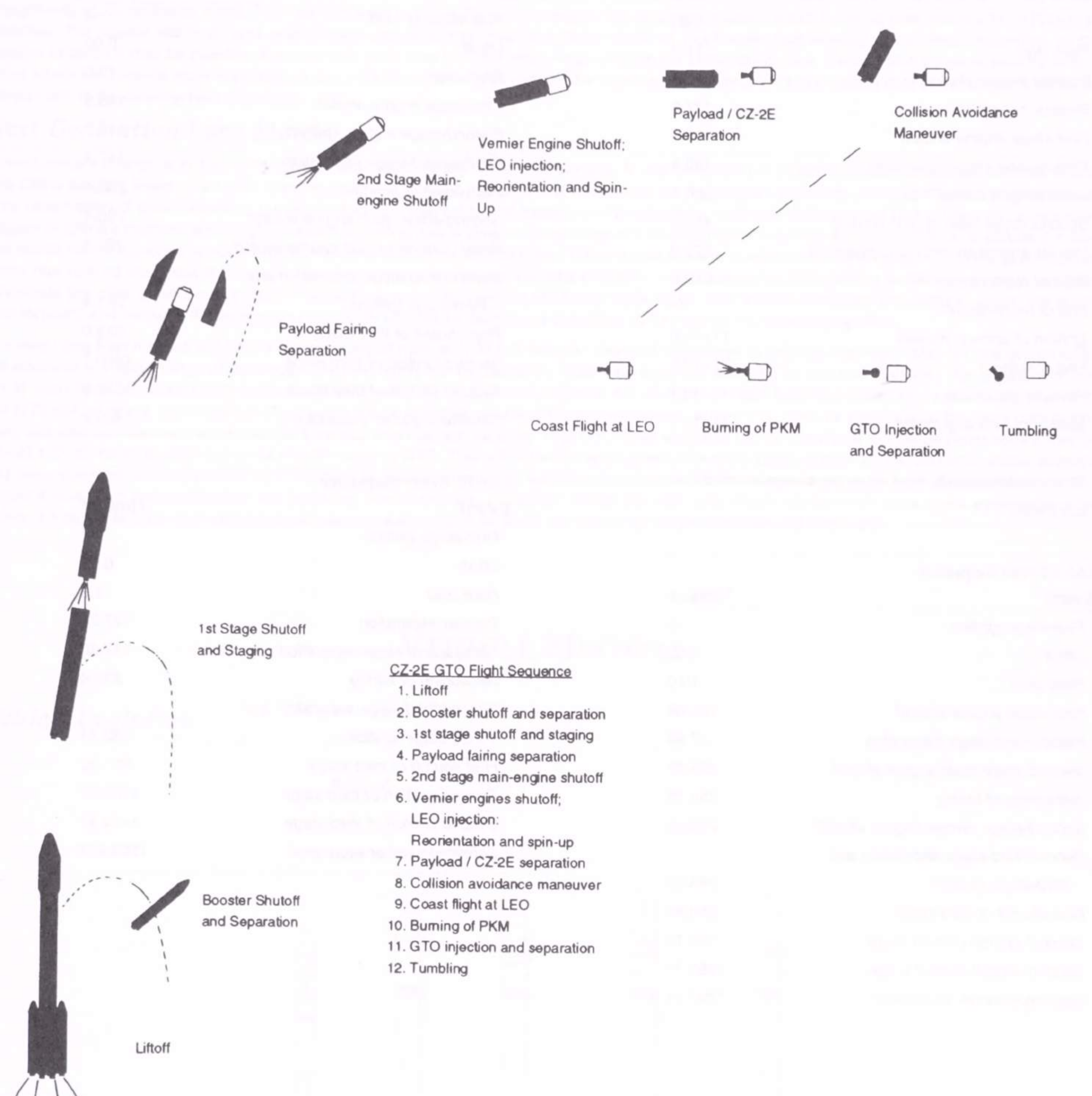
The new South Launch Site is a departure from standard Chinese launch facilities. Built in the late 1990s, primarily to support China's manned spaceflight program, the South Launch Site follows an integrate–transfer–launch approach similar to the Space Shuttle, H-IIA, or Ariane 5, in which the vehicle is built up and tested vertically in a large, fixed facility, and then transferred to the launch area. Stages for the LM-2F or LM-2E are first received and checked out in the Horizontal Transit Building (BL1) in a 78 x 24 m (256 x 79 ft) hall. They are then transferred to one of two high bays in the adjacent Vertical Processing Building (BLS), which looks similar to the Space Shuttle VAB. The launch vehicle is integrated in one of the 26.8 x 28 x 81.4 m (88 x 92 x 267 ft) vertical bays, mounted on top of a 750 t (1.6 Mlbm) mobile launch platform. Meanwhile, the spacecraft can be processed and fueled in a series of nearby processing and hazardous operations buildings. When the fully integrated launch vehicle is ready, the mobile launch platform transports it 1.5 km (0.9 mi) to the launch pad at a top speed of 1.7 km/h (1 mi/h). The fixed umbilical tower at the pad is an 11-story steel structure with a height of 75 m (246 ft). The umbilical tower includes a rotating service structure that can enclose the vehicle for access at the pad.

Taiyuan Satellite Launch Center

TSLC is based amid rugged terrrain in Shanxi Province, 500 km (310 nm) southwest of Beijing. The site is used to launch LM-4 vehicles southward into polar orbit. It was used for the first time for a Long March launch in September 1988 for China's first weather satellite. This was the first launch of a LM-4. More recently, TSLC has been used as the launch site for a number of commercial Iridium spacecraft on LM-2C/SD launch vehicles. Few details have been released about this launch site.

Production and Launch Operations

Flight Sequence



LM-2E Typical Flight Sequence

Production and Launch Operations

LM-2E Event Sequence for LEO Mission

Event	Time, s
Liftoff	0.00
Pitch over	11.0
Booster engine shutoff	125.8
Booster separation	127.3
First-stage engine shutoff	158.9
First/second-stage separation	160.4
Jettisoning of fairing	200.4
Second-stage main engine shutoff	459.9
Vernier engine shutoff (LEO injection)	572.9
Start of reorientation	582.9
End of reorientation	T1
Ignition of spin-up rockets	T1+10.0
End of spin-up	T1+10.5
Payload separation	T1+10.7
Ignition of tumbling rocket	T1+14.7

T1 = mission-specific time when payload reorientation is completed

LM-3 Event Sequence

Event	Time, s
First-stage ignition	–3
Liftoff	0.00
Pitch over	10.0
First-stage engine shutoff	126.66
First/second-stage separation	127.89
Second-stage main engine shutoff	255.25
Jettisoning of fairing	259.25
Second-stage vernier engine shutoff	262.25
Second/third-stage separation and third-stage ignition	263.25
First shutoff of third stage	688.88
Second ignition of third stage	935.70
Second shutoff of third stage	1253.71
Satellite/launcher separation	1292.71

LM-3A Event Sequence

Event	Time, s
First-stage ignition	–3
Liftoff	0.00
Pitch over	13.0
First-stage engine shutoff	145.9
Second-stage engine ignition	146.7
First/second-stage separation	147.4
Jettisoning of fairing	232.4
Second-stage main engine shutoff	255.4
Second-stage vernier engine shutoff	260.4
Second/third-stage separation and third-stage ignition	261.4
First shutoff of third stage	619.0
Second ignition of third stage	1200.1
Second shutoff of third stage	1320.8
Satellite/launcher separation	1420.8

LM-3B Event Sequence

Event	Time, s
First-stage ignition	–3
Liftoff	0.00
Pitch over	11.0
Booster separation	127.24
First/second-stage separation	147.36
Jettisoning of fairing	232.4
Second/third-stage separation and third-stage ignition	332.31
First shutoff of third stage	631.32
Second ignition of third stage	1281.23
Second shutoff of third stage	1459.91
Satellite/launcher separation	1559.907

Vehicle Upgrade Plans

Hainan Island Launch Site

For several years Chinese authorities have studied the creation of a new launch site on Hainan Island, either near Sanya, on its southern coast or near Wengchang on its northeast coast. The new site would offer multiple advantages. The southerly location would increase performance for GTO-bound launches. The coastal site in an area less densely populated than mainland China would be much safer than existing inland sites. Depending on the location selected it may be possible to launch both polar and low inclination missions from the same launch site. It would also be accessible by ship, a factor which will become more important as Long March vehicles grow larger. The estimated 2-billion-CNY ($242 million) cost of the project may be an impediment to the development of the site.

Next Generation Long March

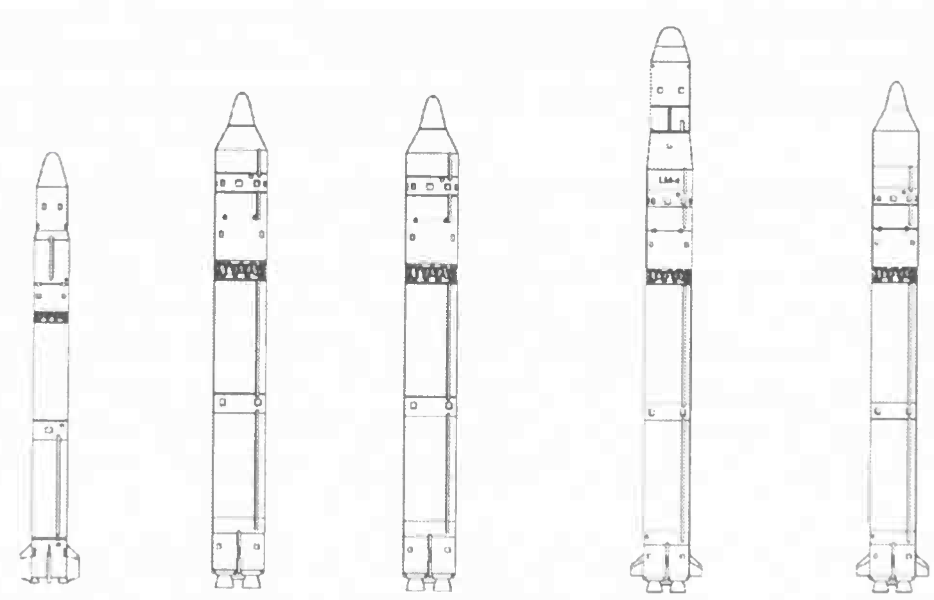
A revolutionary change is in the works for the Long March family in the next decade. A new generation of launch vehicles, sometimes referred to outside China as Long March 5 or LM-X, is in the preliminary design phase. China's objectives for the program are driven partly by internal needs and partly by observation of global trends in the launch industry. Internal needs include a requirement for a system that can launch much heavier payloads in support of China's manned spaceflight program, including space station components and eventual lunar missions. Another priority is to reduce the cost and failure rate of launches by consolidating missions onto fewer vehicle types. There is also a desire to eliminate toxic propellants. On the global scene, China has noticed that while the LM-3B was once among the largest vehicles available, every competitor is beginning to field even larger vehicles. Worldwide the older generation of missile-derived space launch vehicles are being replaced by new vehicles designed from scratch, which are large, cost-focused, and modular. A new larger Long March with similar features is perceived to be required to remain competitive.

The new Long March series will be the first not derived from an ICBM. A modular approach is planned to achieve high flight rates for one design while still addressing a large range of payloads. The new family will be based on three new elements powered by two new engines. The largest is a 5-m (16 ft) diam hydrogen-fueled core stage powered by two 500-kN (110-klbf) engines. A 3.35-m (11-ft) diam kerosene-fueled stage powered by two 1.2-MN (270-klbf) engines, and a 2.25-m (7.4-ft) diam stage powered by one of the same engines can each be used as strap-ons on the large core stage. They can also be used as the core for medium and small launch vehicles. Together, these elements can be combined in various permutations that lift 1.5–25 t (3–55 klbm) to LEO or 1.5–14 t (3–31 klbm) to GTO. This will allow the deployment of a small space station for the domestic space program and allow commercial service for two 6-t (13-klbm) heavy communication satellites comanifested on one large launch vehicle; thus competitive with the largest Ariane 5. In conjunction with the proposed new launch site on Hainan Island, the new Long March series would be a major advancement for China. If fully funded, the new vehicles could be available as soon as 2008, but they may not be launched until after 2010.

Vehicle History

Vehicle Evolution

Retired

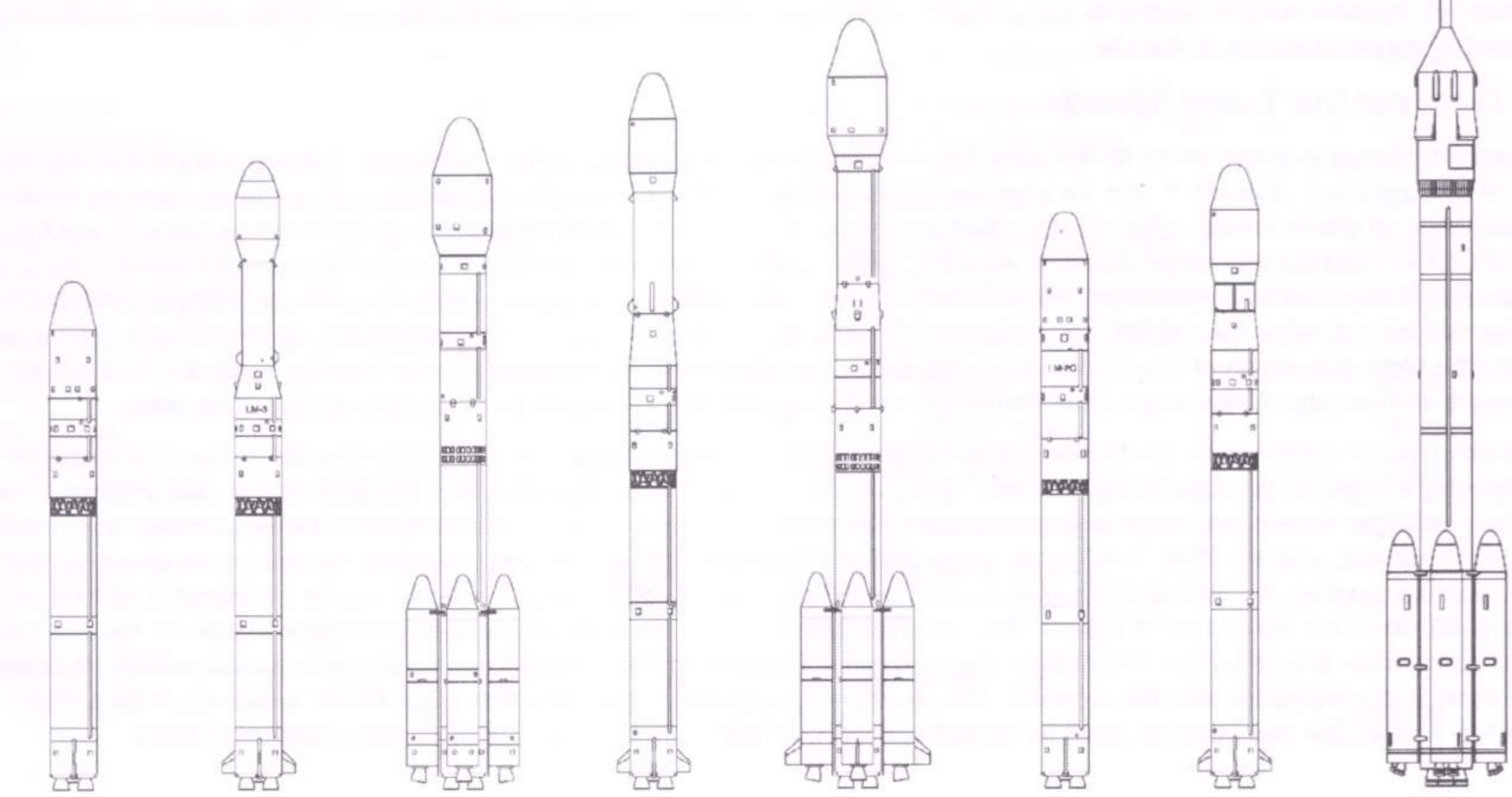


Vehicle	**LM-1**	**LM-2**	**FB-1**	**LM-4**	**LM-2D**
Period of Service	1970–1971	1974	1974–1981	1988–1990	1992–1996
LEO Payload	300 kg (660 lbm)	2200 kg (4400 lbm)	1200 kg (2600 lbm)		3500 kg (7700 lbm)

Vehicle History

Vehicle Evolution

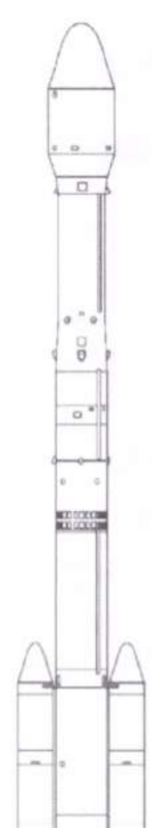
Operational



Vehicle	LM-2C	LM-3	LM-2E	LM-3A	LM-3B	LM-2C/SD LM-2C/CTS	LM-4B	LM-2F
Period of Service	1975–Present	1984–Present	1990–Present	1994–Present	1996–Present	1997–Present	1999–Present	2000
LEO Payload	3900 kg (8600 lbm)		9500 kg (20950 lbm)		11,200 kg (28.5 deg)	?	?	?
GTO Payload	1400 kg (3100 lbm)	1500 kg (3300 lbm)	3500 kg (7710 lbm)	2600 kg (5700 lbm)	5100 kg (11,250 lbm)			

(Current performance is shown. Original performance may have been lower for some vehicles.)

In Development



Vehicle	LM-3C
Period of Service	?
LEO Payload	?
GTO Payload	3800 kg (8400 lbm)

Vehicle History

Vehicle Description

•LM-1	Three-stage vehicle derived from the DF-4 (CSS-3) IRBM. First two stages used nitric acid/UDMH propellant and the third stage was solid. One engine each for first and second stage, both controlled by jet vanes. 2.05-m (6.7-ft) diameter fairing.
•LM-2	Two-stage vehicle derived from the DF-5 (CSS-4) ICBM. Both stages use N_2O_4/UDMH. Four first-stage engines with gimbal control and one second-stage engine with four verniers for control.
•FB-1	Two-stage liquid vehicle. Similar to LM-2.
•LM-2C	Similar to LM-2 with upgrades for improved reliability and performance.
•LM-3	Derivative of LM-2C with aerodynamic fins on first stage and new LOX/LH_2 third stage used for GTO missions. 2.6-m (8.5-ft) and 3.0-m (9.8-ft) diameter fairings.
•LM-4	Derivative of LM-2C with aerodynamic fins on first stage, stretched first and second stages and addition of a new N_2O_4/UDMH third stage. 2.9-m (9.5-ft) and 3.35-m (11.0-ft) diameter fairings.
•LM-2E	Similar to LM-2C with stretched stages and four N_2O_4/UDMH strap-on boosters for increased performance. 4.2-m (13.8-ft) diameter fairing. CPKM available for GTO missions.
•LM-2D	Similar to LM-4 without third stage.
•LM-3A	Similar to LM-3 with stretched first two stages and a new LOX/LH_2 third stage with two engines derived from the LM-3 first stage. 3.35-m (11.0-ft) and 4.0-m (13.1-ft) diameter fairings.
•LM-3B	Similar to LM-3A with strap-on boosters from LM-2E. Used for heavy GTO payloads.
•LM-2C/SD	Modification of LM-2C with new third stage for orbit circularization and deployment of two Iridium spacecraft.
•LM-4B	Similar to LM-4 with improved third stage and larger diameter payload fairing.
•LM-3C	Similar to LM-3B with two rather than four strap-on boosters.
•LM-2F	Modification of LM-2E with improved reliability to launch humans.

Historical Summary

The first rockets, similar to modern fireworks, were invented in China in 970 by Feng Jishen. The Chinese word for rocket means firing arrow. Rockets were used in battle by the Chinese in 1083 and 1275. These early rockets included the fundamental elements of a modern rocket: a combustion chamber, firing system, energetic fuels, and a stabilizer system.

Dr. Tsien Hsue-shen (also known as Qian Xuesen) led the development of the modern Chinese space program. During the 1930s he studied briefly at the Massachusetts Institute of Technology, later ultimately graduating with a doctorate in aeronautical engineering from the California Institute of Technology (CalTech), where he was a protégé of Theodore von Kármán, studying with and later working for him. During the 1940s he cofounded the Jet Propulsion Laboratory and worked for the U.S. Army on the first U.S. missile, the Private A. At the end of World War II, Tsien was part of the U.S. team that scoured Germany in search of the German V-2 rocket secrets. After the war he published a textbook entitled *Jet Propulsion*. In 1950 he became the Robert Goddard Professor of Jet Propulsion at CalTech. In the summer of 1950, he became a target in Senator Joseph McCarthy's investigations and was accused of being a communist. His security clearance was revoked, and he was placed under house arrest. For years he was caught up in legal disputes, and despite pleas and testimony by many of his colleagues, who still strongly defend his innocence, Tsien was ultimately deported to China in September 1955. He and 93 other scientists were exchanged for 76 U.S. prisoners of war taken in Korea. Tsien was warmly welcomed in China, and China quickly proceeded to establish a space program.

On 17 February 1956 Tsien submitted a proposal to establish research facilities for aeronautics and missile development. On 8 October 1956, the Central Committee of the Communist Party established the Fifth Research Academy of the Ministry of National Defense to develop a Chinese space program, and Tsien was appointed its first director. China, realizing it didn't have the capability to develop even basic rocket hardware, turned to the Soviet Union for help. The Soviets agreed to supply missiles, documentation, and specialists, with the first R-2 rockets arriving in 1958 along with 100 specialists. The Chinese proceeded to set up Project 1059 with the initial goal of copying the Russian R-2. Soviet help was required to develop even basic technical skills such as welding. China began construction of its first launch site, now known as Jiuquan Satellite Launch Center (JSLC), in April 1958 at Shuang Chengzi base, located in the Gobi Desert. Relations between the Soviet Union and China broke down 12 August 1960, primarily over the Soviet Union's refusal to share some aspects of nuclear technology, resulting in the abrupt departure of the approximately 1400 Russian technical advisors in China.

On 5 November 1960, the Chinese launched their first indigenously built missile, a copy of the R-2, which they named the Dong Feng 1 (East Wind). By 1961 China had 14 factories with upwards of 15,000 people working on its missile and space program. After the success of Dong Feng 1, the Chinese proceeded to design and build the Dong Feng 2, having a range of 1500 km. The first launch attempt of the Dong Feng 2 failed on 21 March 1962, and it did not achieve a successful flight until 29 June 1964.

On 10 August 1965, Zhou Enlai approved Project 651 with the goal of building a satellite, which stipulated "that the satellite should be visible from the ground and that its signals should be heard all over the world." Project 651 combined the Shanghai Institute (which was moved to Beijing and renamed the Seventh Research Division of the Eighth Institute of Design in the Seventh Ministry of Engineering Industry and became better known as the Beijing Institute of Spacecraft Systems Engineering) with the Chinese Academy of Sciences, to form the Chinese Academy of Space Technology to build China's first satellite.

On 27 October 1966 the Chinese launched the Dong Feng 2A with a live nuclear warhead, which exploded in the Xinjiang Desert. The Dong Feng 3 (known in the West as CSS-2), with a cluster of four engines using storable propellants, made its first flight on 26 December 1966. Dong Feng 4 made its first flight on 30 January 1970. The addition of a small upper stage to the Dong Feng 4 resulted in the Chang Zheng 1 (Long March, LM-1) with its first launch occurring on 24 April 1970, carrying the Dong Fang Hong (East is Red) satellite into LEO. This launch demonstrated China's capability to develop its own rocket technology using only Chinese resources. The first and second stages were propelled by nitric acid/UDMH liquid engines. The

Vehicle History

Courtesy China Great Wall Industry Corporation.
A Long March 3B is prepared for launch.

third stage was powered by a solid motor. The satellite was a spherical polyhedron 1 m (39 in.) in diameter made of fiberglass wound on aluminum alloy. It carried a tape recorder and radio, transmitting the anthem, "The East is Red." The repeating 1-min transmission consisted of two sequential 20-s broadcasts of "The East is Red," a 5-s gap, 10 s of telemetry, and a 5-s gap. This broadcast was transmitted for 28 days until the batteries died, but the satellite remains in orbit to this day.

Development of Long March 2 (LM-2) started in 1970, with its first launch occurring 5 November 1974. After taking off, the vehicle lost its attitude stability, and the vehicle was automatically destroyed by the vehicle's airborne range safety device. The result of analysis showed that this failure was caused by a broken wire necessary for delivering the pitch rate gyro signal. After thorough investigation and research, the control system design was modified to improve its reliability. Quality control during production and functional checkout was improved. Meanwhile, payload capability was increased and the modified launch vehicle was designated as LM-2C. The first LM-2C was launched on 26 November 1975. The flight was a complete success and China's first recoverable satellite, Fanhui Shi Weixing (recoverable experimental satellite), was launched into orbit. The purpose of this satellite has never been reported. While it likely started as a military photoreconnaissance satellite, later versions have been used to conduct microgravity experiments in orbit. The military value from orbiting a reconnaissance satellite for a week once a year is limited. The Long March 2C has gone on to become the most reliable and frequently used rocket of the Chinese launch vehicle fleet.

The Shanghai Second Bureau of Machinery and Electrical Equipment was given the responsibility of developing a space launch vehicle, which became known as the Feng Bao 1 (Storm), and its first flight took place on 18 September 1973. The FB-1 had performance slightly less than the Long March 2C, and was much less reliable, which, along with low availability, resulted in it being phased out in 1981 after demonstrating a multiple manifesting capability. It is believed that this launch vehicle was being developed by the Chinese military to launch satellites having military purposes. Little was known about either the vehicle or the satellites that it launched.

The Shanghai work force was transferred to work on developing the Long March 3 (LM-3). The Long March 3 is a three-stage launch vehicle. It is mainly used for GTO missions. The first and second stages of LM-3 were based on the LM-2C. The new third stage was propelled by a LOX and LH_2 engine with restart capability. Meanwhile, a new launch site was developed closer to the equator, better suited for GTO launches. Xichang Satellite Launch Complex (XSLC) is located in a hilly, inhabited area, at 28.25° N (similar in latitude to Cape Canaveral). The first tests of mockup launchers began in March 1982, and on 29 January 1984 the first test flight of LM-3 was carried out. The payload was an experimental communications satellite.

To more readily reach polar, and in particular, sun-synchronous orbits, the Chinese developed a launch site at a former missile base, near the industrial coal town of Taiyuan. The Taiyuan Satellite Launch Complex (TSLC) was first used by another rocket derivative, Long March 4 (LM-4), on 6 September 1988 to launch China's first weather satellite, Feng Yun 1 (Wind and Cloud) into SSO. The LM-4 is based on the LM-2C with the addition of a storable propellant third stage using the engine of the LM-1 second stage. For Motorola, the Chinese have developed a smart dispenser to permit the LM-2C to launch two Iridium satellites at a time. TSLC has been the site of many Iridium operations and maintenance LM-2C/SD launches.

In 1988, Long March won commercial contracts to launch one Asiasat and two Australian Aussat satellites (renamed Optus by the time they were launched). In April 1990, China launched its first commercial satellite, Asiasat, aboard a LM-3. Based on the technology and flight experience of LM-2C and LM-3, a more powerful commercial launch vehicle, Long March 2E (LM-2E) was developed. With the addition of a perigee kick motor (PKM) the LM-2E is used for GTO missions. The first launch occurred on 16 July 1990, 18 months after development began. The LM-2E became the workhorse for launching commercial satellites during the early 1990s, successfully launching four commercial telecommunication satellites to GEO. However, two Hughes satellites did not make it to orbit.

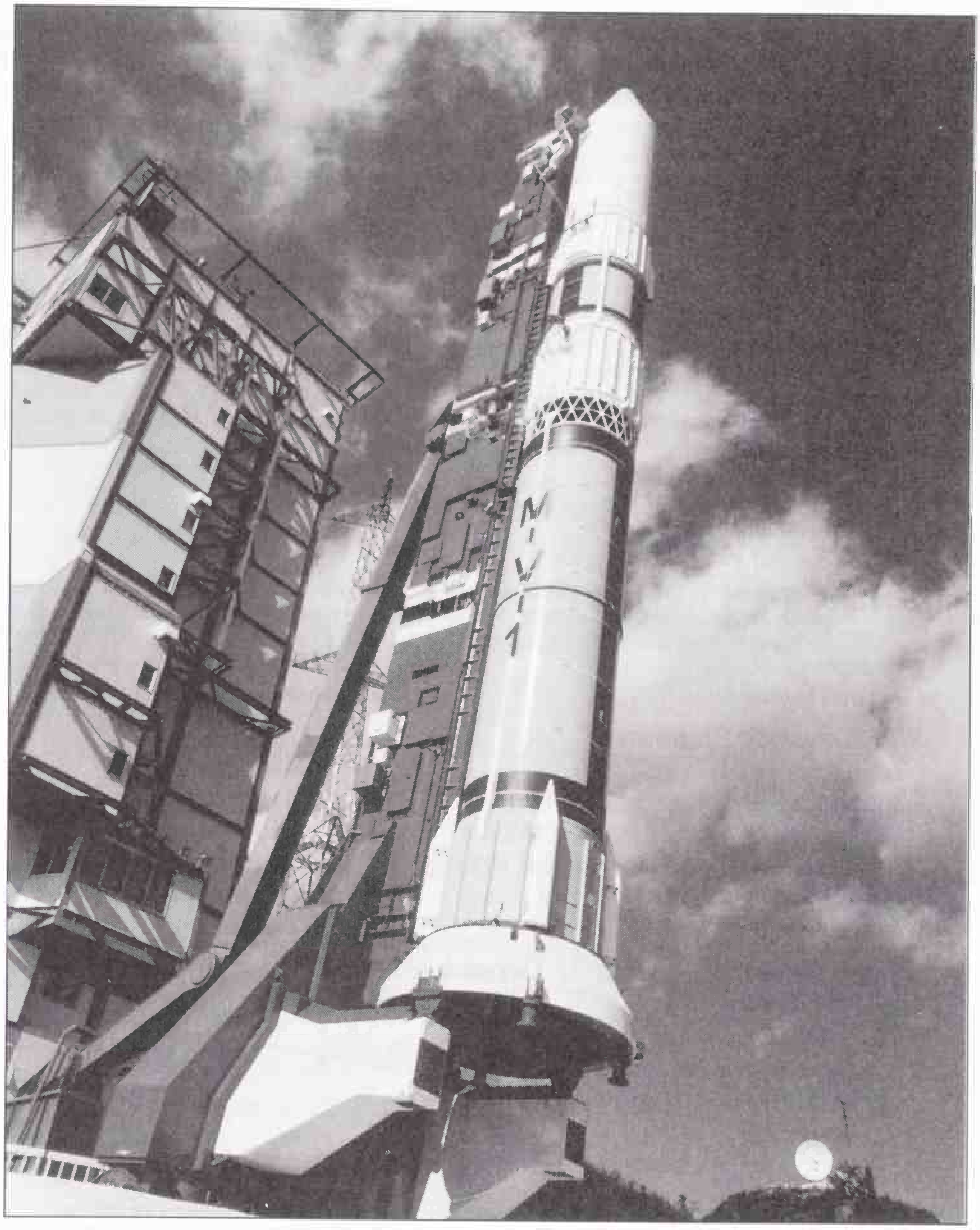
After the introduction of LM-2E, China proceeded with the development of an improved vehicle, LM-3B, with a cryogenic upper stage for about 50% greater performance to GTO than a LM-2E with a PKM. LM-3B became available in 1996. Two seconds after lifting off on 14 February 1996 carrying the Loral-built Intelsat 708 to a GTO orbit, the launch vehicle pitched over, impacting, and exploding near a village close to the launch site. At least six people were killed, though some eyewitness accounts claim a much greater death toll. A run of failures cast a pall on the reliability and credibility of the Chinese commercial launch program.

Considerable controversy has surrounded the accident investigations of the Hughes and Loral failures. This controversy has entangled Hughes, Loral, the international insurance industry, and various branches and departments of the U.S. government. It has resulted in congressional investigations into accusations of illegal technology transfer. On 25 May 1999, a declassified version of the "Report of the Select Committee on U.S. National Security and Military/Commercial Concerns with the People's Republic of China" (referred to as the Cox report) was released, accusing U.S. companies of compromising U.S. national security while using Long March launch vehicles to deliver their spacecraft into orbit. Hughes, Loral, and Motorola deny any intent of wrongdoing or transfer of technology to the Chinese. The report's recommendation for the tightening of export controls for U.S. satellite and launch technology, and the ramifications of this policy change, have caused concerns in the satellite and launch industry worldwide.

As a result of the U.S. crackdown on satellite exports to China, Long March has been sidelined in the commercial launch market. Although improved reliability has attracted several customers, no commercial launches have actually occurred since 1999.

The most significant activity in China's space program in recent years has been the development of a human spaceflight program. In 1999 China unveiled the LM-2F launch vehicle and the Shenzhou spacecraft. Shenzhou is loosely based on the design approach used on the Russian Soyuz, but is a larger and more capable spacecraft. A series of four unmanned test flights between 1999 and 2002 have demonstrated that China has the capability to launch yuhanguan (astronauts). The first flight carrying a passenge, Lt. Col. Yang Liwei, lifted off 15 October 2003. This made China only the third nation with the capability to launch people into space.

M-V



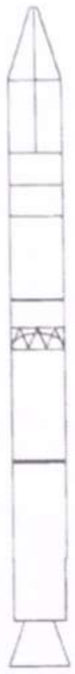
Courtesy JAXA.

The M-V is a small solid-propellant launch vehicle built and operated by Japan's Institute of Space and Astronautical Science. It is used to launch small science spacecraft to LEO or on interplanetary trajectories.

Contact Information

Program Information:
Japanese Aerospace Exploration Agency
3-1-1 Yoshinodai
Sagamihara, Kanagawa 229-8510
Japan
Phone: (0427) 513911
Web site: www.isas.ac.jp/info/rockets-e.html

M-V



M-V

General Description

Summary

The M-V is the sixth generation in the family of Mu vehicles. The Mu series are small, all solid-propellant launch vehicles originally used by Japan's space science agency, the Institute of Space and Astronautical Science (ISAS), now JAXA*, to launch astronomy satellites and planetary missions. The M-V has 2.4 times more performance than the M-3SII launch vehicle it replaced.

Status

Operational. First launch in 1997.

Origin

Japan

Key Organizations

Marketing Organization	Not marketed
Launch Service Provider	JAXA
Prime Contractor	IHI Aerospace

Primary Missions

Small spacecraft to LEO or Earth-escape trajectories

Estimated Launch Price

7 billion yen ($57 million) (ISAS, 2002)

Spaceport

Launch Site	Mu Launch Complex, Kagoshima, Japan
Location	31.2° N, 131.1° E
Available Inclinations	31–85 deg

Performance Summary

200 km (108 nmi), 31 deg	1900 kg (4200 lbm)
200 km (108 nmi), 90 deg	1370 kg (3000 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	1220 kg (2700 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	960 kg (2100 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 31 deg	1280 kg (2800 lbm) with optional kick motor
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	4
Launch Vehicle Successes	3
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	1

Flight Rate

0–1 per year

*JAXA, the Japanese Aerospace Exploration Agency, was formed 1 October 2003 by the merger of the Institute of Space and Astronautical Science (ISAS), National Aerospace Laboratory (NAL), and National Space Development Agency of Japan (NASDA). References to historical events will preserve the original organization's name. All ongoing and new references will be to JAXA.

NOMENCLATURE

The M-V is the current generation of the Mu launch vehicle series. It replaced the previous version, the M-3SII.

COST

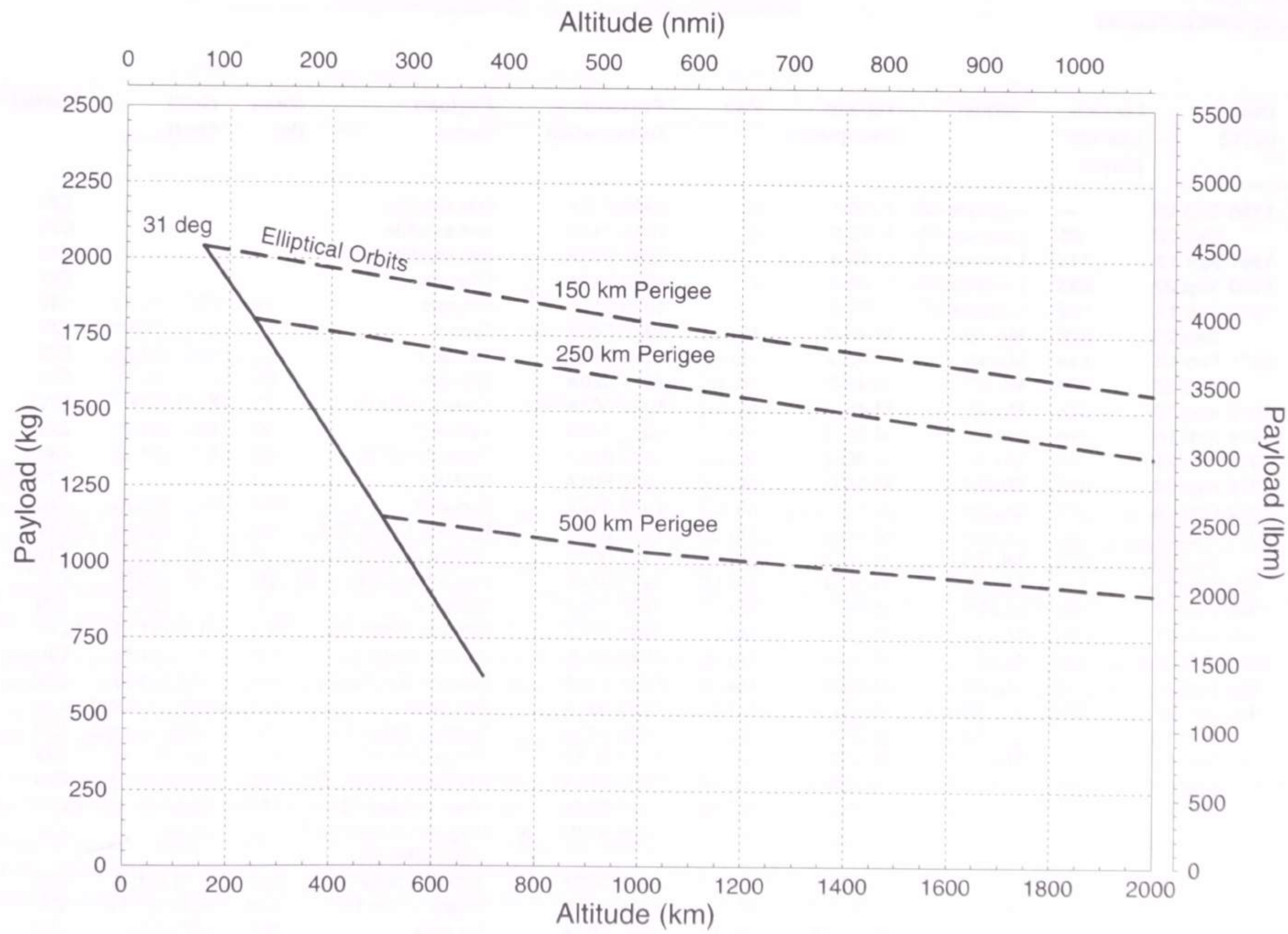
The M-V's total development cost was expected to be about 24.2 billion yen ($197 million) consisting of 5.5 billion yen ($45 million) for the first vehicle, 13.1 billion yen ($106 million) for development, and 5.6 billion yen ($45 million) for ground facility requirements. A program to upgrade the second stage motor cost 1.5 bilion yen ($12 million) in the late 1990s. According to the ISAS website in 2002, the per-flight cost of the M-V is about 7 billion yen ($57 million).

AVAILABILITY

The M-V has been operational since 1997. The planned flight rate is one per year, although this varies depending on readiness of payloads. The M-V is not available commercially and is used only to launch science spacecraft for the Japanese government. The long term future of M-V is uncertain. Japan's three space agencies merged in October 2003, and government funding for the M-V will be reduced as the H-IIA will become the rocket of choice for government payloads. However, the M-V program could continue to be funded by industry. The M-V may continue to be used at least until 2007–2008 when it is scheduled to launch the Planet-C Venus Climate Orbiter mission.

PERFORMANCE

The M-V can reach inclinations from 31 deg to approximately 90 deg from the launch site at Kagoshima. Because of range safety restrictions, the M-V cannot reach high inclination orbits by flying south directly. Instead it must fly southeast and then perform a dogleg maneuver toward the south. This results in decreased polar orbit performance. In addition to the LEO performance shown below, the M-V has a capability of 300–400 kg (660–880 lbm) for planetary missions using an optional fourth stage.

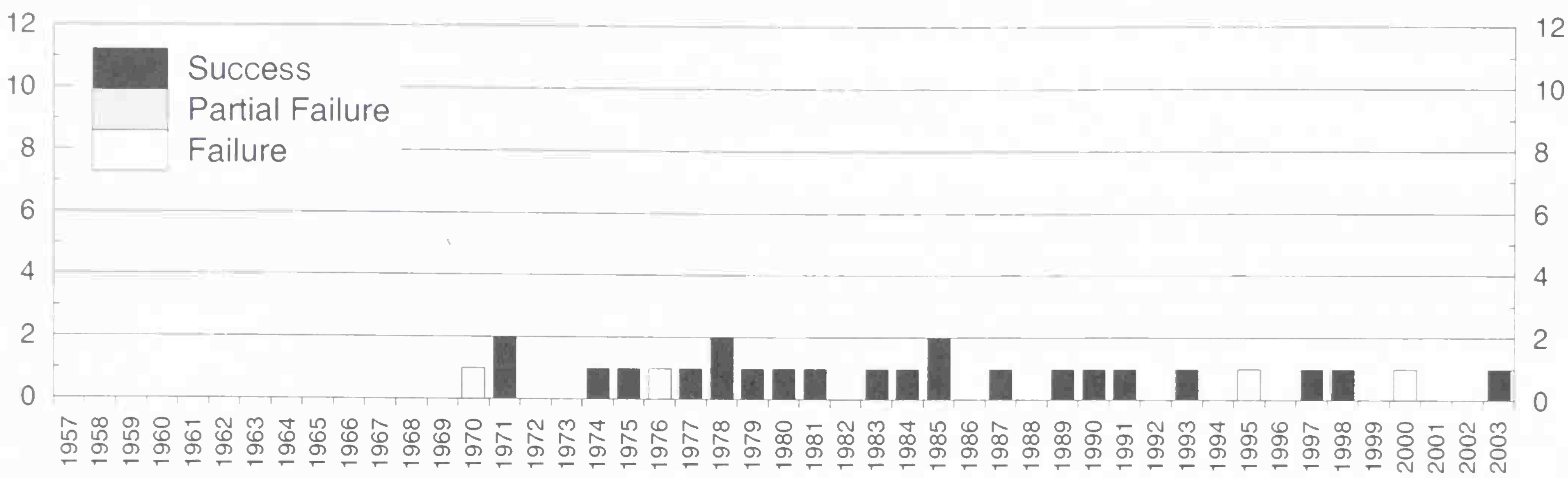


M-V: LEO Performance from Kagoshima

Flight History

Launches of Japan's first orbital launch vehicle family, the Lambda 4S, which predated the Mu series, are included in the table below for reference. They are not included in the Mu series launch totals or graph.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	M-V	Combined Mu series
Total Orbital Flights	4	27
Launch Vehicle Successes	3	23
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	1	4

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T,F		**1966 Sep 26**	**—**	**Lambda-4S**	**L-4S-1**	**L**	1966 F10A		test satellite			CIV	Japan
T,F		**Dec 20**	**85**	**Lambda-4S**	**L-4S-2**	**L**	1966 F13A		test satellite			CIV	Japan
T,F		**1967 Apr 13**	**114**	**Lambda-4S**	**L-4S-3**	**L**	1967 F03A		test satellite			CIV	Japan
F		**1969 Sep 22**	**893**	**Lambda-4S**	**L-4S-4**	**L**	1969 F13A		Ohsumi			CIV	Japan
		1970 Feb 11	142	Lambda-4S	L-4S-5	L	1970 011A		Ohsumi	12	EEO (31.1)	CIV	Japan
F	**1**	**Sep 25**	**226**	**Mu-4S**	**M-4S-1**	**Mu LC**	1970 F08A		Tansei			CIV	Japan
	2	1971 Feb 16	144	Mu-4S	M-4S-2	Mu LC	1971 011A		Tansei 1	62	LEO (29.7)	CIV	Japan
	3	Sep 28	224	Mu-4S	M-4S-3	Mu LC	1971 080A		Shinsei	65	LEO (32.1)	CIV	Japan
	4	1972 Aug 19	326	Mu-4S	M-4S-4	Mu LC	1972 064A		Denpa (REXS)	75	EEO (31)	CIV	Japan
	5	1974 Feb 16	546	Mu-3C	M-3C-1	Mu LC	1974 008A		Tansei 2	65	EEO (31.2)	CIV	Japan
	6	1975 Feb 24	373	Mu-3C	M-3C-2	Mu LC	1975 014A		Taiyo (SRATS)	86	EEO (31.5)	CIV	Japan
F	**7**	**1976 Feb 04**	**345**	**Mu-3C**	**M-3C-3**	**Mu LC**	1976 F01A		CORSA	92		CIV	Japan
	8	1977 Feb 19	381	Mu-3H	M-3H-1	Mu LC	1977 012A		Tansei 3	230	EEO (65.8)	CIV	Japan
	9	1978 Feb 04	350	Mu-3H	M-3H-2	Mu LC	1978 014A		Kyokko 1 (EXOS A)	103	EEO (65.4)	CIV	Japan
	10	Sep 16	224	Mu-3H	M-3H-3	Mu LC	1978 087A		Jikiken (EXOS B)	100	EEO (31.2)	CIV	Japan
	11	1979 Feb 21	158	Mu-3C	M-3C-4	Mu LC	1979 014A		Hakucho (CORSA B)	100	LEO (29.9)	CIV	Japan
	12	1980 Feb 17	361	Mu-3S	M-3S-1	Mu LC	1980 015A		Tansei 4	134	LEO (38.7)	CIV	Japan
	13	1981 Feb 21	370	Mu-3S	M-3S-2	Mu LC	1981 017A		Hinotori (Astro A)	185	LEO (31.3)	CIV	Japan
	14	1983 Feb 20	729	Mu-3S	M-3S-3	Mu LC	1983 011A		Tenma (Astro B)	185	LEO (31.5)	CIV	Japan
	15	1984 Feb 14	359	Mu-3S	M-3S-4	Mu LC	1984 015A		Ohzora (EXOS C)	180	LEO (74.6)	CIV	Japan
	16	1985 Jan 07	328	Mu-3S-2	M-3S2-1	Mu LC	1985 001A		Sakigake	141	Halley Comet	CIV	Japan
	17	Aug 18	223	Mu-3S-2	M-3S2-2	Mu LC	1985 073A		Suisei (Planet A)	141	Halley Comet	CIV	Japan
	18	1987 Feb 05	536	Mu-3S-2	M-3S2-3	Mu LC	1987 012A		Ginga (Astro C)	420	LEO (31.1)	CIV	Japan
	19	1989 Feb 21	747	Mu-3S-2	M-3S2-4	Mu LC	1989 016A		Akebono (EXOS D)	295	EEO (75.1)	CIV	Japan
	20	1990 Jan 24	337	Mu-3S-2	M-3S2-5	Mu LC	1990 007A		Hiten (Muses A)	185	Moon	CIV	Japan
				Mu-3S-2	M-3S2-5	Mu LC	1990 007B	A	Hagoromo (Muses A subsatellite)	12	Moon	CIV	Japan
	21	1991 Aug 30	583	Mu-3S-2	M-3S2-6	Mu LC	1991 062A		Yohkoh (Solar A)	420	LEO (31.3)	CIV	Japan
	22	1993 Feb 20	540	Mu-3S-2	M-3S2-7	Mu LC	1993 011A		Asuka (Astro D)	420	LEO (31.1)	CIV	Japan
F	**23**	**1995 Jan 15**	**694**	**Mu-3S-2**	**M-3S2-8**	**Mu LC**	1995 F01A		Express 1	765	LEO (31)	CIV	Germany
	24	1997 Feb 12	759	M-5	M-V-1	Mu LC	1997 005A		Haruka (Muses B)	800	EEO (31.4)	CIV	Japan
	25	1998 Jul 03	506	M-5	M-V-3	Mu LC	1998 041A		Nozomi (Planet B)	540	Mars	CIV	Japan
F	**26**	**2000 Feb 10**	**587**	**M-5**	**M-V-4**	**Mu LC**	2000 F01A		ASTRO E	1650	LEO	CIV	Japan
	27	2003 May 09	1184	M-5	M-V-5	M-V	2003 019A		Muses C (Haybusa)	530	Asteroid	CIV	Japan

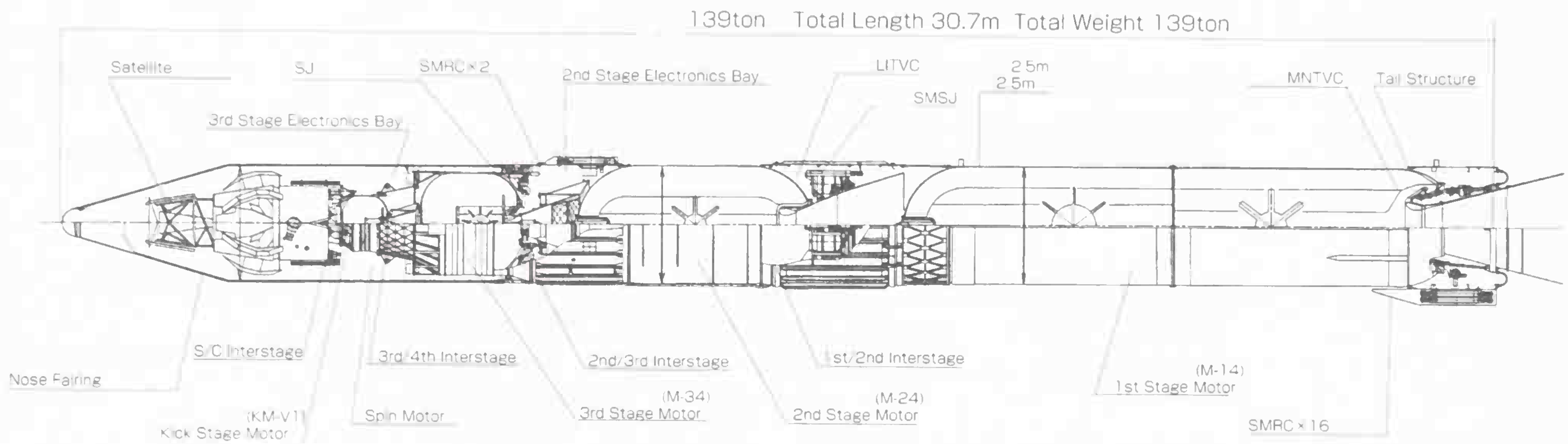
Mu LC is the Mu Launch Complex at Kagoshima Space Center, Kagoshima, Japan
T = Test Flight, **F** = Failure, **P** = Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly
Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Flight History

Failure Descriptions:

F	1966 Sep 26	L-4S-1	1996 F10	Failure to orbit.
F	1966 Dec 20	L-4S-2	1966 F13	Fourth stage failed to ignite.
F	1967 Apr 13	L-4S-3	1967 F03	Fourth stage failure.
F	1969 Sep 22	L-4S-4	1969 F13	Fourth-stage attitude control system failure.
F	1970 Sep 25	M-4S-1	1970 F08	Fourth stage failed to ignite.
F	1976 Feb 04	M-3C-3	1976 F01	Second-stage thrust vector control failure.
F	1995 Jan 15	M-3S2-8	1995 F01	The second stage attitude control system began an oscillation at T+103 that caused the vehicle to veer off course around T+140 s. The payload, a recoverable experiment satellite, entered an unstable 120-km orbit that lasted only two and a half revolutions and was discovered months later in Ghana. The failure was attributed to improper modeling of vehicle dynamics stemming from the much heavier than normal satellite.
F	2000 Feb 10	M-V-4	2000 F01A	Onboard camera showed chunks of the graphite nozzle throat insert being ejected at T+25, T+38 and T+41 s. Hot gases leaked through the nozzle damaging the control system, causing pitch control to be lost at T+53 s and yaw control to fail at T+69 s. Subsequent stages operated normally but could not make up for the performance shortfall, leaving the vehicle in an unstable orbit with a perigee of 80 km.

Vehicle Design



Courtesy JAXA.

	M-V
Height	30.8 m (101 ft)
Gross Liftoff Mass	137.5 t (303 klbm)
Thrust at Liftoff	4220 kN (950 klbf)

Stages

The first stage is composed of an M-14 motor, a tail shroud, and an interstage joint. The M-14 solid motor is made of two segments. The case is made of improved HT-230 maraging steel. The motor burn profile is designed for a relatively short, high-thrust burn to reduce gravity losses. However, it also has a very long tailoff to provide sufficient thrust to maintain attitude control while the vehicle gains altitude to perform staging at a lower dynamic pressure. Stage separation is initiated by the ignition of flexible linear shaped charges (FLSCs), which cut the interstage structure into three separating pieces. The second stage then ignites for the fire-in-the-hole hot separation.

The second stage M-24 motor has been replaced by the new M-25 motor, which flew for the first time in May 2003. The steel case of the M-24 motor was replaced with a carbon–fiber composite case. This significantly reduced the cost of the second stage and reduced its weight by about 800 kg (1750 lbm). In addition, increased strength of the case allowed the motor chamber pressure to be doubled, increasing specific impulse. The previous LITVC system was replaced with a less-expensive movable nozzle system similar to that used on the first and third stages.

The third stage is composed of a nearly spherical M-34 motor, an instrument section, and a payload adapter. The M-34 motor nozzle is a gimbaled, conical-bell design with a simple collapsible cup-type extendible exit cone system.

The first and second stages use maraging steel cases and reinforced plastic nozzles. The third stage uses a carbon fiber–epoxy filament-wound case and a reinforced plastic nozzle. All stages have a star perforated-propellant grain internal burning configuration. Their grains are cast-in-case aluminized composite propellants with an HTPB binder. Their igniters are a flame producing rocket motor type with dual squib initiators.

Vehicle Design

	Stage 1	Stage 2	Stage 3	Stage 4 (Optional)
Dimensions				
Length	13.8 m (45.3 ft)	6.6 m (21.7 ft)	3.6 m (11.8 ft)	1.5 m (4.9 ft)
Diameter	2.5 m (8.2 ft)	2.5 m (8.2 ft)	2.2 m (7.2 ft)	1.2 m (3.9 ft)
Mass				
Propellant Mass (each)	71,490 kg (157,600 lbm)	33,000 kg (72,800 lbm)	10,000 kg (22,000 lbm)	1312 kg (2890 lbm)
Inert Mass (each)	12,070 kg (26,610 lbm)	3270 kg (7220 lbm)	1000 kg (2200 lbm)	118 kg (260 lbm)
Gross Mass (each)	83,560 kg (184,200 lbm)	36,270 kg (80,000 lbm)	11,000 kg (24,300 lbm)	1430 kg (3150 lbm)
Propellant Mass Fraction	0.85	0.91	0.91	0.92
Structure				
Type	Monocoque	Filament-wound monocoque	Monocoque	Monocoque
Material	HT-230 steel	CFRP	CFRP filament wound	CFRP filament wound
Propulsion				
Motor Designation	M-14 (IHI Aerospace)	M-25 (IHI Aerospace)	M-34 (IHI Aerospace)	KM-V1 (IHI Aerospace)
Number of Motors	1	1	1	1
Propellant	HTPB (BP-204J)	HTPB (BP-208J)	HTPB (BP-205J)	HTPB
Number of Segments	2	1	1	1
Average Thrust	Vacuum: 3780 kN (850 klbf)	1520 kN (340 klbf)	290 kN (65.2 klbf)	52 kN (11.7 klbf)
Isp	Vacuum: 274 s	292 s	301 s	298 s
Chamber Pressure	59 bar (853 psi)	118 bar (1706 psi)	59 bar (853 psi)	49 bar (710 psi)
Nozzle Expansion Ratio	11.0:1	31:1	96.0:1	89:1
Attitude Control				
Pitch, Yaw	Hydraulic nozzle gimbal ± 5 deg	Nozzle gimbal	Nozzle gimbal	Spin stabilized
Roll	Solid motors	Solid motors	Hydrazine thrusters	Hydrazine thrusters
Staging				
Nominal Burn Time	Web time: 49 s Action time: 85 s	62 s	102 s	73 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Burn to depletion
Stage Separation	FLSC/ Fire-in-the-hole	V-clamp/spring	V-clamp/spring	V-clamp/spring

Attitude Control System

The first stage uses the movable nozzle thrust vector control (MNTVC) system installed at the nozzle of an M-14 motor to control the pitch and yaw attitudes. The nozzle employs a flexible bearing to achieve the required thrust vector deflection of 5 deg. The two electrohydraulic actuators that control the nozzle are pressurized by a small solid-propellant gas-generator-driven turbopump. The roll attitude is controlled by small solid motors for roll control (SMRC) installed around the tail shroud.

The second-stage pitch and yaw attitudes are controlled by the MNTVC system of the M-25 motor. The roll attitude is controlled by the SMRC device installed in the upper interstage section. The pitch, yaw, and roll attitudes after the M-25 motor burns out are controlled by the solid motor side jet (SMSJ) device installed around the M-25 motor nozzle.

An electromechanically actuated MNTVC device controls the pitch and yaw attitudes for the third stage. A hydrazine-fueled side jet device is installed around the M-34 motor nozzle and controls the roll attitude during the motor firing and the pitch, yaw, and roll attitudes after motor burnout.

Avionics

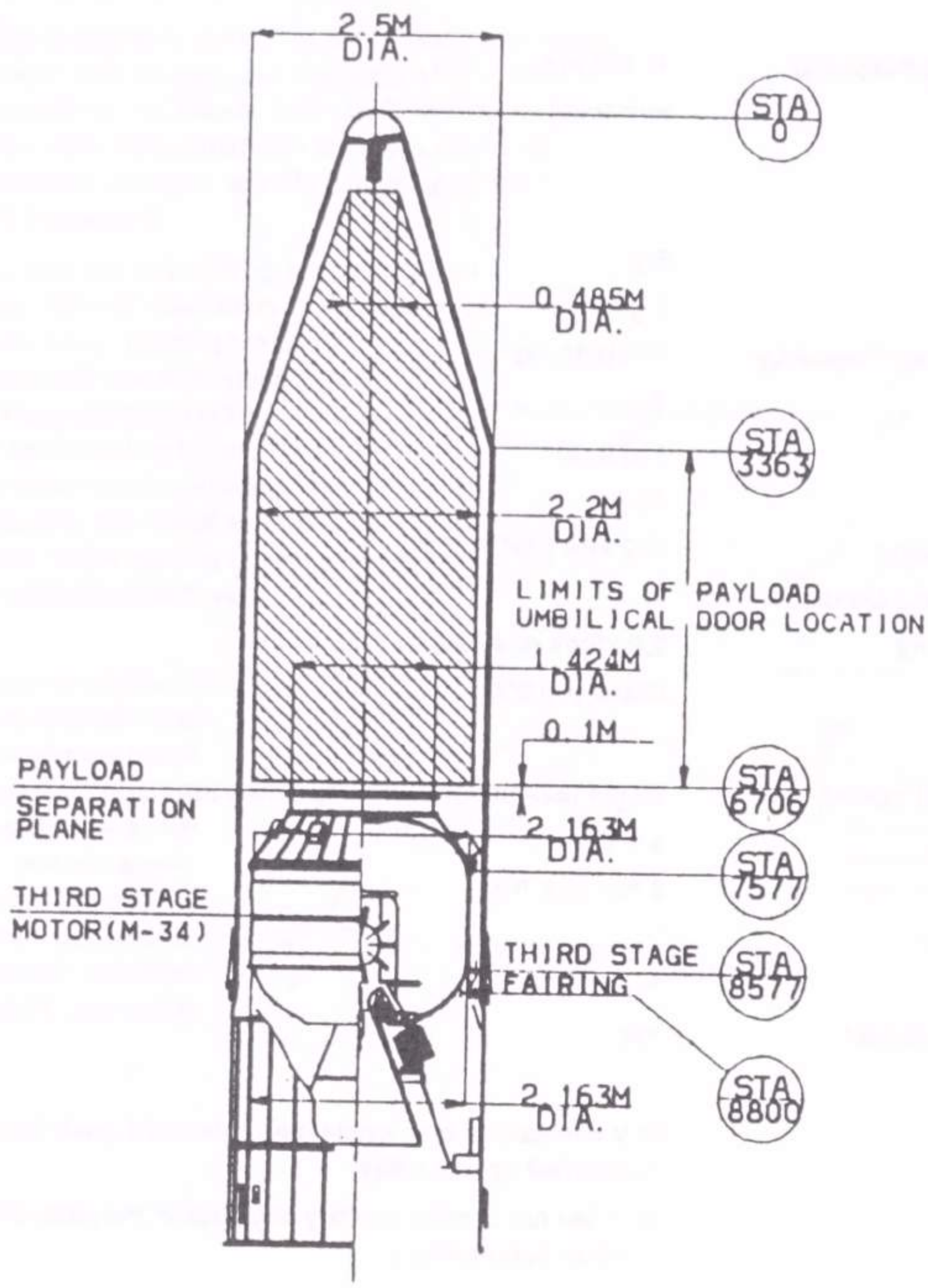
The M-V control system uses an inertial navigation system (INS) that includes strap-down fiber-optical gyros (FOGs), accelerometers, and computers on the third stage. Three rate gyros on the first stage provide damping signals for the first stage pitch, yaw, and roll control. The attitude and orbit determination and control calculations are carried out by an onboard microprocessing system. The primary guidance for orbit injection is done from the ground by a tracking radar and a real-time computer. The INS serves as a backup to the radio guidance system. The instrumentation system for the M-V vehicle is similar to that of the M-3SII series. The guidance and control system, S-band telemetry system, and C-band radar transponder are on the third stage. The vhf telemetry system is on the first and second stages. The L-band radar transponder is on the second stage. The C-band transponder system is also used for commanding of guidance and control.

Vehicle Design

Payload Fairing

The nose fairing is made of honeycomb sandwich shell with carbon–fiber reinforced plastic (CFRP) face sheets covered with cork for thermal protection. Its clam shell type jettison is initiated by an action of expandable shielded mild detonating cord (ESMDC) laid along the split lines. The fairing encloses both the payload and the third stage.

Dimensions in meters



Length	9.19 m (30.1 ft)
Primary Diameter	2.5 m (8.2 ft)
Mass	700 kg (1543 lbm)
Sections	2
Structure	Composite sandwich
Material	Honeycomb sandwich with CFRP face sheets

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	2200 mm (86.6 in.)
Maximum Cylinder Length	3473 mm (136.7 in.)
Maximum Cone Length	2563 mm (100.9 in.)
Payload Adapter Interface Diameter	1603 mm (63.1 in.)
Payload Integration	
Nominal Mission Schedule Begins	T–30 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T–120 min
On-Pad Storage Capability	Indefinite
Last Access to Payload	?
Environment	
Maximum Axial Load	5 *g*
Maximum Lateral Load	7 *g*
Minimum Lateral/Longitudinal Payload Frequency	15 Hz/15 Hz
Maximum Acoustic Level	?
Overall Sound Pressure Level	148.8 dB
Maximum Flight Shock	10 *g*
Maximum Dynamic Pressure on Fairing	142 kPa (2965 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	?
Maximum Pressure Change in Fairing	2.9 kPa/s (0.4 psi/s)
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±5 km (±2.4 nmi), ±0.01 deg inclination
Attitude Accuracy (3 sigma)	± 1 deg
Nominal Payload Separation Rate	2 m/s (6.6 ft/s)
Deployment Rotation Rate Available	?
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	M-V has carried only single payloads in the past. Contact JAXA for more information on comanifest opportunities.
Auxiliary Payloads	M-V has not carried auxiliary payloads in the past. Contact JAXA for more information on possible opportunities.

Production and Launch Operations

Launch Operations—Kagoshima Space Center

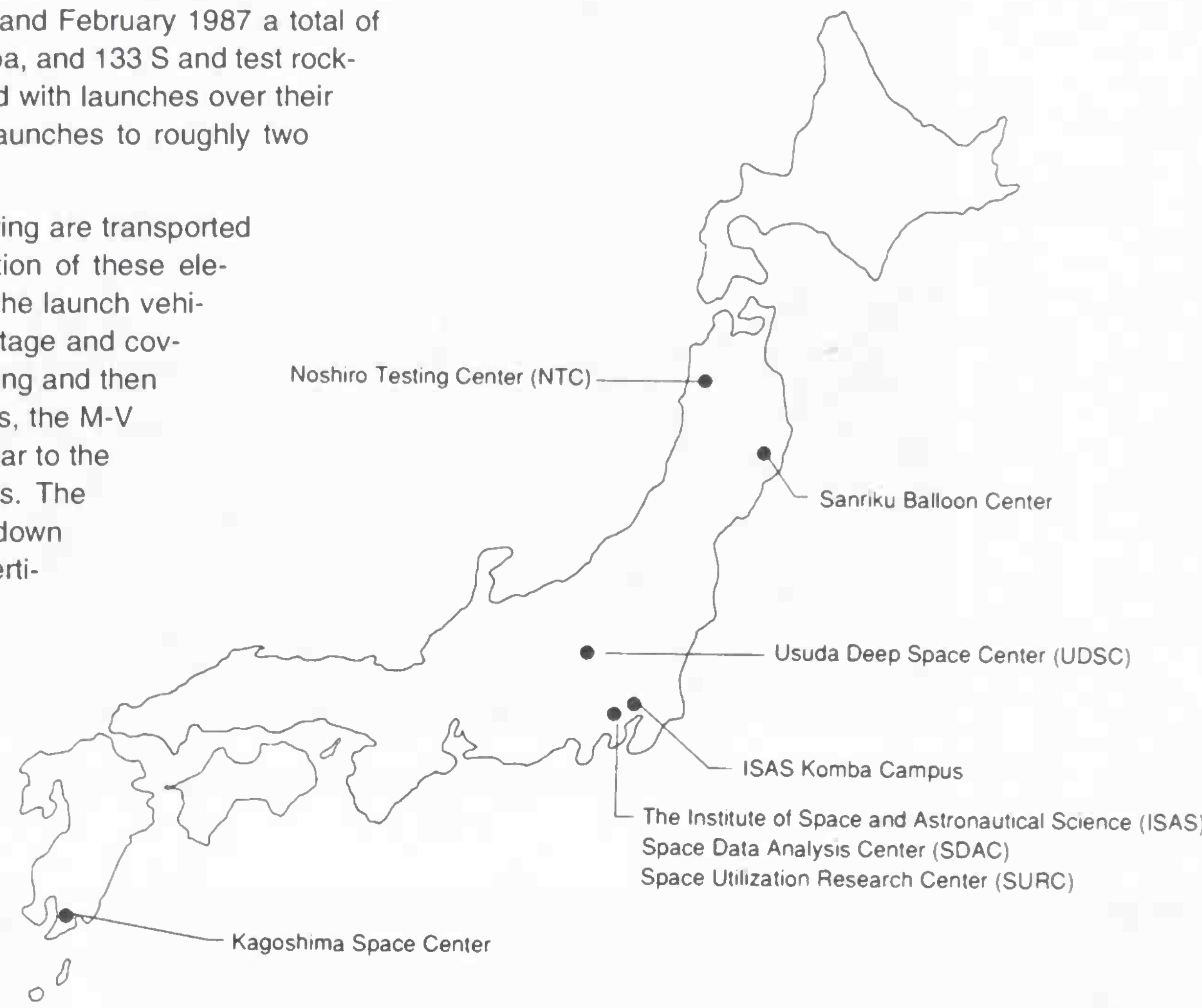
All Mu launches are conducted at the Kagoshima Space Center (KSC) located on the southern tip of Kyushu, the southernmost point in Japan's major island chain. Kagoshima is located in Uchinoura-cho, which lies on the east coast of Ohsumi Peninsula, Kagoshima Prefecture. Construction of a site for the launch of sounding rockets began in February 1962 with extensions for satellite launches by Mu-vehicles completed by 1966. Although launch dates are greatly restricted because of fishing rights, Kagoshima has remained available to ISAS for launching scientific satellites.

The land around KSC is mostly hilly. Various facilities, such as those for launching rockets; telemetry, tracking, and command stations for rockets and satellites; and optical observation posts, are placed in areas prepared by flattening the tops of several hills and connected to each other with roads. Buildings in the Mu Center have the total floor space of 12,750 m^2 (137,000 ft^2). The facilities used for Mu launch preparation include the Satellite Preparation Building, Mu Assembly Building, and Mu Launch Complex.

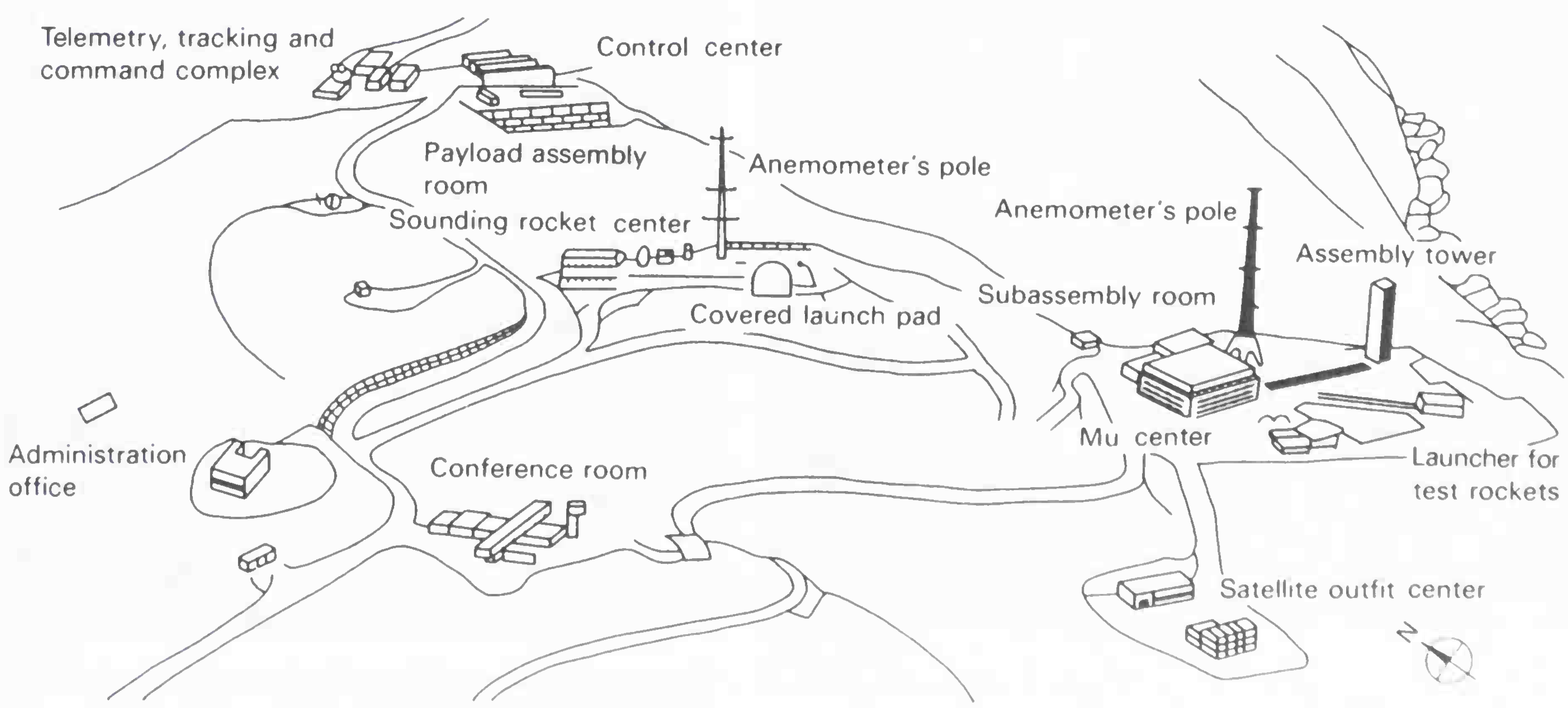
In addition to orbital launches, KSC also supports a number of suborbital rocket types. Between 1962, when the temporary use of KSC began, and February 1987 a total of 293 rockets were launched: 19 Mu, 25 Lambda, 116 Kappa, and 133 S and test rockets. Objections by local fishermen to the noise associated with launches over their fishing areas have resulted in the restriction of space launches to roughly two months a year (generally February and September).

Vehicle elements such as stage motors and the nose fairing are transported separately to the launch site by trucks. Vehicle integration of these elements is carried out inside the assembly tower, building the launch vehicle vertically. The spacecraft is integrated with the third stage and covered by the nose fairing in advance at the assembly building and then transported to the tower. Like all other Mu launch vehicles, the M-V is attached along one side to a vertical launcher rail, similar to the launchers used for some sounding rockets and missiles. The launcher can be set at a specific elevation angle pointing down range. As a result, the M-V does not normally launch vertically, but at an angle.

It is normal at KSC to integrate the vehicle onsite two months before launch using a dummy spacecraft. This takes about three weeks. Another two weeks are spent just before launch to bring the spacecraft aboard the launch vehicle and check out the whole system—launch vehicle and spacecraft. Although this procedure provides some margin in the schedule, these two phases can be merged into one if the spacecraft can be prepared in time. In this case, the total launch operation would take four weeks with 150–250 personnel involved.

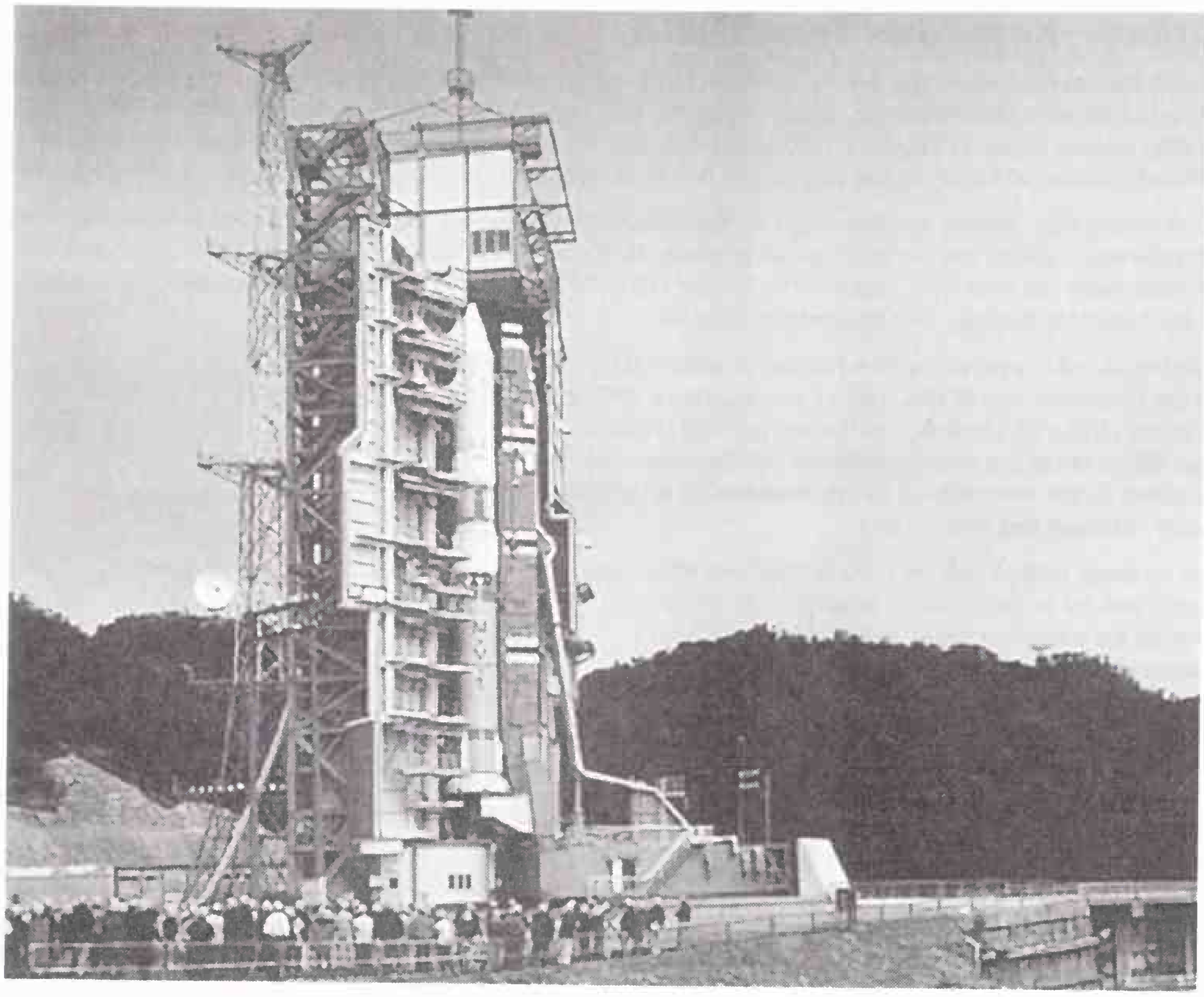


Launch Facilities



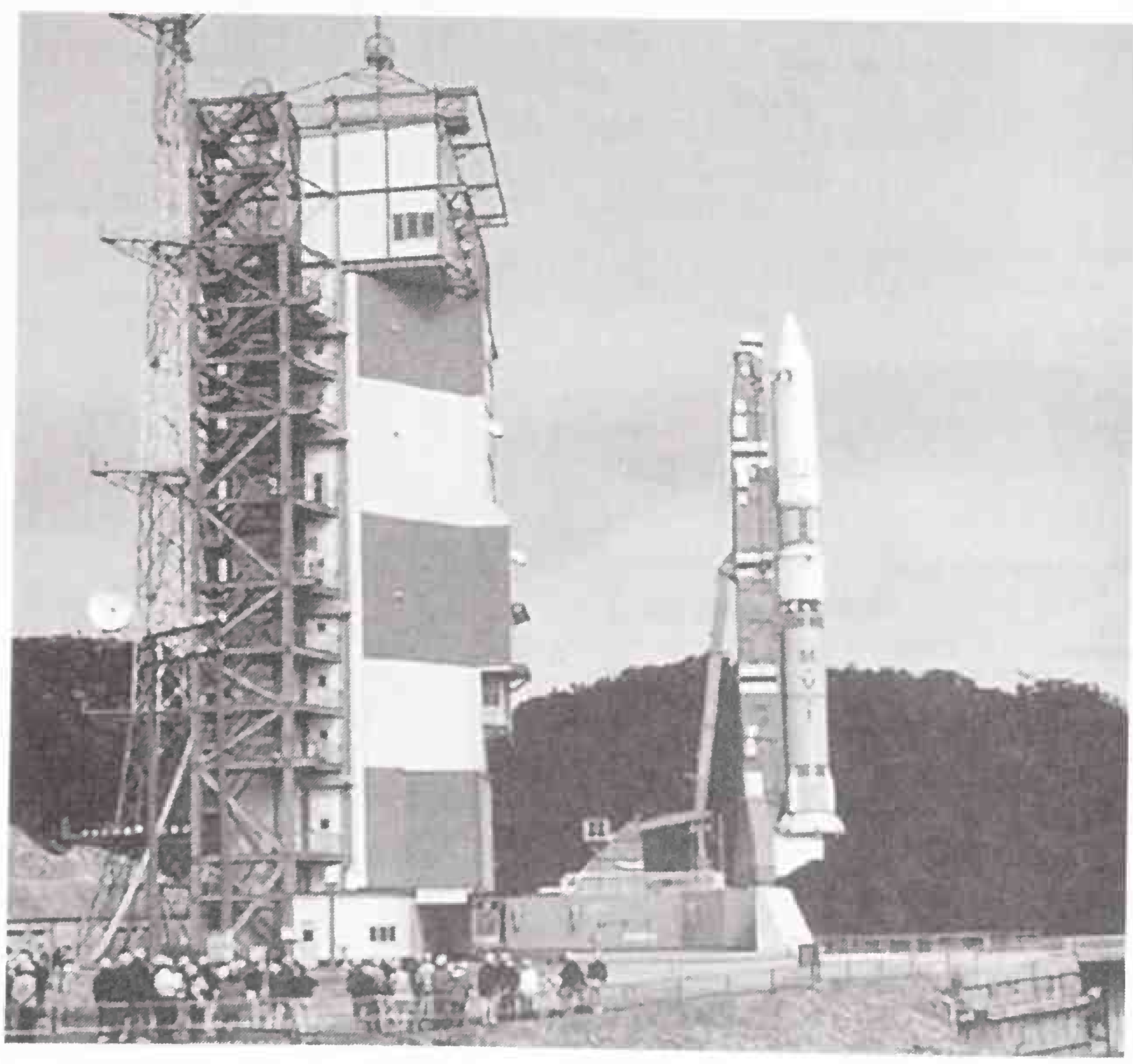
Kagoshima Space Center Facilities Layout

Production and Launch Operations



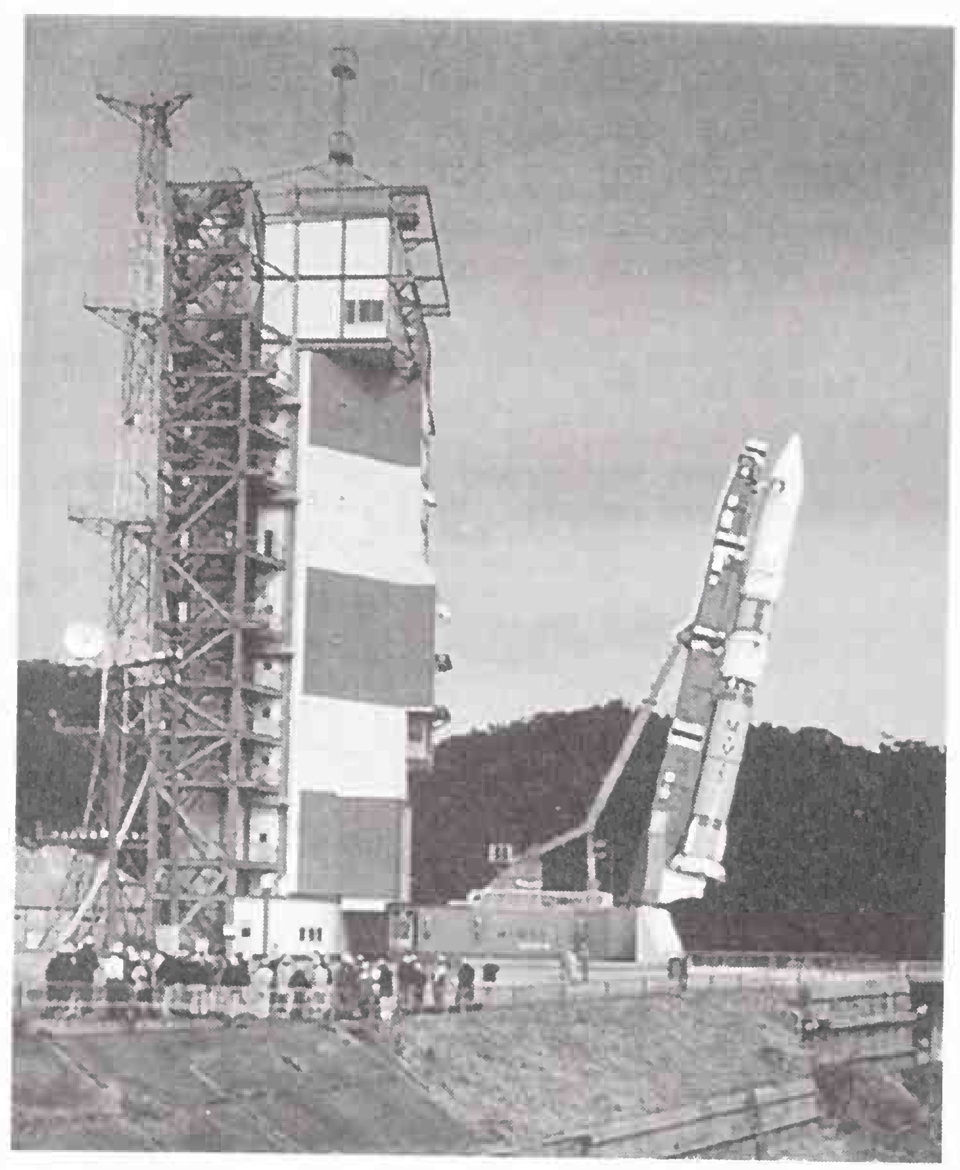
Courtesy JAXA.

The M-V's unique launch process begins when it is assembled vertically in the stationary assembly tower.



Courtesy JAXA.

The rotating launch mount swivels the M-V out of the tower into the launch position over an exhaust deflector.



Courtesy JAXA.

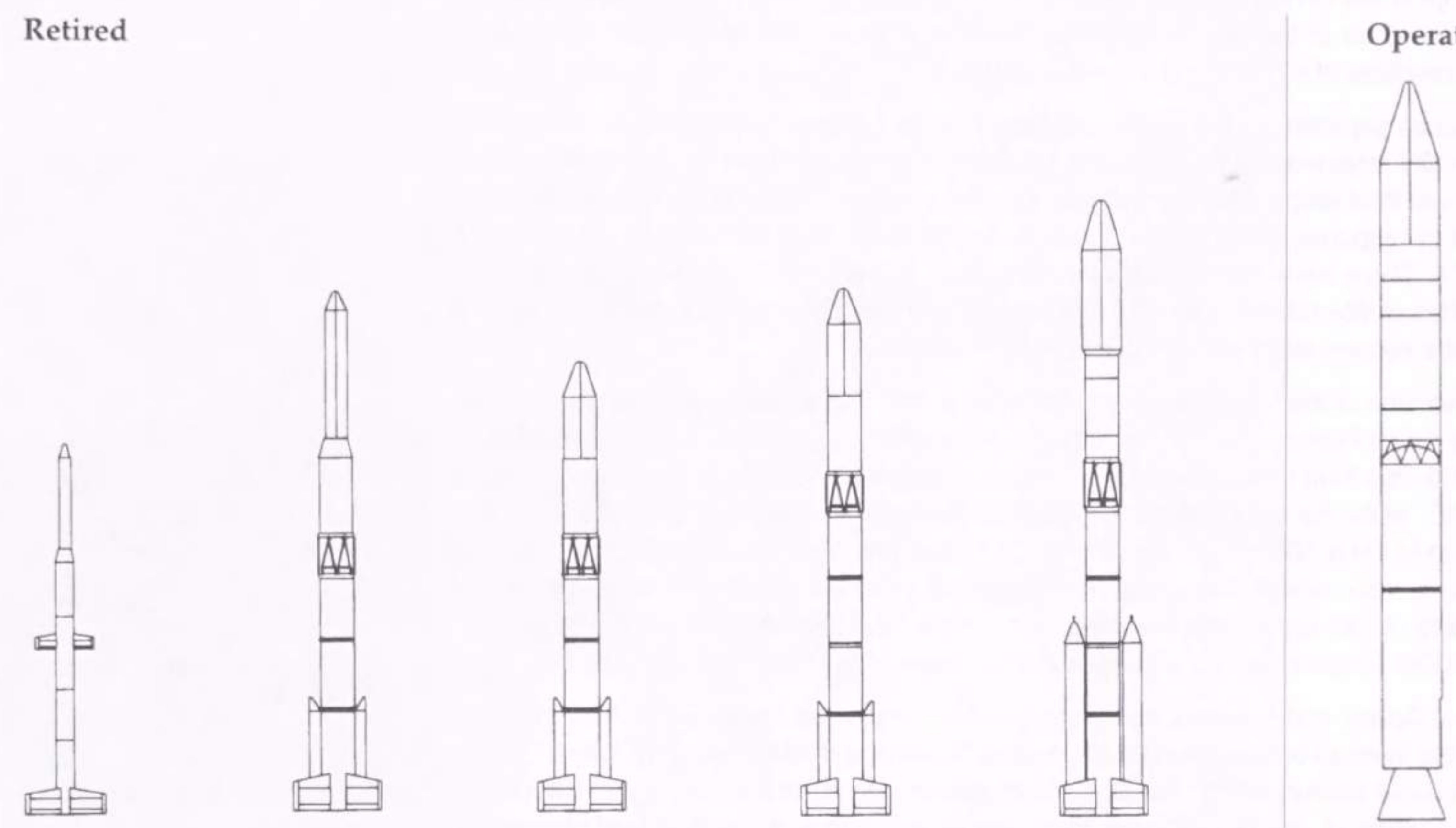
The rocket is positioned over the exhaust deflector and lowered to the desired elevation angle pointing downrange for launch.

Vehicle Upgrade Plans

Future plans for the M-V focus on efforts to reduce the cost of the vehicle, in order to keep the program alive in the face of competition from the H-IIA. IHI Aerospace is working to reduce the cost of the basic M-V configuration by 35–50%, possibly by replacing the first-stage motor with the SRB-A from the H-IIA program and by upgrading the third stage. Another possibility is to develop a smaller derivative of M-V, dubbed M-V Lite, which would consist of the upgraded M-V without the first stage. This smaller vehicle would cost roughly $35 million to develop, and as little as $13 million per launch. It could be developed around 2005 if funding is available.

Vehicle History

Vehicle Evolution



Vehicle	L-4S	M-4S	M-3C	M-3H/M-3S	M-3SII	M-V
Period of Service	1966–1970	1970–1972	1974–1979	1977–1984	1985–1995	1997–Present
Payload to 185 km (100 nmi) 31 deg	26 kg (57 lbm)	180 kg (400 lbm)	195 kg (430 lbm)	290 kg (640 lbm)	780 kg (1720 lbm)	1900 kg (4200 lbm)

Vehicle Description

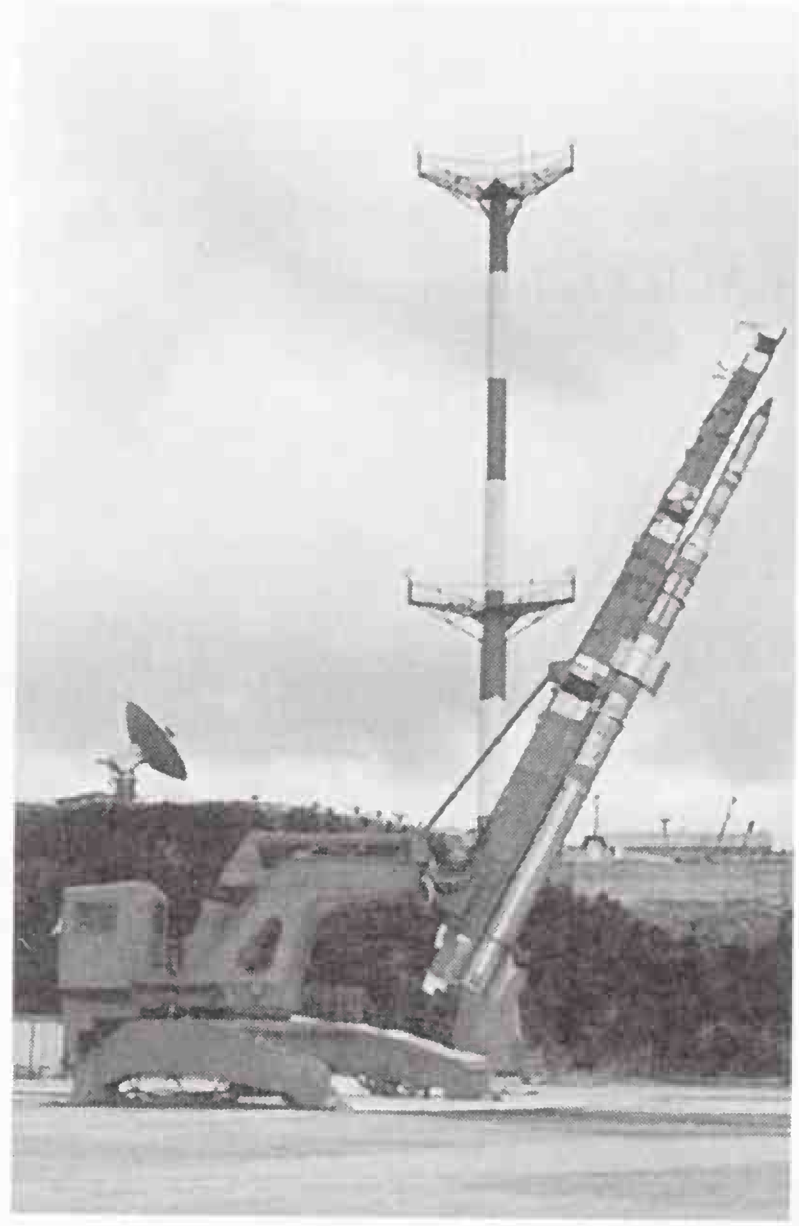
- **L-4S** This Lambda rocket was Japan's first rocket to reach orbit. Four-stage solid-propellant vehicle with the first three stages unguided and the fourth stage with attitude control.
- **M-4S** First member of the Mu family. Four-stage solid-propellant vehicle utilizing fins and spinning for attitude stabilization. Guidance provided by radio guidance.
- **M-3C** Same as M-4S except for an improved second stage, third-stage motor was replaced by an enlarged M-4S fourth stage, and the second stage used LITVC and hydrazine side jets for roll control.
- **M-3H** Same as M-3C except longer first stage, fairing was lengthened, and option existed for a fourth stage.
- **M-3S** Same as M-3H except added first stage LITVC and SMRC.
- **M-3SII** Same as M-3S except enlarged strap-on boosters with steerable nozzles, lengthened second stage, enlarged third stage, and wider and longer fairing.
- **M-V** New development with larger diameter stage motors and fairing, no strap-on boosters.

VEHICLE HISTORY

Historical Summary

In 1966, the Japanese Space Activities Council (SAC) proposed a long-range program concerning the launch and use of satellites. It urged that Japan develop the technology required to launch its own scientific and experimental applications satellites. Early attempts to orbit a spacecraft used a modified sounding rocket known as the L-4S. After four failures, the Japanese successfully orbited the 24-kg (52 lbm) Ohsumi satellite (named after the launch site location) on 11 February 1970 using the L-4S vehicle. Although the satellite carried little more than accelerometers and thermometers, the launch represented Japan's first successful orbital mission.

The success of the final L-4S launch gave the Japanese the confidence to proceed with their efforts. To orbit a larger payload, a new vehicle designed specifically as a satellite launcher was needed. The Institute of Space and Aeronautical Science at the University of Tokyo was given responsibility for the design, development, and operation of this new launcher designated Mu. The first vehicle in the Mu family of launchers was the M-4S. The M-4S was an all-solid-propellant vehicle that made use of aerodynamic and spin-stabilization control techniques. Responsibility for vehicle production and motor fabrication was assigned to the Nissan Motor Company, Ltd. Work was subcontracted by Nissan to other aerospace corporations in Japan. The first attempt to launch the complete vehicle was made in September 1970. The launch was unsuccessful because of the failure of the fourth stage to ignite. The M-4S later successfully orbited three satellites of 67–75 kg (150–165 lbm) mass.



Courtesy ISAS.

The fifth Lambda 4S rocket performed Japan's first successful space launch in 1970. With a gross weight of only 9399 kg (20,721 lbm), it was also the lightest launch vehicle in history.

To expand launch capabilities, the M-3C was designed as a second-generation launcher. The three-stage M-3C retained the first stage of the M-4S but incorporated an improved second stage and a new third stage. The second-stage motor used secondary fluid injection for thrust vector control in response to an onboard autopilot. The M-3C was successfully tested on 16 February 1974. There have been three other launches of the M-3C. Two were successful in orbiting satellites of 85–100 kg (190–220 lbm) mass, and the other launch failed because of a malfunction of a second-stage thrust-vector control system.

An improved version of the M-3C was designated the M-3H. This vehicle successfully launched a test satellite on 19 February 1977. Two other launches were successful, carrying satellites of 90–130 kg (200–290 lbm) mass. The second and third stages of this vehicle remained the same as on the M-3C, while the length of the first-stage core motor was increased by one-third. The capability of the M-3H is 50% larger than that of the M-3C. The next version of the M-series was the M-3S. This launch vehicle was similar to M-3H, except that the first stage had guidance and control capability. A 181-kg (400-lbm) satellite was orbited successfully during the first flight on 17 February 1980. Three more successful launches followed through 1984.

The Institute of Space and Astronautical Science (ISAS) was established on 14 April 1981, by reorganizing the Institute of Space and Aeronautical Science at the University of Tokyo. ISAS is one of the National Interuniversity Research Institutes belonging to the Ministry of Education, Science, and Culture of Japan. ISAS was in charge of research and development of scientific satellites and their launch vehicles (the Mu-vehicles), while the National Space Development Agency (NASDA) was in charge of development of application satellites and their launch vehicles (the N-, J-, and H-vehicles). The administration of space activities in Japan is coordinated by SAC under the prime minister's office. These two organizations were merged together with National Aerospace Laboratory in October 2003 to form the Japanese Aerospace Exploration Agency (JAXA).

In 1981, ISAS began the research and development of the M-3SII, built primarily for the Halley's comet mission, the first Japanese interplanetary flight. This launch vehicle is a three-staged solid rocket with two strap-on boosters and a kick stage that is optional for high-energy missions such as interplanetary flights. In contrast with the step-by-step improvements, which were usual in the evolution of the Mu-family, significant enhancement of the launch capability was required this time. Only the first stage was inherited from the predecessor, M-3S. The size and thrust of the second and third stages were increased. The eight small strap-on motors of the M-3S were replaced by two much larger strap-on boosters. These modifications enabled the launch of a 780-kg (1720-lbm) payload into LEO or 170 kg (375 lbm) into heliocentric orbit.

Among the other major items to be developed were the interstage structures and separation mechanisms, the nose fairing, and the attitude control system. A movable nozzle system, MNTVC, was introduced to the strap-on booster, while the secondary LITVC systems had been commonly used in the Mu-family. The control system was digitized to increase control logic design flexibility and to reduce power and weight needs. The first two M-3SII launches, with an optional fourth stage, injected the first Japanese interplanetary probes, Sakigake and Suisei, toward Halley's comet in 1985. M-3SII had 2.7 times the performance of the M-3S.

In 1989, ISAS was given approval by the SAC to embark on the development of a new solid-propellant launch vehicle to replace the M-3SII then in use. Increase in demand for the launch of large scientific satellites by ISAS led to the decision to develop the new launch vehicle. This required an amendment to Japan's official space policy. Previously, ISAS was limited to the development of the 1.4-m (4.6-ft) diameter Mu-series of rockets to avoid competition with the NASDA's larger N- and H-series launch vehicles and to avoid duplicating investment. The new rocket, designated M-V, has a 2.5-m (8.2-ft) diameter—an 80% increase. The M-V is a three-stage solid-propellant rocket, with an optional kick stage for high-energy missions. Range safety restrictions limit the vehicle to 110 t (240 klbm) of propellant. As a result, the vehicle is capable of 1815 kg (4000 lbm) to LEO or 300–400 kg (660–880 lbm) for planetary missions. To achieve this capability all stages were newly developed, with no direct heritage from the current M-3SII. Sizing effects and design simplifications such as removal of the strap-on boosters will reduce recurring cost. At the launch site, only the launch tower used for the M-3SII needed to be modified. A sea level test stand was constructed at Noshiro Testing Center—ISAS's static firing test facility for the developments of both solid and cryogenic engines—for the first-stage motor firing test.

The first M-V launch occurred in 1997, carrying the Muses B radio astronomy satellite. The M-V supported a shift in the focus of ISAS missions from LEO to interplanetary spacecraft. Japan's first two interplanetary spacecraft were launched to Halley's comet in 1985 on the M-3SII, followed by a small lunar probe in 1990. Most ISAS missions have been LEO astronomy satellites or simple experimental engineering missions. In contrast, about half of M-V launches are scheduled to carry planetary probes. The second M-V flight launched the Nozomi (Planet-B) spacecraft to Mars in July 1998. Future launches will carry a lunar orbiter and the Muses-C asteroid-rendezvous mission.

MINOTAUR

Courtesy Orbital Sciences Corporation.

The Minotaur is a small solid-propellant launch vehicle built from the stages of the Minuteman II missile and Pegasus space launch system. It is used to launch small experimental payloads for the U.S. government.

Contact Information

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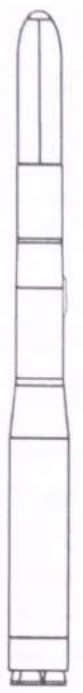
Minotaur



Minotaur

General Description

Summary

The Minotaur is a small ground-launched, solid-propellant, and inertially guided space launch vehicle composed of components from the Minuteman II ICBM and Pegasus XL space launch vehicle. It was developed as part of the U.S. Air Force Orbital/Suborbital Program (OSP), which is intended to provide inexpensive orbital and ballistic launch services using a combination of excess missile assets and off-the-shelf commercial components. Development and recurring launch costs are reduced by utilizing commercially developed flight-proven components and propulsion from the Pegasus launch vehicle. Launch support costs are further reduced by adapting much of the remote site stool launch approach demonstrated by the Taurus launch vehicle.

Status

Operational. First launch 2000.

Origin

United States

Key Organizations

Marketing Organization	U.S. Air Force Space and Missile Systems Center Detachment 12, Rocket Systems Launch Program (RSLP), (Det 12/RP)
Launch Service Provider	USAF SMC/TEBL
Prime Contractor	Orbital Sciences Corporation

Primary Missions

Small U.S. government, U.S. government-sponsored, and university LEO payloads

Estimated Launch Price

$17–20 million (RSLP)

Spaceports

Launch Site	Vandenberg AFB, California
Location	34.7°N, 120.6°W
Available Inclinations	70–110 deg

Launch Site	Cape Canaveral AFS, Florida
Location	28.5°N, 81.0°W
Available Inclinations	28.5–51 deg

Launch Site	Kodiak Launch Complexm Alaska
Location	57.6°N, 152.2°W
Available Inclinations	67–116 deg

Launch Site	Wallops Flight Facility, Virginia
Location	37.9°N, 75.4°W
Available Inclinations	28.5–100 deg

Performance Summary

Performance for Minotaur with its baseline payload fairing is given. The mass of a spacecraft separation system must be subtracted from the value given.

185 km (100 nmi), 28.5 deg	607 km (1339 lbm)
185 km (100 nmi), 90 deg	470 km (1036 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	513 km (1130 lbm)
Sun Sychronous Orbit: 800 km (432 nmi), 98.6 deg	317 km (700 lbm)
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Launch Vehicle Successes	2
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–2 per year

Nomenclature

Minotaur is the nickname adopted for the space launch vehicle configuration in the OSP, more formally referred to as OSP SLV. The Minotaur name refers to the combination of Minuteman and Taurus hardware (although Minotaur actually uses more Pegasus than Taurus components), and shares the mythological theme of Orbital Sciences Corporation's (OSC) other launch vehicles, Pegasus and Taurus.

Cost

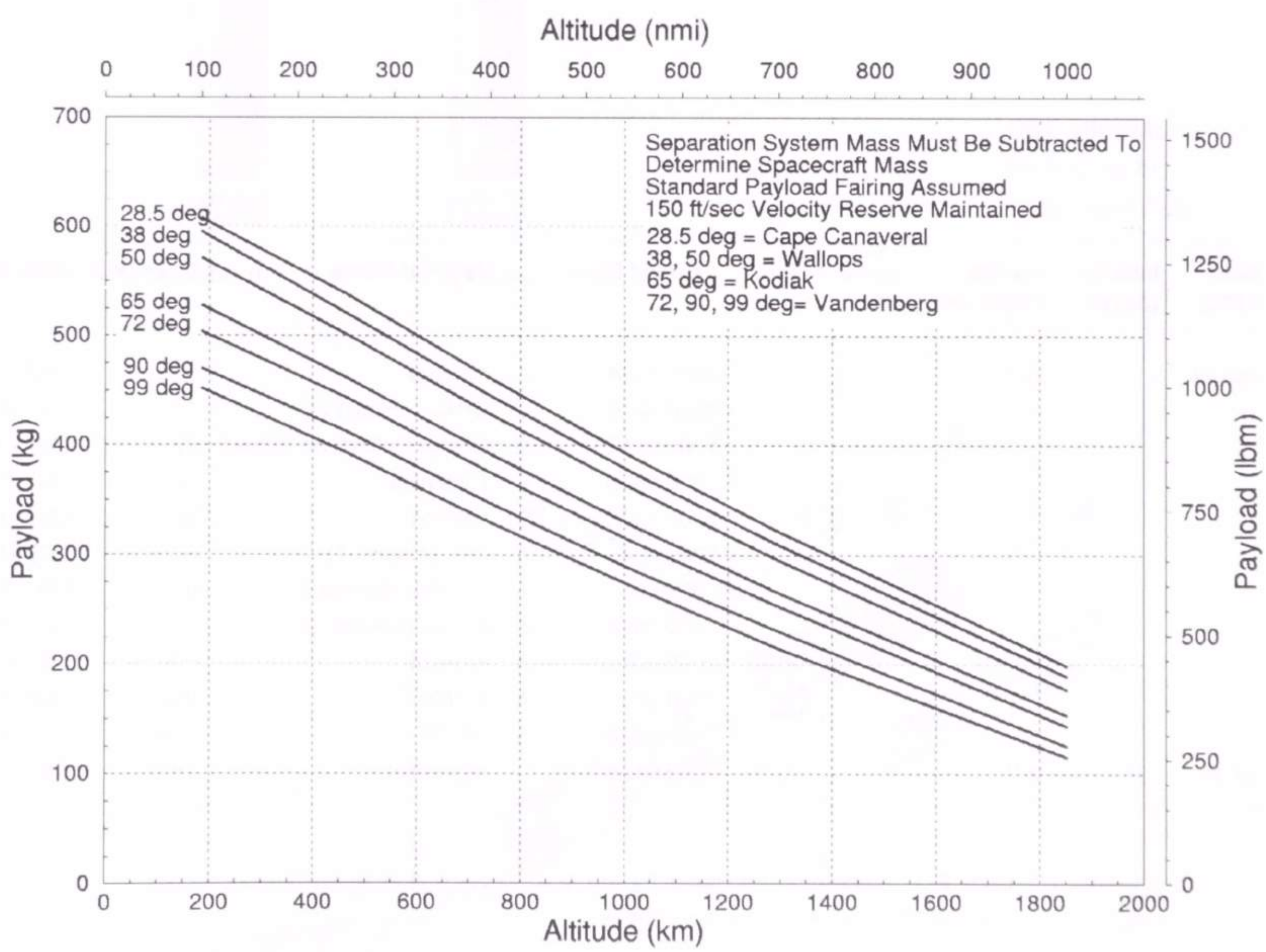
The first Minotaur launch cost approximately $22–$23 million, including the costs associated with development of the launch system. According to Orbital, the typical cost of launching Minotaur is approximately $17–20 million, including the launch vehicle (provided by OSC), program management (Air Force SMC), systems engineering/technical assistance (Northrop Grumman), and launch range costs.

Availability

The Minotaur is available only for U.S. government spacecraft or university satellites with U.S. government sponsorship. U.S. policy prevents launch vehicles built from surplus missile components from being used to launch commercial payloads. The standard time from contractual initiation to launch is 18 months with a minimum of 60 days between launches. A total of five launches are planned under the first OSP contract, which would be carried out through 2004–2006.

Orbital has been awarded a follow-on contract, OSP-2, which includes additional launches of Minotaur, as well as the development of a new launch vehicle based on decommissioned Peacekeeper boosters. The contract is an Indefinite Delivery, Indefinite Quantity (IDIQ) contract, which does not guarantee that the launch vehicle will be built or set a specific launch date.

Performance



Minotaur: LEO Performance

Flight History

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Launch Vehicle Successes	2
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

			Date (UTC)	Interval (days)	Vehicle/ Flight Desig.	Launch Site	Pad	Payload Desig		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1	2000	Jan 27	-	M-1	V	CLF	2000 004A	A	ASUSat	5	LEO (100.2)	NGO	USA
								2000 004B	C	OPAL 1 (Squirt 2)	13	LEO (100.2)	NGO	USA
								2000 004C	C	Optical Calibration Sphere	22	LEO (100.2)	MIL	USA
								2000 004D	C	Falconsat 1	52	LEO (100.2)	NGO	USA
								2000 004E	C	JAWSAT	64	LEO (100.2)	NGO	USA
								2000 004L	A	JAK (Artemis 1)	0.2	LEO (100.2)	NGO	USA
								2000 004J	A	Thelma (Artemis 2)	0.5	LEO (100.2)	NGO	USA
								2000 004K	A	Louise (Artemis 3)	0.5	LEO (100.2)	NGO	USA
								2000 004M	A	Stensat	0.5	LEO (100.2)	NGO	USA
								2000 004H	A	MEMS 1	0.25	LEO (100.2)	MIL	USA
								2000 004H	A	MEMS 2	0.25	LEO (100.2)	MIL	USA
	2		Jul 19	174	M-2	V	CLF	2000 042A		MightySat 2.1	120	SSO	MIL	USA

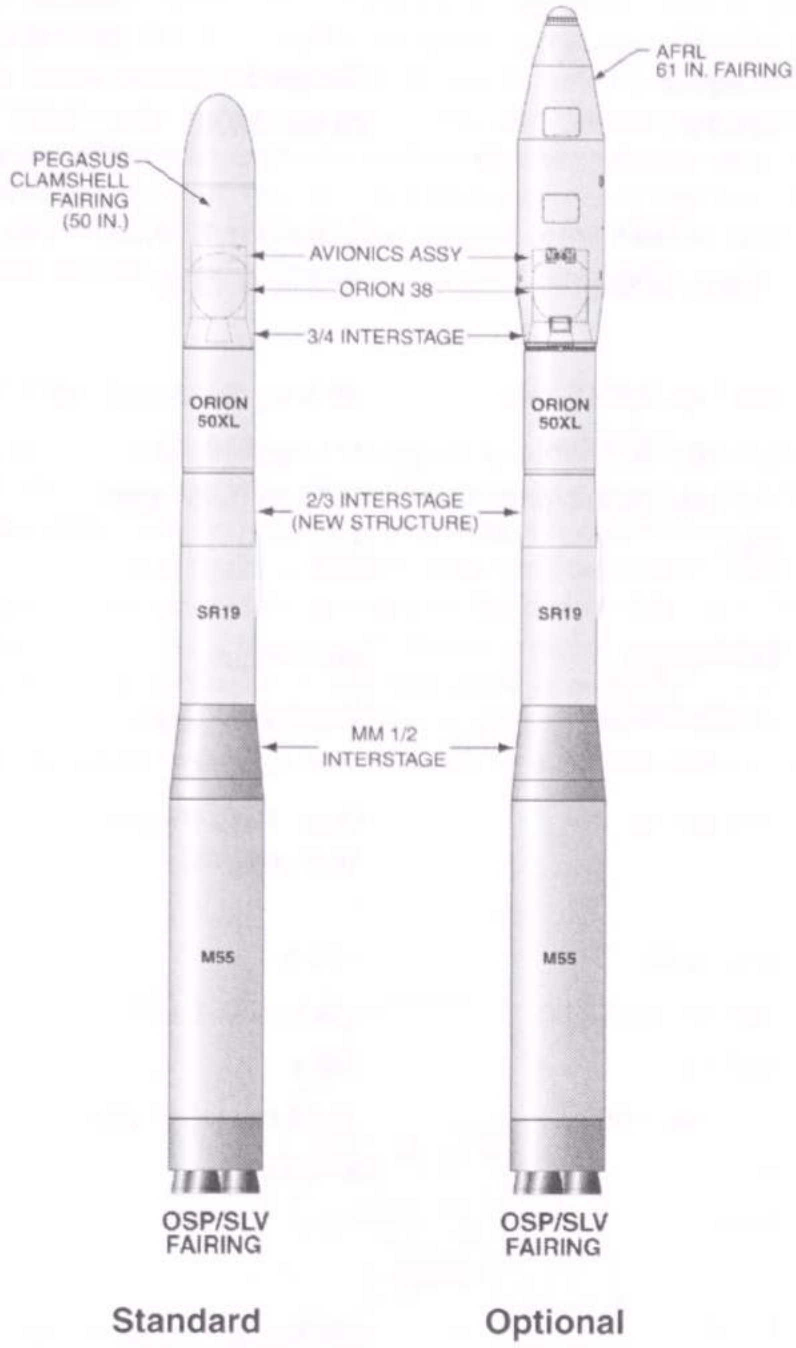
V = Vandenberg AFB

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

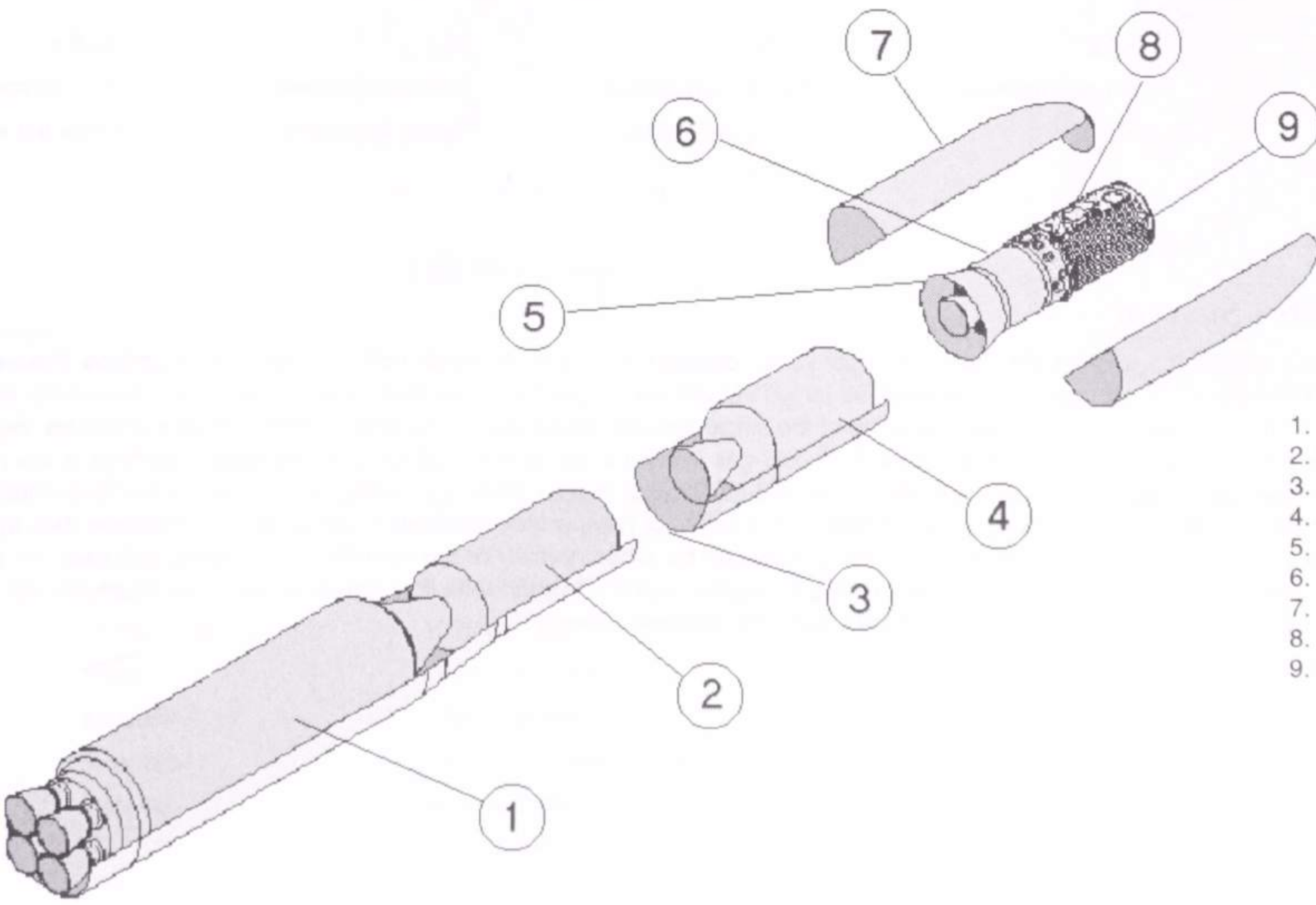
Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Vehicle Design

Overall Vehicle



	OSP Minotaur
Height	19.21 m (63 ft)
Gross Liftoff Mass	36.2 t (79.8 klbm)
Thrust at Liftoff	?



1. Stage 1-M55A1
2. Stage 2-SR-19
3. Stage 2–3 Interstage
4. Stage 3-Orion 50XL
5. Stage 3–4 Interstage
6. Stage 4-Orion 38
7. Fairing
8. Avionics Assembly
9. Payload

Courtesy Orbital Sciences Corporation.

Vehicle Design

Stages

The Minotaur is an all solid-propellant launch vehicle. The major elements of the vehicle are the M55A1 motor (Minuteman II Stage 1), SR-19 motor (Minuteman II Stage 2), Orion 50XL (Pegasus Stage 2), Orion 38 (Pegasus Stage 3), Pegasus avionics section, and Pegasus XL fairing.

	Stage 1 M-55A 1	Stage 2 SR-19	Stage 3 Orion 50XL	Stage 4 Orion 38
Dimensions				
Length	7.49 m (24.57 ft)	4.12 m (13.53 ft)	3.58 m (11.75 ft)	1.34 m (4.39 ft)
Diameter	1.67 m (5.47 ft)	1.33 m (4.35 ft)	1.28 m (4.18 ft)	0.97 m (3.17 ft)
Mass				
Propellant Mass	20785 kg (45830 lbm)	6237 kg (13753 lbm)	3915 kg (8633 lbm)	771 kg (1699 lbm)
Inert Mass	2248 kg (4955 lbm)	691 kg (1524 lbm)	416 kg (918 lbm)	126 kg (278 lbm)
Gross Mass	23077 kg (50885 lbm)	7032 kg (15506 lbm)	4332 kg (9551 lbm)	897 kg (1977 lbm)
Propellant Mass Fraction	0.90	0.89	0.90	0.86
Structure				
Type	Monocoque	Monocoque	Monocoque	Monocoque
Material	D6AC Steel	6Al-4V Titanium	Graphite–epoxy	Graphite–epoxy
Propulsion				
Engine Designation	M55A1	MM SR-19	Orion 50XL (Alliant Techsystems)	Orion 38 (Alliant Techsystems)
Number of Motors	1	1	1	1
Propellant	TP-H1011 Type II	ANB-3066	HTPB	HTPB
Average Thrust	792 kN (178 klbf)	268 kN (60.3 klbf)	154 kN (34.5 klbf)	32 kN (7.2 klbf)
Vacuum Isp	237 s	287.5 s	289 s	290.1 s
Chamber Pressure	50.4 bar (741 psi)	30.1 bar (445 psi)	70.24 bar (1019 psi)	45.22 bar (656 psi)
Nozzle Expansion Ratio	10:1	25:1	58.6:1	67.5:1
Restart Capability	None	None	None	None
Attitude Control				
Pitch, Yaw	Nozzle gimbal ±8 deg	LITVC	Electromechanical nozzle gimbal ±3 deg	Electromechanical nozzle gimbal ±3 deg
Roll	Nozzle gimbal ±8 deg	Warm gas RCS	Nitrogen cold-gas RCS	Nitrogen cold-gas RCS
Staging				
Nominal Burn Time	60.8 s	65.54 s	72.5 s	69.6 s
Shutdown Process	Burn to completion	Burn to completion	Burn to completion	Burn to completion
Stage Separation	Hot fire	Spring separation	Spring separation	Does not separate

Attitude Control System

The Minotaur first stage utilizes the existing Minuteman II thrust-vector control (TVC) system, which uses four nozzles to achieve three-axis control. Control during the second-stage SR-19 motor burn is provided by LITVC with warm gas for roll control. Upper stage (both Orion 50XL and Orion 38 stages) thrust-vector control is supplied by gimbaled actuation of the single flexseal nozzle from independent pitch and yaw actuators. Roll attitude is controlled by cold-gas jets located on the avionics structure. This cold-gas system is identical to that used on Pegasus. Openings in the payload fairing permit reaction control before payload fairing separation. The cold-gas control is also employed during every coast phase to maintain three-axis control. Following orbital insertion, the Minotaur fourth stage executes a series of prespecified commands contained in the mission data to provide the desired initial payload attitude (before payload separation). The attitude can be either inertially or spin stabilized. For inertial attitudes, the payload and fourth stage can be oriented to an accuracy of better than ±1 deg in angular position in each axis. For spin-stabilized, the maximum spin rate attainable depends on the payload and spent fourth-stage combined spin axis moment of inertia.

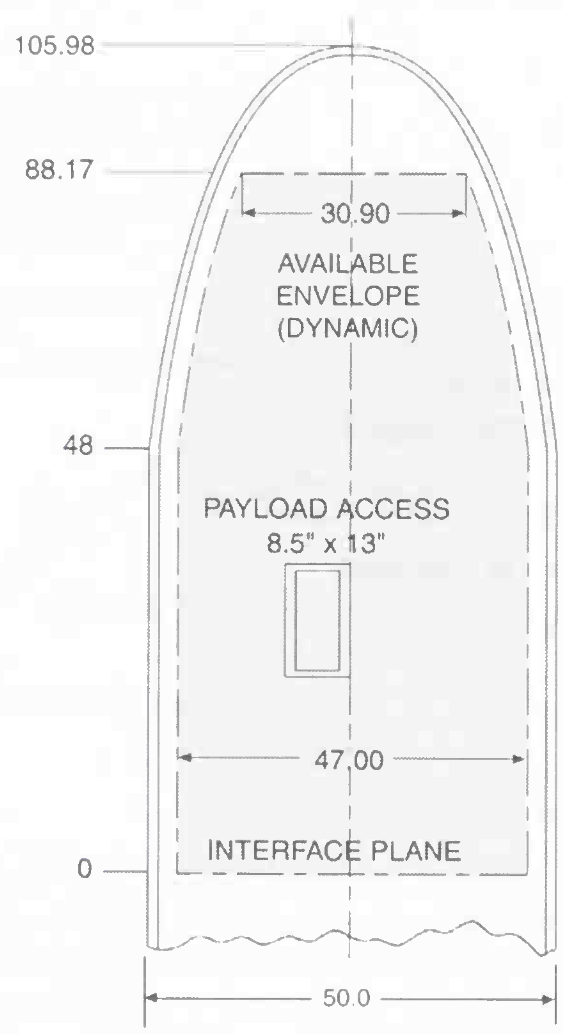
Vehicle Design

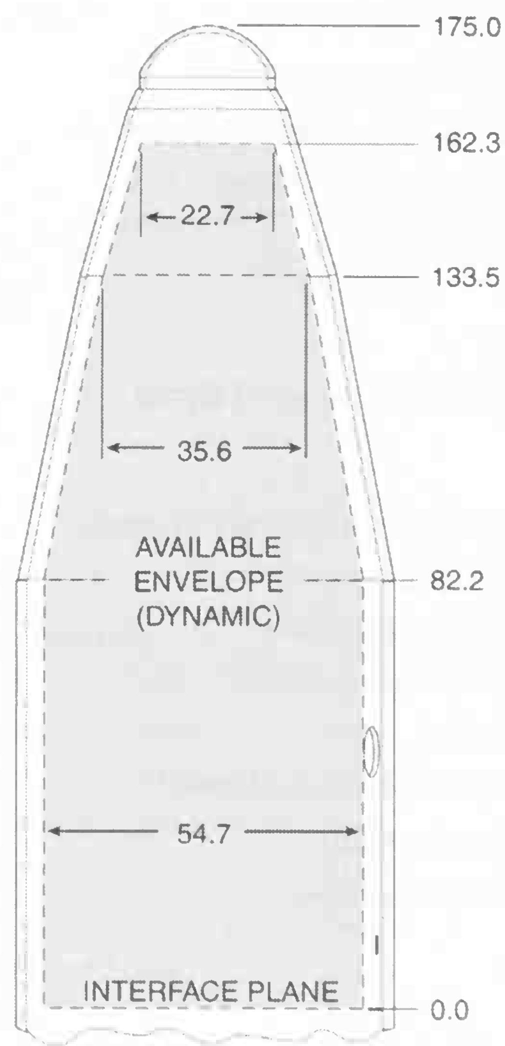
Avionics

Minotaur utilizes avionics similar to those on Pegasus and Taurus with modifications to increase the capability and flexibility of the Minotaur system. As with Pegasus, the avionics are mounted onto the cylindrical avionics structure attached to the Orion 38 with the upper end of the avionics structure providing the payload interface plane. The structure is fabricated from a composite material [12.7-mm (0.5-in.) aluminum honeycomb core with graphite–epoxy face sheets] with the avionics components mounted primarily onto the outer circumference of the structure. Some components such as the Honeywell SIGI inertial navigation system (INS) are mounted to the inner diameter of the structure to facilitate packaging within the fairing volume. The other avionics components include the flight computer, telemetry transmitter, telemetry multiplexer, ordnance and RCS thrusters, dual flight termination receivers, radar transponder, batteries, and various other components and harnesses. A GPS position beacon (GPB) is an option when C-band capability is not available or desired. The GPB is a stand-alone unit that was flown on the first Minotaur, as well as several suborbital missions. An additional encoder can be purchased to increase the amount of information gathered from the vehicle during flight. This option was exercised on the first mission to help define the exact environments experienced during this flight. This increases the total bit rate available to 2 Mbps.

Payload Fairing

The Minotaur uses the Pegasus fairing and interfaces to provide an adaptable, flight-proven system without additional development risk. The payload fairing is identical to the Pegasus fairing. Payload access is provided by a single 216×330 mm (8.5×13 in.) door that can be positioned as required on a mission-by-mission basis. A second door is available as a nonstandard option. The payload interface is the forward flange of the avionics structure via a 985.8-mm (38.81-in.) bolt ring. Optional adapter cones and separation systems are also available in various sizes. A thermal protection system (TPS) is used on the fairing to minimize the temperature of the fairing inner wall. Temperatures inside the fairing are maintained within 93°C (200°F). A low emissivity fairing liner is available, if required, to reduce radiation exchange between the fairing and the payload. Contamination control options, maintaining a Class 100,000 or 10,000 environment, are available on the Minotaur. The payload customer may procure a larger fairing to accommodate their space vehicle. The U.S. Air Force is pursuing a joint large fairing developmental effort between Orbital, Air Force Research Laboratory (AFRL), and Boeing that will use composite materials manufactured by a fiber placement system.





	OSP Standard Fairing	AFRL Fairing
Length	4.42 m (14.5 ft)	6.2 m (20.4 ft)
Primary Diameter	1270 mm (50 in)	1.5 m (20.4 ft)
Mass	194 kg (427 lbm)	342 kg (753 lbm)
Sections	2 shell halves	2 shell halves
Structure	Carbon-composite shell structure	Grid-stiffened
Material	60-mm (2.36-in.) face sheets over 12.7-mm (0.5-in.) aluminum honeycomb	Carbon-epoxy, fiber placement

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	Standard PLF: 1118 mm (44.0 in.)
	Optional PLF: 1390 mm (54.7 in)
Maximum Cylinder Length	Standard PLF: 1110 mm (43.72 in.)
	Optional PLF: 2088 mm (82.2 in)
Maximum Cone Length	Standard PLF: 1016 mm (40.0 in.)
	Optional PLF: 2035 mm (80.1 in)
Payload Adapter Interface Diameter	985.8 mm (38.81 in.)
Payload Integration	
Nominal Mission Schedule Begins	T–18 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T–2 min
On-Pad Storage Capability	Indefinite
Last Access to Payload	T–24 h (shorter times can be coordinated)
Environment	
Maximum Axial Load	+7 to 13 *g* depending on payload
Maximum Lateral Load	± 3.3 *g* (typical 800 lb payload) Note: Highly dependent on spacecraft mass and structure
Minimum Lateral/Longitudinal Payload Frequency	20 Hz/20 Hz
Maximum Acoustic Level	140 dB OASPC
Maximum Flight Shock	3000 *g* from 1850 to 10,000 Hz exclusive of payload separation
Maximum Dynamic Pressure on Fairing	60 kPa (1250 psf)
Nominal Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	4.1kPa/sec (0.6 psi/s)
Cleanliness Level in Fairing	Class 100,000 standard
	Class 10,000 optional
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±18.5 km (±10 nmi) insertion apse ±96.6 km (±50 nmi) non-insertion apse, ±0.2 deg inclination
Optional Enhanced Injection Accuracy (3 sigma)	±18.5 km (±10 nmi) overall, ±0.1 deg inclination
Attitude Accuracy (3 sigma)	±1.0 deg, ±0.5 deg/sec prior to separation
Nominal Payload Separation Rate	0.6 to 0.9 m/s (2 to 3 ft/s)
Deployment Rotation Rate Available	Up to 100 rpm
Loiter Duration in Orbit	40 min (approximate)
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple Manifest or Comanifest	A Pegasus-developed Dual Payload Attachment Fitting (DPAF) is available to enable comanifested payloads. The lower payload is contained within a canister structure that supports the upper payload. After separation of the first payload, the upper surface of the DPAF is jettisoned to deploy the second payload. The lower payload volume is 660 mm (26 in) in diameter, and 559 mm (22.9 in) in height. An AFRL-developed structural space frame dispenser called the multiple payload adapter (MPA) is available with up to four bays for microsatellites. The MPA is a modular design that can be configured to accommodate a variety of spacecraft configurations.

The standard launch service does not include a separation system. A contractor-provided separation system is available as an option. The baseline is a 965-mm (38-in.) separation system. In addition, 584-mm (23-in.) and 432-mm (17-in.) separation systems that have flown on Pegasus are available. An optional payload isolation system is available that reduces transient loads and vibration and shock levels experienced by the spacecraft. It reduces payload performance by 9–18 kg (2040 lbm). The system is flight proven on the first two Minotaur missions and Taurus.

Production and Launch Operations

Production

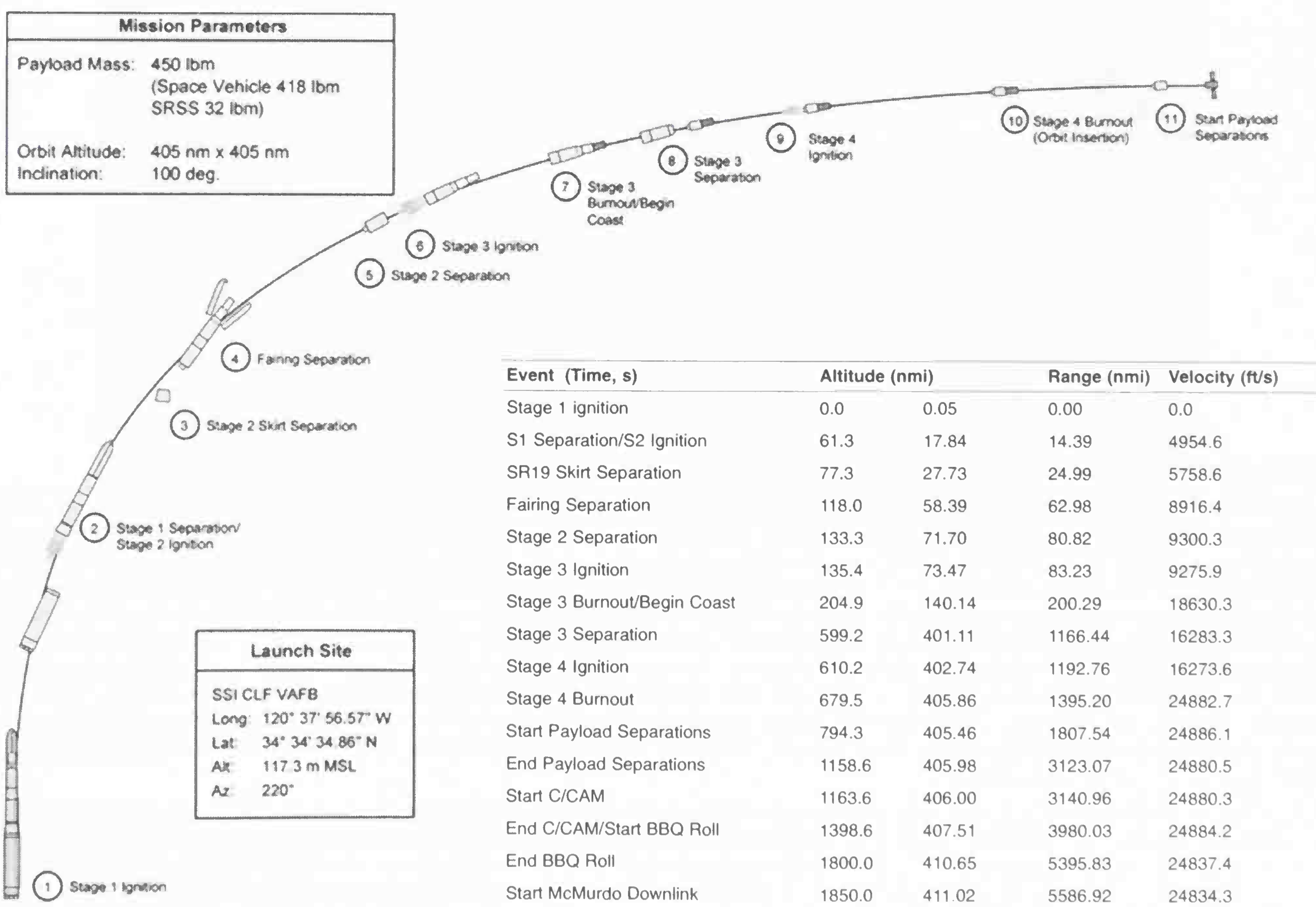
Orbital Sciences Corporation is the primary contractor for the OSP Minotaur launch vehicle. The first two stages are existing missile stages provided by the U.S. government. The second two stages are produced by ATK Thiokol. Litton produces the LN-100LG inertial measurement and Oettle Reichler provides the flight computer.

Launch Operations

The launch operations concept for the OSP Minotaur program is evolved from the Taurus and Pegasus processes. The system is self contained and is capable of supporting a launch from a remote location if necessary. The first stage of the Minotaur stack is a Minuteman II M-55, which is a START-treaty covered booster. This limits the launch sites to those that have been declared a space launch site or a test range in strategic missile treaties. Four commercial spaceports and a USAF-owned launch pad support Minotaur operations. The commercial spaceports that are available are located at Wallops Flight Facility, Virginia; Cape Canaveral Air Force Station, Florida; Vandenberg AFB, California; and Kodiak Island, Alaska. The USAF pad is also located at Vandenberg AFB. Launch site selection will be driven by the technical requirements of the mission. The most important criteria for launch site selection is the planned orbital inclination of the mission. A site at Vandenberg AFB (the commercial spaceport or government pad) or the Kodiak Island spaceport will be used for high-inclination orbits. The Wallops Island or Cape Canaveral spaceports would be used to support lower-inclination orbits. Launch site selection would be made at least one year before the launch date. Please see the Spaceports chapter for further information on these spaceports.

All launches to date have been conducted from Space Launch Complex 8 (SLC-8) at the California Spaceport Commercial Launch Facility (CLF) on Vandenberg AFB. The SLC-8 is located on the south side of VAFB, immediately south of SLC-6. It is a commercially operated facility, developed by Spaceport Systems Inc. (SSI), consisting of equipment buildings, a launch stand, and a flame trench capable of supporting vehicles with up to 4.5 MN (1 Mlbf) of thrust. A permanent moveable gantry structure is currently being constructed under an Air Force contract to support Minotaur launch applications. The first mission to use the tower is scheduled for late 2004. The lower two stages of Minotaur were designed to operate in climate controlled missile silos. To maintain environmental conditions within design limits, the lower stages are wrapped in a yellow plastic cover dubbed "the banana." See the Spaceports chapter for information on other launch sites, and the Athena chapter for information on the Kodiak Launch Complex.

Flight Sequence



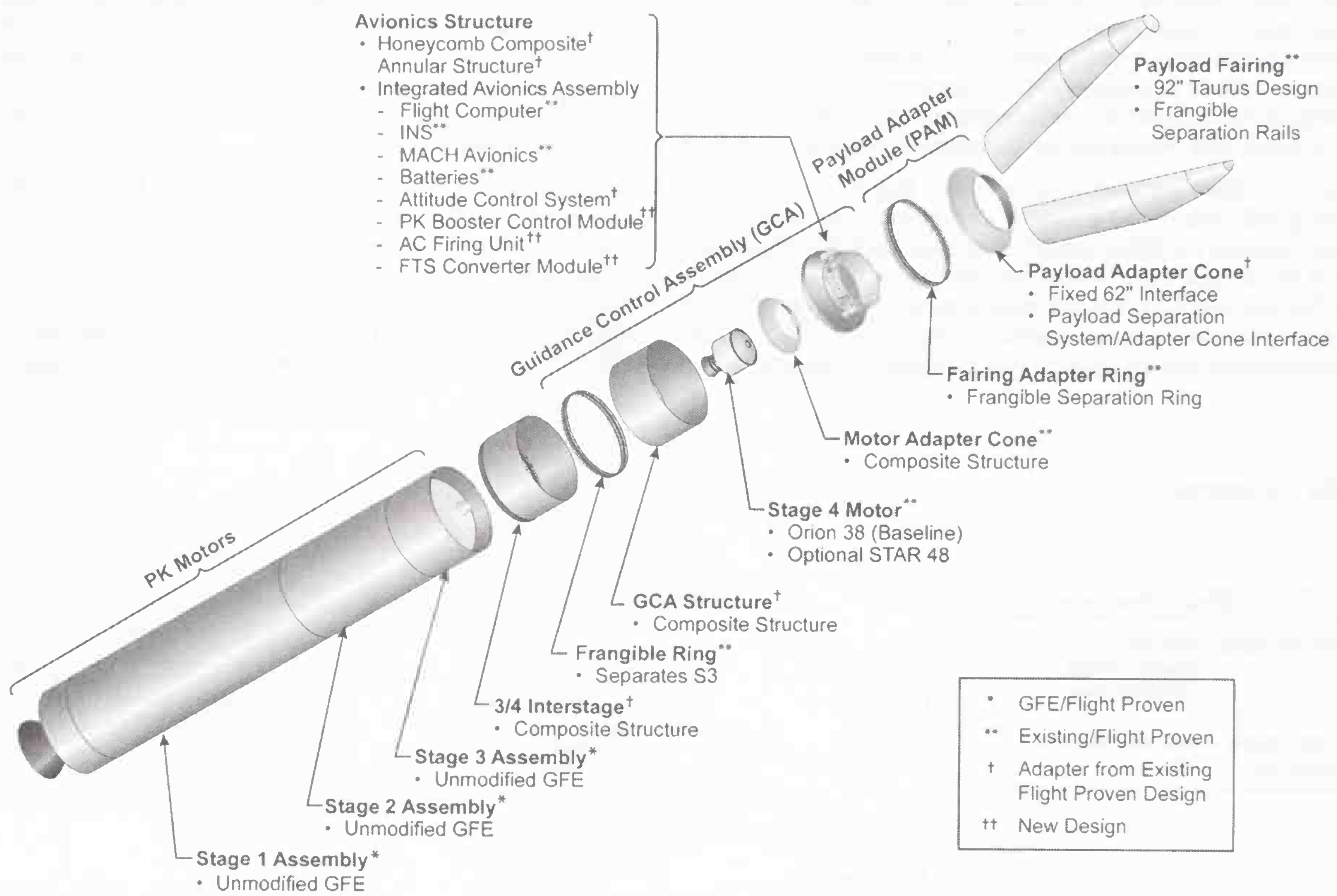
Event (Time, s)	Altitude (nmi)		Range (nmi)	Velocity (ft/s)
Stage 1 ignition	0.0	0.05	0.00	0.0
S1 Separation/S2 Ignition	61.3	17.84	14.39	4954.6
SR19 Skirt Separation	77.3	27.73	24.99	5758.6
Fairing Separation	118.0	58.39	62.98	8916.4
Stage 2 Separation	133.3	71.70	80.82	9300.3
Stage 3 Ignition	135.4	73.47	83.23	9275.9
Stage 3 Burnout/Begin Coast	204.9	140.14	200.29	18630.3
Stage 3 Separation	599.2	401.11	1166.44	16283.3
Stage 4 Ignition	610.2	402.74	1192.76	16273.6
Stage 4 Burnout	679.5	405.86	1395.20	24882.7
Start Payload Separations	794.3	405.46	1807.54	24886.1
End Payload Separations	1158.6	405.98	3123.07	24880.5
Start C/CAM	1163.6	406.00	3140.96	24880.3
End C/CAM/Start BBQ Roll	1398.6	407.51	3980.03	24884.2
End BBQ Roll	1800.0	410.65	5395.83	24837.4
Start McMurdo Downlink	1850.0	411.02	5586.92	24834.3
End McMurdo Downlink /End of Mission	2290.0	412.39	7874.08	24821.2

Minotaur Flight Sequence (Typical)

Vehicle Upgrade Plans

Orbital has been awarded a follow-on contract, dubbed OSP-2, which includes additional launches of Minotaur, as well as the development of a new launch vehicle based on the Peacekeeper missile instead of the Minuteman. The contract is an IDIQ contract, which does not guarantee that the launch vehicle will be built or set a specific launch date.

The new vehicle will use decommissioned motors from Peacekeeper missiles, combined with an Orion 38 fourth stage, avionics and software from Minotaur and Pegasus, and the 2.3-m (92-in.) diameter Taurus payload fairing. The Peacekeeper SLV could carry 1000 kg (2200 lbm) to a 740-km (400-nmi) sun-synchronous orbit, or 1800 kg (4000 lbm) to a 200-km (100-nmi) 28.5 deg orbit. The cost for the first launch is estimated to be $29–31.5 million, after which additional launches would cost $19–22.5 million.



Minotaur IV Peacekeeper-based Space Launch Vehicle

Vehicle History

Historical Summary

In 1991 President George H. W. Bush ordered the demobilization of Minuteman II missiles in compliance with the START II treaty. Between 1993 and 1997, 449 of 450 silos were destroyed (one was left intact as a museum), and several hundred missiles were dismantled and transported to government depots for storage under controlled conditions. These components are controlled and maintained by the Rocket Systems Launch Program (RSLP) located at Kirtland AFB, New Mexico, in support of DoD launch vehicle initiatives. RSLP has used Minuteman assets over the last 30 years, primarily as target vehicles for interceptor and sensor testing. For a number of years, the USAF has considered plans to use these motors for orbital launch systems. By recycling the motors (typically an expensive element of a launch system) it was hoped that satellite launches could be performed at lower costs than launches using available launch vehicles manufactured from scratch. However, the plann sparked objections from some commercial launch service companies who worried that a Minuteman-based launch vehicle would compete unfairly with launch systems in which they had invested. The issue was eventually resolved with a decision to use Minuteman motors only for launches of U.S. government payloads. Development of a Minuteman space launch vehicle was also hindered by the low orbital payload capability of Minuteman configurations. This could be remedied by incorporating larger motors developed for other launch vehicles.

The U.S. Air Force initiated the OSP to produce a consolidated family of Minuteman-derived vehicles for four types of missions: single-reentry vehicle ballistic launch, multiple-payload ballistic launch, flight-test capability for developmental upper stages, and space-lift capability for small U.S. government satellites. The OSP contract was awarded to Orbital Sciences Corporation in 1997. Orbital's proposal included a space launch configuration of OSP that would combine Pegasus XL upper stages with Minuteman lower stages to achieve sufficient performance to meet USAF goals. The contract allowed for up to 24 launches between 1999 and 2004, of which as many as 11 could be satellite launches. Orbital successfully flew the first Minotaur demonstration launch 28 months later in January 2000. The mission carried a collection of micro-, nano-, and picosatellites built by universities and the U.S. Air Force Academy. An operational launch followed a few months later. Four additional Minotaur missions are currently on contract with launches scheduled starting the 4th quarter of CY04 into 2006.

PEGASUS

Courtesy Orbital Sciences Corporation.

L-1011 and Pegasus at takeoff (top) and just before launch (bottom).

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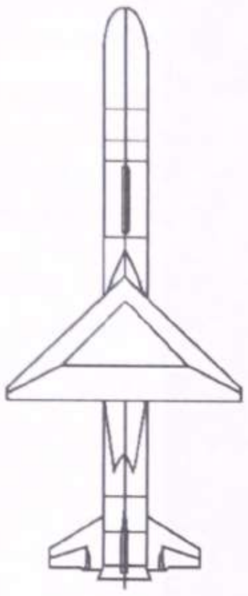
Pegasus XL

General Description



Pegasus XL

Summary

Pegasus is a small commercial launch vehicle developed by Orbital Sciences Corporation. Pegasus is an air-launched, solid-propellant booster with wings. It is launched from Orbital's L-1011 Stargazer aircraft. The air launch provides improved performance compared with other launch vehicles of the same size. It also allows Pegasus to reach a wide range of orbits and to operate from any location with a suitable airfield. The current version of Pegasus is the Pegasus XL, which is longer than the original Pegasus configuration.

Status

Operational. First Pegasus XL launch in 1994.

Origin

United States

Key Organizations

Marketing Organization	Orbital Sciences Corporation
Launch Service Provider	Orbital Sciences Corporation
Prime Contractor	Orbital Sciences Corporation

Primary Missions

Small commercial and government LEO payloads

Estimated Launch Price

$15–25 million (OSC, 2002)

Spaceports

Pegasus can operate from a number of established airfields and ranges. In addition to the sites listed, Pegasus has operated from Edwards AFB, California, and from Gando AFB, Canary Islands. Other sites such as Kodiak Island, Alaska, and Alcântara, Brazil, are options for future missions. Please refer to the Spaceports chapter for information on these sites. Pegasus is released from its carrier aircraft over the ocean. The coordinates of sites listed below reflect the location of the airfield used for takeoff, not the point of release of the launch vehicle.

Launch site	Vandenberg AFB
Location	34.7° N, 120.6° E
Available Inclinations	70–130 deg
Launch site	Wallops Flight Facility
Location	37.9° N, 75.4° W
Available Inclinations	30–65 deg
Launch site	Cape Canaveral Air Force Station
Location	28.5° N, 81.0° W
Available Inclinations	28–58 deg
Launch site	Reagan Test Site (Kwajalein, Marshall Islands)
Location	8.5° N, 167.7° W
Available Inclinations	11–18 deg, equatorial inclinations achieved with dogleg maneuver

Performance Summary

The performance for the Pegasus launch vehicle with a 965-mm (38-in.) marmon clamp interface is shown below.

Available Inclinations:	0–180 deg
185 km (100 nm), 28.5 deg	443 kg (977 lbm)
185 km (100 nm), 90 deg	332 kg (732 lbm)
Space Station Orbit: 407 km (220 nm), 51.6 deg	361 kg (796 lbm)
Sun-Synchronous Orbit: 800 km (432 nm), 98.6 deg	190 kg (420 lbm)

Flight Record (through 31 December 2003)

(Pegasus XL only—see Flight History section for more information)

Total Orbital Flights	25
Launch Vehicle Successes	21
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	3

Flight Rate

5–6 per year

Nomenclature

The only version of the Pegasus launch vehicle currently offered is the Pegasus XL. The designation indicates the extended length compared to the original Pegasus configuration.

Pegasus missions are assigned a unique number based on the order of contract exercise. This number is used for unique hardware and documentation tracking purposes. At launch they are assigned a mission number based upon the actual sequence of launch. Many Pegasus launch vehicles also receive a unique name. For example, the Pegasus which launched Brazil's SCD-1 satellite was named *Santos Dumont* in honor of the pioneering Brazilian aviator.

The carrier Stargazer is named for the first starship of Captain Picard from the *Star Trek* television series.

Cost

Orbital reported in 2002 that typical prices for Pegasus XL launches are between $15 and $25 million, depending on contract terms and conditions, launch site, and services required. NASA reported it spent $19.6 million to launch the WIRE satellite on Pegasus in 1999, and budgeted $15 million for the HETE-2 launch in 2000, although these amounts may include additional launch expenses incurred by NASA beyond the price paid to Orbital.

Pegasus was originally developed with more than $45 million in private funding and a $8.4 million contract from DARPA for the first launch plus options on future launches. Alliant Techsystems invested $30 million in the development of the Orion series of solid motors for Pegasus.

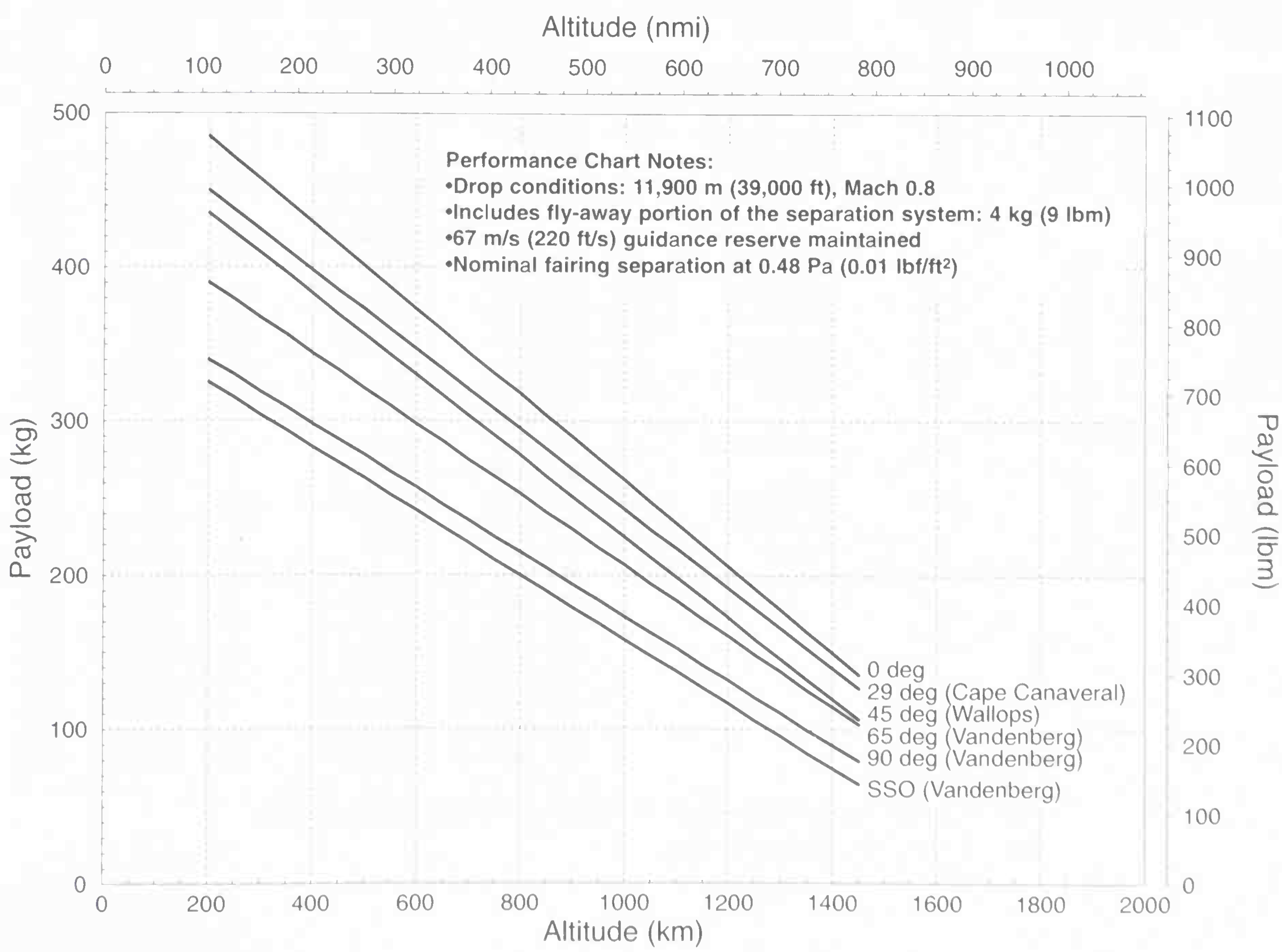
Availability

The first Pegasus launch occurred in 1990. The Pegasus XL first flew in 1994. The original Pegasus configuration is no longer available. Pegasus launch services are available commercially to customers around the world. U.S. government launch services are available through agreements with the U.S. Air Force and with NASA through the SELVS contract run by NASA Kennedy Space Center.

Pegasus has demonstrated a peak flight rate of 5–6 per year and can support a nominal flight rate of 8 per year with sufficient market demand. On a surge basis, Pegasus is capable of launches less than 30 days apart, supporting a flight rate greater than 12 per year. Pegasus has a dual payload attach fitting (DPAF) available to support flight of two small comanifested payloads.

Performance

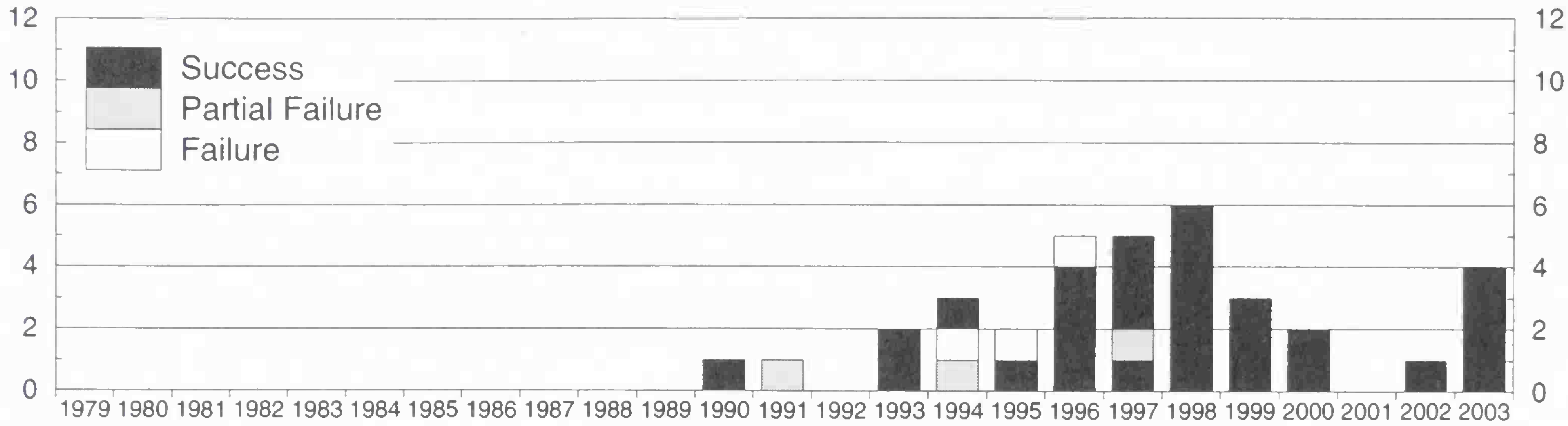
Pegasus missions begin when the Orbital carrier aircraft (OCA) releases the Pegasus booster at an altitude of approximately 12 km (39,000 ft) and a speed of Mach 0.8. The added initial velocity and lower drag of an air launch allow Pegasus to be smaller than a booster of equivalent capability launched from the ground. In addition, the air launch capability means Pegasus can be launched from any ocean area with suitable tracking facilities, and therefore it can reach almost any orbit inclination. In a typical mission, the first-stage booster ignites 5 s after release from the aircraft and quickly initiates a pull-up maneuver to gain altitude. The vehicle decreases its pitch angle about 25 s into flight, and the second-stage ignition occurs shortly after the first stage burns out. There is a long coast period before third-stage ignition to allow the vehicle to reach orbital altitude, and then the third stage provides the remaining velocity to achieve orbit. Because solid motors have relatively large uncertainties in performance, Pegasus trajectories are designed with an additional velocity margin in the form of a flight performance reserve. If motor performance is less than nominal, the velocity margin ensures sufficient performance to reach the target orbit. If motor performance meets or exceeds nominal levels, the additional velocity can be eliminated by steering out of plane, or it can be used to maximize the orbit altitude. Orbital offers a small hydrazine-fueled fourth stage called the hydrazine auxiliary propulsion system (HAPS) located inside the avionics section for minimal impact on available payload volume. Missions using the HAPS stage can achieve higher performance at high altitude and also have improved orbit injection accuracy.



Pegasus XL: LEO Performance

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Pegasus XL	Combined Pegasus Family
Total Orbital Flights	25	35
Launch Vehicle Successes	21	29
Launch Vehicle Partial Failures	1	3
Launch Vehicle Failures	3	3

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Air Field	Aircraft	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1	1990 Apr 05	—	Std	X-1	Edwards AFB	B-52	1990 028A		Pegsat	178	LEO (94.1)	CIV	USA
								1990 028B	A	USA 55	25	LEO (94.1)	MIL	USA
T,P	**2**	**1991 Jul 17**	**468**	**Std/HAPS**	**X-2**	**Vandenberg**	**B-52**	1991 051A	M	Microsat 1–7	7@22	LEO (82)	MIL	USA
	3	1993 Feb 09	573	Std	F-3	Canaveral	B-52	1993 009A	A	OXP 1	15	LEO (25)	CML	USA
								1993 009B		SCD 1	115	LEO (25)	CIV	Brazil
	4	Apr 25	75	Std	F-4	Edwards AFB	B-52	1993 026A		ALEXIS	109	LEO (69.9)	CIV	USA
P	**5**	**1994 May 19**	**389**	**Std/HAPS**	**F-5**	**Edwards AFB**	**B-52**	1994 029A		STEP 2	180	LEO (82)	MIL	USA
F	**6**	**Jun 27**	**39**	**XL**	**F-6**	**Edwards AFB**	**L-1011**	1994 F03A		STEP 1	347		MIL	USA
	7	Aug 03	37	Std	F-7	Edwards AFB	B-52	1994 046A		APEX	261	LEO (69.8)	MIL	USA
	8	1995 Apr 03	243	Std	F-8	Vandenberg	L-1011	1995 016A		Orbcomm 1	43	LEO (70)	CML	USA
								1995 016B		Orbcomm 2	43	LEO (70)	CML	USA
								1995 016C		Microlab 1	74	LEO (70)	CIV	USA
F	**9**	**Jun 22**	**80**	**XL**	**F-9**	**Vandenberg**	**L-1011**	1995 F03A		STEP 3	267		MIL	USA
	10	1996 Mar 09	261	XL	F-10	Vandenberg	L-1011	1996 014A		REX 2	113	LEO (90)	MIL	USA
	11	May 16	68	Std	F-11	Vandenberg	L-1011	1996 031A		MSTI 3	200	SSO	MIL	USA
	12	Jul 02	47	XL	F-12	Vandenberg	L-1011	1996 037A		TOMS-EP	295	SSO	CIV	USA
	13	Aug 21	50	XL	F-13	Vandenberg	L-1011	1996 049A		FAST	192	LEO (83)	CIV	USA
F	**14**	**Nov 04**	**75**	**XL**	**F-14**	**Wallops**	**L-1011**	1996 061A	C	SAC B	191	LEO (38)	CIV	Argentina
								1996 061B	C	HETE	125	LEO (38)	CIV	USA
	15	1997 Apr 21	168	XL	F-15	Gando, Spain	L-1011	1997 018A		Minisat 01	201	LEO (28.5)	CIV	Spain
								1997 018B	A	Celestis 01	1	LEO (28.5)	CML	USA
P	**16**	**Aug 01**	**102**	**XL**	**F-16**	**Vandenberg**	**L-1011**	1997 037A		OrbView 2	309	SSO	CML	USA
	17	Aug 29	28	XL	F-17	Vandenberg	L-1011	1997 047A		FORTE	212	LEO (70)	CIV	USA
S	18	Oct 22	54	XL	F-18	Wallops	L-1011	1997 063A		STEP 4	395	LEO (45)	MIL	USA
	19	Dec 23	62	XL/HAPS	F-19	Wallops	L-1011	1997 084A	M	Orbcomm 8	43	LEO (45)	CML	USA
								1997 084B	M	Orbcomm 10	43	LEO (45)	CML	USA
								1997 084C	M	Orbcomm 11	43	LEO (45)	CML	USA
								1997 084D	M	Orbcomm 12	43	LEO (45)	CML	USA
								1997 084E	M	Orbcomm 9	43	LEO (45)	CML	USA
								1997 084F	M	Orbcomm 5	43	LEO (45)	CML	USA
								1997 084G	M	Orbcomm 6	43	LEO (45)	CML	USA
								1997 084H	M	Orbcomm 7	43	LEO (45)	CML	USA

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Air Field	Aircraft	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	20	1998 Feb 26	65	XL	F-20	Vandenberg	L-1011	1998 012A		SNOE	115	SSO	CIV	USA
S								1998 012B	A	Teledesic T1 (BATSat 1)	70	SSO	CML	USA
	21	Apr 02	35	XL	F-21	Vandenberg	L-1011	1998 020A		TRACE	250	SSO	CIV	USA
	22	Aug 02	122	XL	F-22	Wallops	L-1011	1998 046A	M	Orbcomm 13	43	LEO (45)	CML	USA
								1998 046B	M	Orbcomm 14	43	LEO (45)	CML	USA
								1998 046C	M	Orbcomm 15	43	LEO (45)	CML	USA
								1998 046D	M	Orbcomm 16	43	LEO (45)	CML	USA
								1998 046E	M	Orbcomm 17	43	LEO (45)	CML	USA
								1998 046F	M	Orbcomm 18	43	LEO (45)	CML	USA
								1998 046G	M	Orbcomm 19	43	LEO (45)	CML	USA
								1998 046H	M	Orbcomm 20	43	LEO (45)	CML	USA
	23	Sep 23	52	XL/HAPS	F-23	Wallops	L-1011	1998 053A	M	Orbcomm 21	43	LEO (45)	CML	USA
								1998 053B	M	Orbcomm 22	43	LEO (45)	CML	USA
								1998 053C	M	Orbcomm 23	43	LEO (45)	CML	USA
								1998 053D	M	Orbcomm 24	43	LEO (45)	CML	USA
								1998 053E	M	Orbcomm 25	43	LEO (45)	CML	USA
								1998 053F	M	Orbcomm 26	43	LEO (45)	CML	USA
								1998 053G	M	Orbcomm 27	43	LEO (45)	CML	USA
								1998 053H	M	Orbcomm 28	43	LEO (45)	CML	USA
	24	Oct 23	30	Std	F-24	Canaveral	L-1011	1998 060A		SCD 2	115	LEO (25)	CIV	Brazil
	25	Dec 06	44	XL	F-25	Vandenberg	L-1011	1998 071A		SWAS	284	LEO (69.9)	CIV	USA
S	26	1999 Mar 05	89	XL	F-26	Vandenberg	L-1011	1999 011A		WIRE	254	SSO	CIV	USA
S	27	May 18	74	XL/HAPS	F-27	Vandenberg	L-1011	1999 026A		TERRIERS	124	SSO	CIV	USA
								1999 026B	A	MUBLCOM	43	SSO	MIL	USA
	28	Dec 04	200	XL/HAPS	F-28	Wallops	L-1011	1999 065A	M	Orbcomm 30	45	LEO (45)	CML	USA
								1999 065B	M	Orbcomm 31	45	LEO (45)	CML	USA
								1999 065C	M	Orbcomm 32	45	LEO (45)	CML	USA
								1999 065D	M	Orbcomm 33	45	LEO (45)	CML	USA
								1999 065E	M	Orbcomm 34	45	LEO (45)	CML	USA
								1999 065F	M	Orbcomm 35	45	LEO (45)	CML	USA
								1999 065G	M	Orbcomm 36	45	LEO (45)	CML	USA
	29	2000 Jun 07	186	XL	F-29	Vandenberg	L-1011	2000 030A		TSX 5	250	LEO (69)	MIL	USA
	30	Oct 09	124	Std	F-30	Kwajalein	L-1011	2000 061A		HETE 2	124	LEO (2)	CIV	USA
	31	2002 Feb 05	484	XL	F-31	Canaveral	L-1011	2002 004A		HESSI	293	LEO (38)	CIV	USA
	32	2003 Jan 25	354	XL	F-32	Canaveral	L-1011	2003 004A		SORCE	289	LEO (40)	CIV	USA
	33	Apr 28	93	XL	F-33	Canaveral	L-1011	2003 017A		GALEX	281	LEO (29)	CIV	USA
	34	Jun 26	59	XL	F-34	Vandenberg	L-1011	2003 030A		Orbview 3	305	SSO	CML	USA
	35	Aug 13	48	XL	F-35	Vandenberg	L-1011	2003 036A		SciSat 1	154	LEO (74)	CIV	Canada

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

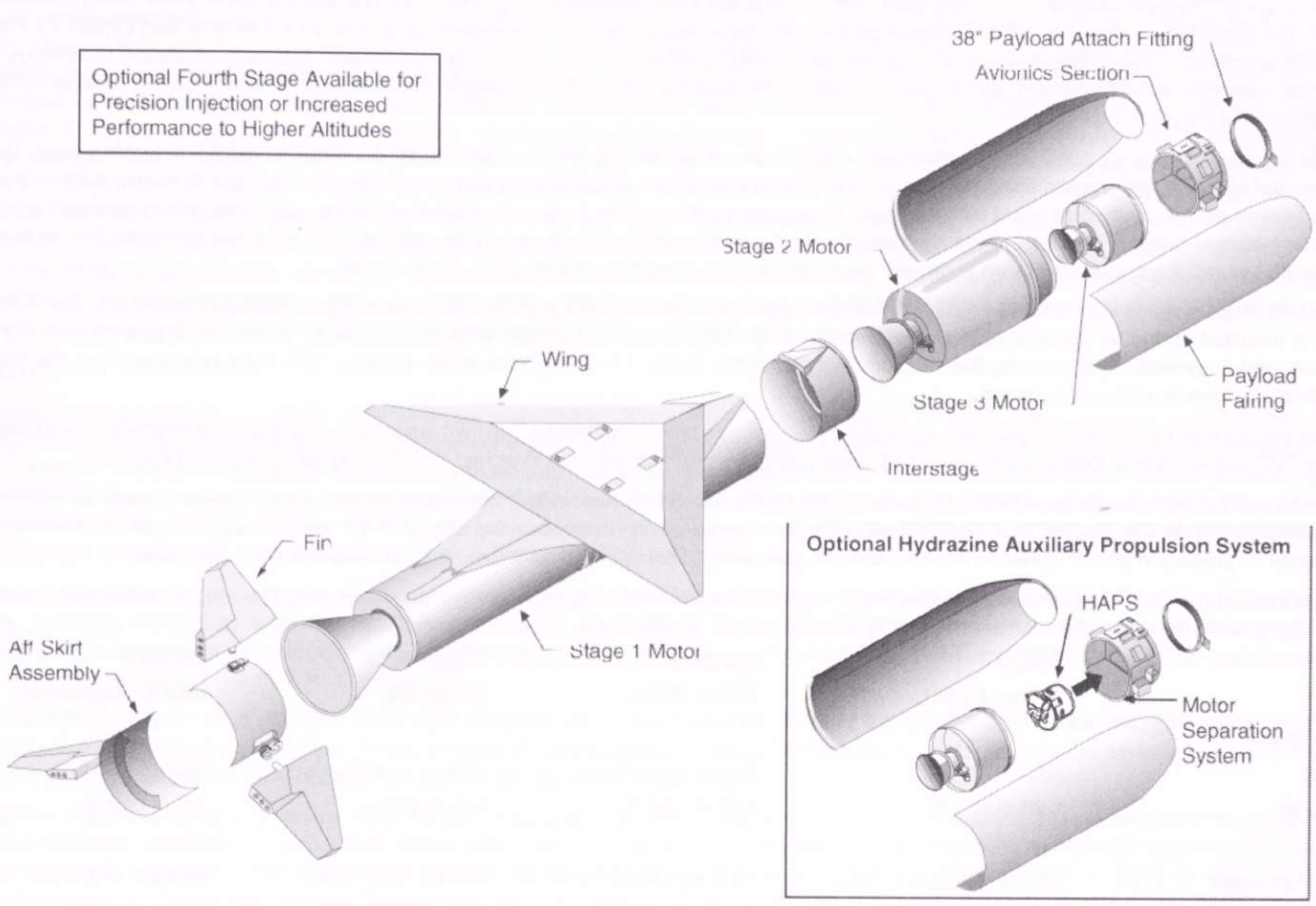
Flight History

Failure Descriptions:

P	1991 Jul 17	X-2	1991 051	Malfunction of the pyrotechnic separation system caused the launch vehicle to veer off course following first-stage separation. Course corrections during subsequent stage burns allowed the launch vehicle to reach orbit, but at a much lower altitude than planned. Spacecraft performed some planned communications experiments, but reentered after six months out of a planned three-year mission.
P	1994 May 19	F-5	1994 029	A software navigation error caused HAPS liquid upper stage to shut down early, resulting in a lower than planned orbit.
F	1994 Jun 27	F-6	1994 F03	At T+39 s vehicle lost control and was destroyed by range safety after first stage burn out. Fault was traced to improper aerodynamics model used in control system autopilot design.
F	1995 Jun 22	F-9	1995 F03	The interstage ring between the first and second stages failed to separate, constraining the second-stage nozzle gimbal and reducing control authority; the vehicle began to tumble out of control during second stage flight, and was destroyed by the range safety officer.
F	1996 Nov 04	F-14	1996 061	Satellites were delivered to the correct orbit, but the launch vehicle separation system failed to deploy them. Failure to separate resulted from a rapid decrease in voltage from the transient battery before the payload separation pyro event. A defective battery exposed to flight staging environments most likely was the failure mechanism.
P	1997 Aug 01	F-16	1997 037	Pegasus reached an orbit with apogee 98 km lower than planned. Orbview-2 reached mission orbit using additional propulsive manuevers. Although considered a partial failure according the definition used in this publication, both Orbital Sciences and the payload customer consider the launch a success.
S	1997 Oct 22	STEP 4	1997 063A	Contact with spacecraft lost after successful launch.
S	1998 Feb 26	Teledesic T1 (BATSat 1)	1998 012B	Satellite was inserted into correct orbit, but did not become operational.
S	1999 Mar 05	WIRE	1999 011A	When the spacecraft was turned on, a power surge prematurely triggered explosive bolts that deployed the cover of the infrared telescope. Solid hydrogen needed to cool the system sublimated and vented, causing the spacecraft to spin out of control. Control was later regained, but not before all the hydrogen had been lost.
S	1999 May 18	TERRIERS	1999 026A	Spacecraft was inserted into the correct orbit. The spacecraft failed to align itself properly, preventing the solar panels from charging the spacecraft batteries and resulting in the loss of the mission.

Vehicle Design

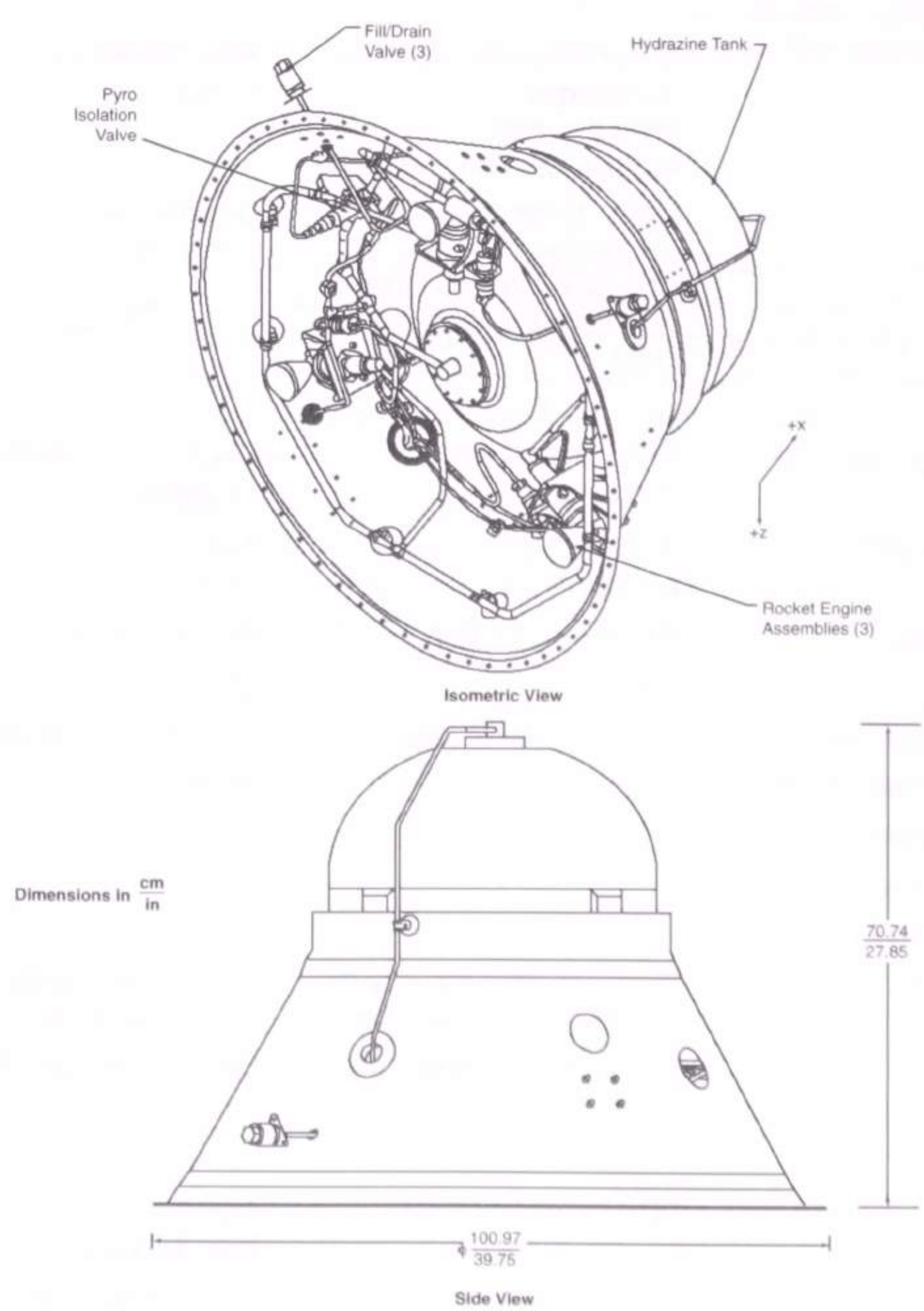
Overall Vehicle



Courtesy Orbital Sciences Corporation.

Pegasus XL

Length	16.9m (55.4 ft)
Gross Liftoff Mass	23 t (51 klbm)
Thrust at Launch	726 kN (163 klbf)



HAPS stage

Vehicle Design

Stages 1–3 and HAPS

The first stage of Pegasus consists of an Orion solid motor, a delta planform composite wing, and an aft skirt assembly with three composite aerodynamic control fins. The Orion family of motors was developed by Hercules (now part of Alliant Techsystems) as part of a joint venture with Orbital for the Pegasus program. The motors have been developed using a conservative design philosophy that includes the use of demonstrated component technology, maximum use of common components and tooling among stages, and the use of class 1.3 propellant. The original Orion 50S motor was stretched to accommodate 24% more propellant for the Pegasus XL vehicle.

The wing configuration is a truncated delta planform with a 45-deg sweptback leading edge and a 6.7-m (22-ft) span. The airfoil is a double wedge. The wing and fins are fabricated of lightweight graphite composite sheets over Nomex foam cores. Graphite wing spars carry the wing load, and aluminum spars in the wing box support the booster while it is attached to the carrier aircraft. The aluminum aft structure supports the three active fins, associated electromechanical fin actuators, and high-voltage battery system. The fins provide aerodynamic control of the vehicle attitude during atmospheric flight. The two horizontal fins were changed on the Pegasus XL to a more anhedral orientation to provide clearance for the landing gear of the L1011 carrier aircraft.

The second-stage motor is very similar in design to the first-stage motor and includes a similar core burning grain and forward dome igniter. A flight termination charge is mounted on the aft dome of each motor to satisfy both range safety and aircraft safety requirements. If initiated, it cuts through the graphite case, insulation, and propellant, compromising the structural integrity of the motor. The Orion 50XL motor contains 30% more propellant than the Orion 50 used on the original Pegasus and Taurus boosters.

The third-stage motor incorporates a head–end grain design to maximize propellant density. The third stage also uses a flexseal nozzle and electromechanical actuators for TVC and employs a toroidal igniter. The third-stage Orion 38 motor has not been stretched from the original Pegasus configuration.

There is an optional liquid fourth stage for the Pegasus XL, the HAPS. The HAPS (now in its third generation) uses a hydrazine propulsion subsystem located inside the existing avionics deck to increase launch vehicle accuracy and improve performance above an altitude of 600 km (320 nmi). The system uses three axially directed thrusters for propulsion and off-pulses to provide pitch and yaw control. Roll control uses the existing cold-gas reaction control system.

The inert mass shown for Stage 1 reflects only the motor itself and the attached wing saddle, truss, and fasteners. The wing structure weighs an additional 285 kg (630 lbm), while the mass of the aft skirt assembly with the tail fins is unknown.

	Stage 1 Orion 50SXL	**Stage 2 Orion 50XL**	**Stage 3 Orion 38**	**Stage 4 HAPS (Optional)**
Dimensions				
Length	10.3 m (33.8 ft)	3.11 m (10.2 ft)	1.34 m (4.4 ft)	0.71 m (2.3 ft)
Diameter	1.28 m (4.2 ft)	1.28 m (4.2 ft)	1 m (3.2 ft)	1.0 m (3.3 ft)
Mass				
Propellant Mass	15,000 kg (33,100 lbm)	3915 kg (8633 lbm)	770 kg (1697 lbm)	59 kg (130 lbm)
Inert Mass	1340 kg (3019 lbm)	416 kg (918 lbm)	410 kg (900 lbm)	22 kg (50 lbm)
Gross Mass	16,400 kg (36,100 lbm)	4331 kg (9551 lbm)	1180 kg (2600 lbm)	?
Propellant Mass Fraction	0.91	0.90	0.65	
Structure				
Type	Motor: Filament-wound monocoque Wing and Fins: composite–foam sandwich	Filament-wound monocoque	Filament-wound monocoque	Monocoque
Material	Motor: Graphite–epoxy composite Wing and Fins: Graphite–epoxy sheets over Nomex foam	Graphite–epoxy composite	Graphite–epoxy composite	Tank: titanium
Propulsion				
Engine Designation	Orion 50S-XL(Alliant Techsystems)	Orion 50-XL(Alliant Techsystems)	Orion 38 (Alliant Techsystems)	MR-107 (Aerojet)
Number of Engines	1 (1 segment)	1 (1 segment)	1 (1 segment)	3
Propellant	HPTB	HTPB	HTPB	Hydrazine
Average Thrust	594 kN (133.5 klbf)	153 kN (34.5 klbf)	47.2 kN (10.6 k lbf)	Approximately 220 N (50 lbf) each
Isp	295 s	289 s	287 s	Approximately 225 s
Chamber Pressure	7515 kPa (1090 psi)	7026 kPa (1019 psi)	4523 kPa (656 psi)	?
Nozzle Expansion Ratio	39.5:1	58.6:1	67.5:1	?
Restart Capability	None	None	None	Yes
Tank Pressurization	—	—	—	Helium
Attitude Control				
Pitch and Yaw	Three electromechanically actuated aerodynamic fins	Electromechanical nozzle gimbaling ±3 deg	Electromechanical nozzle gimbaling, ±3 deg	Thruster pulsing
Roll	Three electromechanically actuated aerodynamic fins	Nitrogen cold-gas RCS	Nitrogen cold-gas RCS	Nitrogen cold-gas RCS
Staging				
Nominal Burn Time	68.6 s	69.4 s	68.5 s	Variable
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Command shut down
Stage Separation	Linear-shaped charge	Linear-shaped charge	Marmon clamp	Marmon clamp

Vehicle Design

Attitude Control System

The Pegasus ACS is fully autonomous. A combination of open-loop steering and closed-loop guidance is employed during the flight. First-stage guidance utilizes a pitch profile optimized using simulations. The fins, mounted on the Pegasus first stage, aerodynamically provide pitch, yaw, and roll control during the first-stage powered flight and the coast period after burnout. During this stage, the vehicle attitude is controlled by the fin actuator system (FAS), which consists of electrically actuated fins located at the aft end of the first stage. Guidance for the second and third stages uses an adaptation of an algorithm that was first developed for the Space Shuttle ascent guidance. Electrically activated thrust vector controllers on the second- and third-stage nozzles control pitch and yaw attitudes. A gaseous nitrogen reaction control system, located forward of the third stage, is used to provide roll control throughout second- and third-stage flight. The cold-gas reaction system is also employed in every coast period to maintain three-axis control. The payload fairing provides openings for the two pods of the RCS thrusters so that reaction control is available before payload fairing separation. The exact sequence of attitude and guidance modes during a flight is controlled by the mission data load (MDL) software and depends upon mission specific requirements. Following orbital insertion, the Pegasus third stage executes a series of prespecified commands contained in the MDL to provide the desired initial payload attitude before payload separation. An inertially fixed, sun-pointing or spin-stabilized attitude may be specified. For inertial attitudes the payload and third stage or the payload and HAPS can be oriented to better than ±3 deg in angular position in each axis. Sun-pointing attitudes can be achieved to an accuracy of 4 deg. For a spin-stabilized initial attitude, the maximum spin rate achievable depends on the payload and spent third stage combined spin-axis moment of inertia.

Avionics

The Pegasus avionics system is a digital distributed processor design. Mission reliability is achieved by using simple designs, high-reliability components, high design margins, and extensive testing at the subsystem and system level. The core of the Pegasus avionics system is a multiprocessor, 32-bit flight computer. The flight computer communicates with the IMU, the launch panel electronics on the carrier aircraft, and all vehicle subsystems using standard RS-422 digital serial data links. Most avionics on the vehicle feature integral microprocessors to perform local processing and to handle communications with the flight computer. This RS-422 architecture enables more efficient integration and test, as it allows unit and system-level testing to be accomplished using commercially available ground support equipment with off-the-shelf hardware.

The avionics subsystem is mounted to the third-stage motor and serves as a mounting structure for most vehicle avionics. These include an IMU, flight computer, telemetry transmitter, telemetry multiplexer, ordnance and thruster driver units, RCS thrusters, dual flight termination receivers, radar transponder, batteries, various other components, and the harness. The structure is composed of a graphite cylindrical section. The avionics structure also provides the mechanical interface for the payload.

Pegasus is controlled by the multiprocessor 32-bit flight computer, which communicates via serial interfaces with individual microprocessors in the vehicle's smart actuator and sensor assemblies. These distributed microprocessors manage the actuators, ordnance initiation devices, and telemetry data-gathering systems, including initialization, self-test, and health reporting via telemetry.

The telemetry system doubles as a ground checkout unit. During flight all critical vehicle performance parameters are transmitted to the ground using two S-band telemetry channels. A C-band radar transponder is provided to improve the ability of ground stations to track the vehicle during ascent. A fully redundant UHF flight termination system is provided to satisfy range safety requirements. Antenna systems for these three RF links are installed on both the second- and third-stage motors. The electrical power system on Pegasus consists of two individual battery and power distribution subsystems.

The electronic airborne support equipment (ASE) for the Pegasus is contained in a single pallet onboard the L-1011 carrier aircraft. The ASE controls the aircraft power supply interface to the launch vehicle, downloads mission data to the vehicle flight computer, enables the vehicle for drop, and provides reference data for the vehicle IMU alignment. The Pegasus flight computer controls microprocessors to manage actuators, ordnance initiation devices, and telemetry data-gathering systems. Two IMUs are used in conducting a Pegasus flight. The vehicle contains an IMU internally that is initialized in flight by a precision IMU carried in the carrier aircraft's ASE.

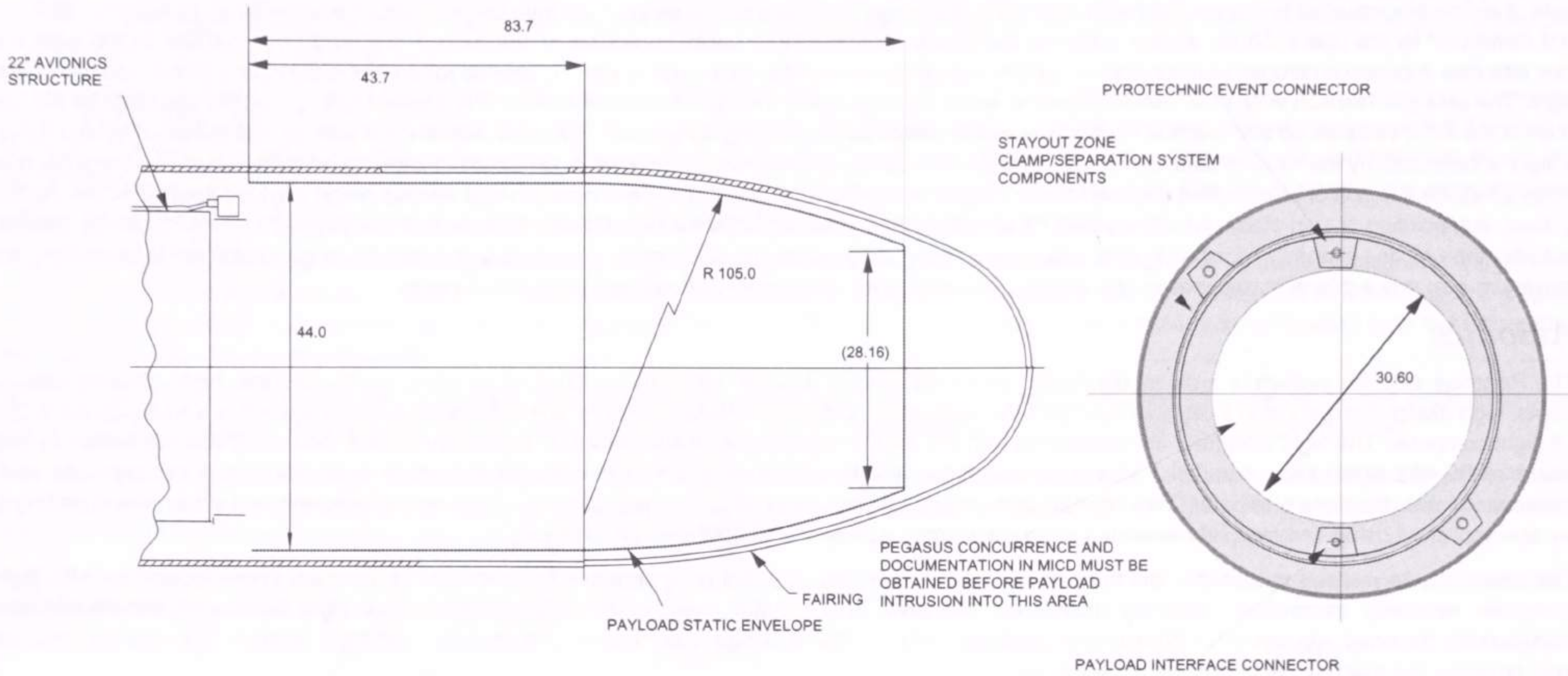
Payload Fairing

The payload fairing consists of a two-piece carbon–composite shell structure, a nose cap integral to one of the shell halves, and a separation system. The fairing maintains the 1.27 m (50 in.) outside diameter of the second-stage motor and completely encloses the smaller diameter third-stage motor, payload, and avionics subsystem. Openings are provided for two sets of RCS thruster pods, payload access doors as required, and pyrotechnic bolt cutters for separation of the fairing clamp rings. Pressure-relief cutouts near the base of the fairing provide bulk venting. When on the ground and during captive flight, the payload area is cooled and maintained under positive pressurization by an air conditioning system. During an abort or in response to special payload requirements, dry gaseous nitrogen can be purged through the fairing from tanks inside the L-1011 carrier aircraft.

Each shell half is composed of a cylinder and ogive sections, which are held together with two titanium straps along the cylinder and a retention bolt in the nose. A cork and room temperature vulcanizing thermal protection system provides protection to the graphite–composite fairing structure. The amount applied has been determined to optimize fairing performance and payload environment protection. The two straps are tensioned using bolts, which are severed during fairing separation with pyrotechnic bolt cutters, while the retention bolt in the nose is released with a pyrotechnic separation nut. The base of the fairing is retained with Orbital's low-contamination frangible separation joint. These ordnance events are sequenced for proper separation dynamics. A hot-gas generator internal to the fairing is also activated at separation to pressure two piston-driven push-off thrusters. These units, in conjunction with cams, force the two fairing halves apart. The halves rotate about fall-away hinges, which guide them away from the satellite and launch vehicle. The fairing and separation system are qualified through a series of structural, functional, and contamination ground vacuum tests and have been successfully flown on all Pegasus XL missions.

Vehicle Design

Static Envelope



Courtesy Orbital Sciences Corporation.

Length	4.42 m (14.5 ft)
Primary Diameter	1.27 m (4.16 ft)
Mass	170 kg (373 lbm)
Sections	Two shell halves
Structure	Carbon–composite shell structure

Payload Accommodations

Payload Compartment	
Maximum Dynamic Payload Diameter	1168 mm (46.0 in.)
Maximum Cylinder Length	1110 mm (43.72 in.)
Maximum Cone Length	1016 mm (40.0 in.)
Payload Adapter Interface Diameter (bolt circle)	985.8 mm (38.81 in.)
Payload Integration	
Nominal Mission Schedule Begins	T–104 weeks
Launch Window	
Last Countdown Hold Not Requiring Recycling	T–4 min
On-Pad Storage Capability	Indefinite
Last Access to Payload	T–3 h
Environment	
Maximum Axial Load	+11 *g* (payload weight dependent)
Maximum Lateral Load	+4.7 *g* static during pull up, ±3.75 *g* in z-axis at base with sinusoidal input during drop transient.
Minimum Lateral/Longitudinal Payload Frequency	20 Hz/20 Hz
Maximum Acoustic Level	119 dB at 800 Hz
Overall Sound Pressure Level	124.8 dB
Maximum Flight Shock (including separation)	3500 *g* from 1000–10,000 Hz
Maximum Dynamic Pressure on Fairing	48 kPa (1018 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	4542 W/m^2 (0.4 BTU/ft^2/s)
Maximum Pressure Change in Fairing	Very low
Cleanliness Level in Fairing	Class 100,000 or better available
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	For 500 km (270 nmi) circular polar orbit: ±111 km (60 nmi), ±0.3 deg inclination With hydrazine precision injection kit: ±28 km (15 nmi), ±0.05 deg inclination
Attitude Accuracy (3 sigma)	±3 deg inertial, ±4 deg sun pointing
Nominal Payload Separation Rate	1 m/s (3 ft/s) (mass dependent)
Deployment Rotation Rate Available	Payload mass dependent
Loiter Duration in Orbit	> 60 min
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Multiple and comanifesting are common. Payloads can be supported in a dual payload attach fitting or the lower satellite(s) may be load bearing.
Auxiliary Payloads	Auxiliary payloads can be accommodated depending on available performance and volume.

Production and Launch Operations

Production

Orbital Sciences Corporation is the primary contractor for the Pegasus launch vehicle.

Subcontractor	Responsibility
Alliant Techsystems	Solid motors and payload fairing
Scaled Composites	Wing and fins
Parker	TVC and RCS
Litton	IMU
Oettle Richler	Flight computer

Launch Operations

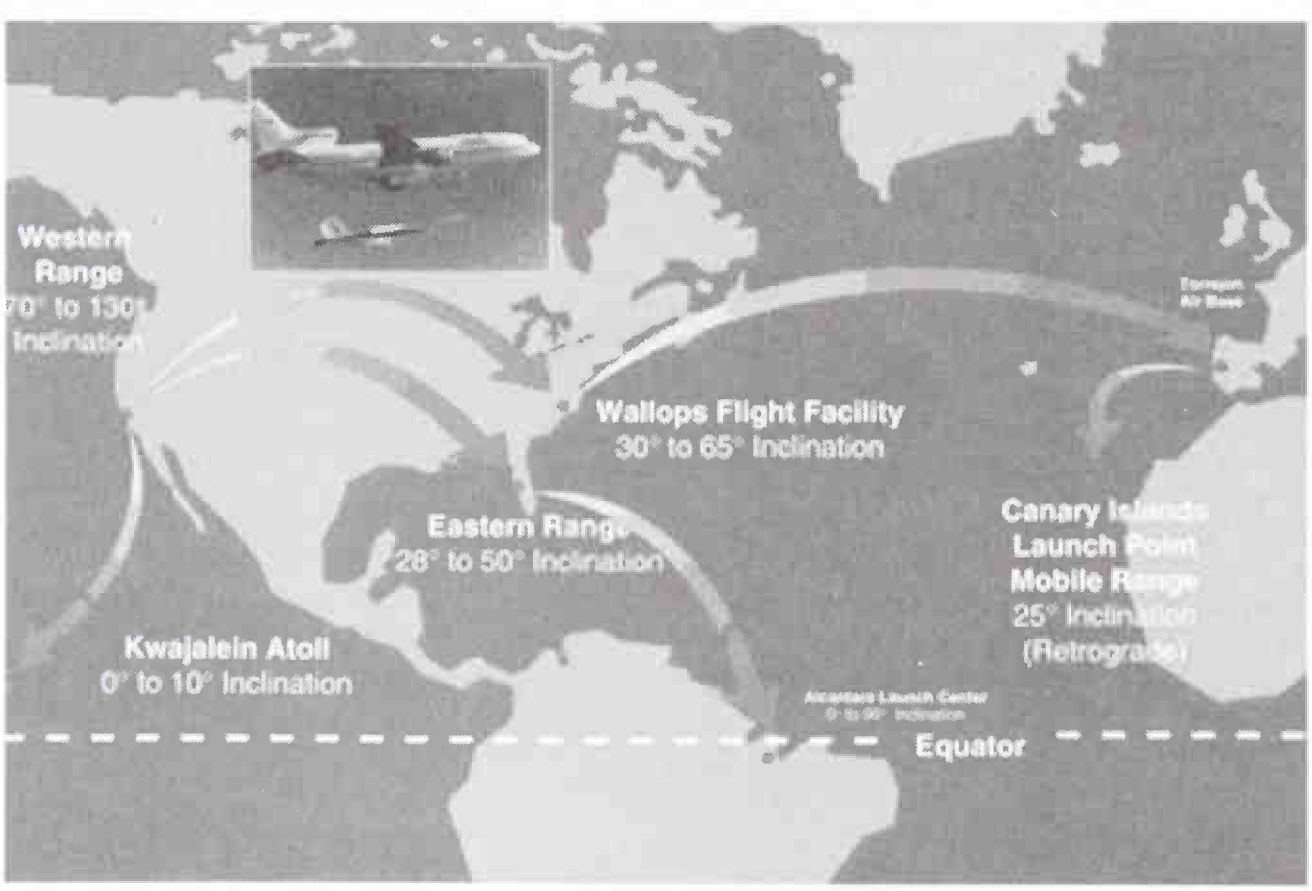
Pegasus was designed to simplify and minimize field integration personnel, facilities, and equipment. The build up of Pegasus begins with the delivery of the SRM sections to the integration site at Vandenberg AFB, California. Motors are shipped in standard ordnance transportation vans (TARVANs) on custom-designed handling dollies. The motor sections remain on these handling dollies throughout the integration process, which eliminates the need for lifting motors in the field. Upon arrival, stages are removed from the TARVANs and placed on a custom designed multifunction assembly and integration trailer (AIT). The AIT has integral lifting jacks that allow it to be elevated to TARVAN bed height so the motor sections can be offloaded directly onto the AIT bed. The AIT is then lowered to floor level for motor offloading to one of the four vehicle integration rails. The avionics subsystem is delivered to the field completely integrated, acceptance-tested, and ready for integration with the third-stage motor. The wing, fins, and payload fairing are received with all thermal protection and instrumentation installed. Once the vehicle has been integrated and tested, the payload is mated and the fairing installed. The rocket is transferred back to the AIT, and the AIT is used to transport Pegasus to the carrier aircraft, elevate it, and align it for mating. The combined AIT and custom dolly system provides full six-degree-of-freedom movement capability for the finished vehicle. A portable air conditioning unit provides filtered, dehumidified air for the payload and avionics during transport. Once the rocket arrives at the airfield, a ground air conditioner provides thermally conditioned clean air until this function is transferred to the airborne air condition system on the carrier aircraft, which provides environmental control until drop.

No launch pad is required for Pegasus, only a runway capable of accommodating wide-body aircraft. Pegasus missions are launched from a modified L-1011 aircraft named Stargazer. Early government-sponsored launches were carried aloft by the NASA-owned B-52 aircraft that supported air drops of the X-15, X-24, X-38, and other experimental aircraft. Using an aircraft instead of a fixed launch complex reduces infrastructure costs. In addition, because launch is initiated well offshore, range safety requirements and associated concerns are reduced compared to typical launch sites.

Launch begins with the release of Pegasus from the carrier aircraft at approximately 11.9 km (39,000 ft) altitude and 0.8 Mach. First-stage ignition occurs 5 s after release from the carrier aircraft, after Pegasus has dropped approximately 90 m (300 ft).

Pegasus launch vehicles are assembled at VAFB, and the "standard mission" includes payload integration and launch from VAFB. In a "ferry mission" the spacecraft is integrated at VAFB, and then flown by the carrier aircraft to another launch site, such as CCAFS, for launch. In a "campaign mission" only the Pegasus is assembled at VAFB, and the spacecraft integration process is conducted at the alternate launch site. This gives Pegasus a significant degree of flexibility to adapt to customer needs and to reach a wider varietry of orbital inclinations. Launches have been conducted from four locations in the United States, plus the Canary Islands, and Kwajelein atoll in the Pacific. The Alcântara launch facility in Brazil is also available for future launches. If the selected launch location does not have tracking, control, and communications systems in place, deployable assets such as the NASA Wallops Mobile Range equipment can be used. Locations of Pegasus launch sites are shown below.

Launch Site

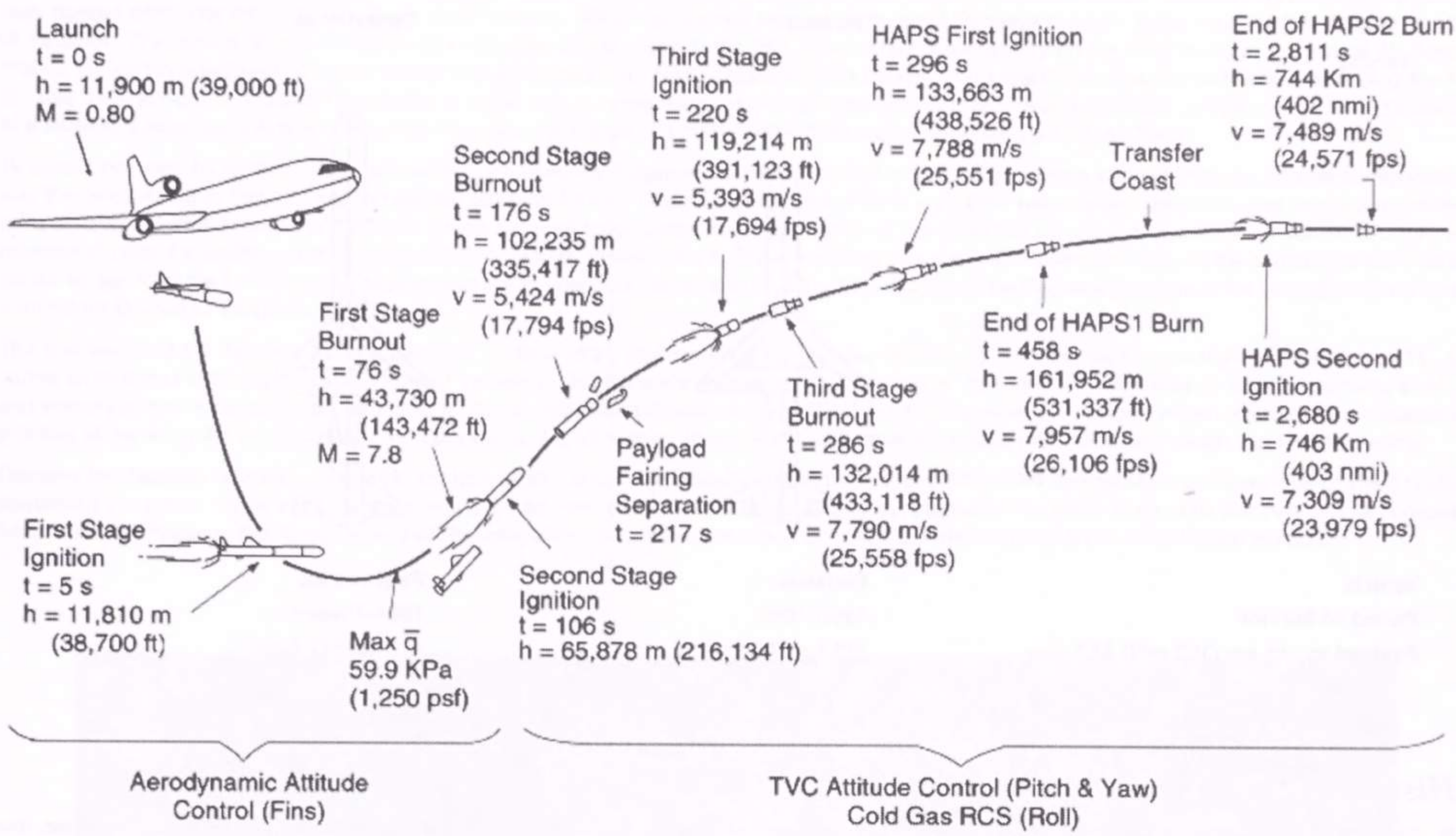


Courtesy Orbital Sciences Corporation.

Examples of Pegasus-Compatible Launch Sites

Production and Launch Operations

Flight Sequence



Courtesy Orbital Sciences Corporation.

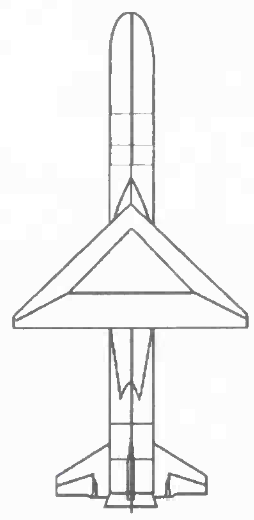
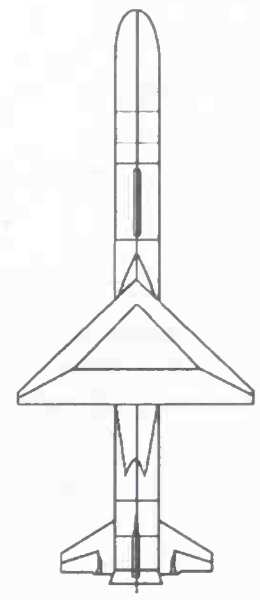
Pegasus XL with HAPS

Pegasus Flight Sequence—Pegasus XL with HAPS to a 741-km (400-nmi) Circular, Polar Orbit with a 251-kg (554-lbm) Payload

Event	Time, s	Altitude, km
Launch	0	11.9
First-stage ignition	5	11.8
First-stage burnout	76	43
Second-stage ignition	106	66
Second-stage burnout	176	102
Payload fairing separation	217	
Third-stage ignition	220	119
Third-stage burnout	286	132
HAPS first ignition	296	134
End of HAPS first burn	458	162
HAPS second ignition	2680	746
End of HAPS second burn	2811	744

Vehicle History

Vehicle Evolution

	Retired	Operational
Vehicle		
Vehicle	**Pegasus**	**Pegasus XL**
Period of Service	1990–1998	1994–Present
Payload to 185 km (100 nmi) 28.5 deg	315 kg (700 lbm)	443 kg (977 lbm)

Historical Summary

In 1987 Orbital Sciences Corporation began development of a new, commercially funded space launch vehicle. Conceived by Dr. Antonio L. Elias, then chief engineer at Orbital, the distinctive air-launched Pegasus space booster was intended to inexpensively launch small, low-cost payloads into space. Orbital entered into a joint venture with Hercules Aerospace in 1988 for the development and production of the Pegasus vehicle. Hercules was responsible for the development of the three brand new solid rocket motors and the payload fairing, while Orbital provided the remaining mechanical and avionics systems, ground and flight software, the carrier aircraft interface, mission integration, and overall system engineering. The total Pegasus development cost of over $50 million was split evenly between the joint venture partners.

In July 1988 Orbital was awarded a $8.4-million, firm, fixed-price contract from DARPA providing for one Pegasus launch vehicle and fixed-price options for five additional missions. All six missions have been exercised and successfully flown. The contract provided for Orbital and Hercules to retain proprietary rights to technology and data developed at private expense under the Pegasus program. Virtually all vehicle development was funded by Orbital and Hercules, with DARPA serving as the anchor tenant for Pegasus launch services. The contract included performance specifications, not vehicle design specifications, allowing Orbital to maintain control over vehicle development. Additional contracts for Pegasus launch services have been awarded by the U.S. Air Force (Air Force Small Launch Vehicle, or AFSLV), NASA (SELVs), the Ballistic Missile Defense Organization (OLS-600), international customers such as the countries of Brazil and Spain, and numerous commercial customers.

Pegasus was the first all-new U.S. space launch vehicle designed since the 1970s. It is a three-stage, solid-propellant, inertially guided, all-composite winged space booster. During the initial series of development flights, the NASA Dryden Flight Research Facility NB-52-008, one of two modified B-52 aircraft used during the X-15 rocket plane research program, was used as the carrier aircraft. DARPA negotiated the necessary agreements to lease this vehicle from NASA Dryden for the first six launches. This particular B-52 aircraft performed Pegasus launches 1–5 and 7. In total, it has performed nearly 450 drops, including test flights for the X-15, which is similar in size and shape to the Pegasus vehicle. Since these initial flights, the Pegasus has been carried aloft by a specially modified Lockheed L-1011 carrier aircraft to level flight launch conditions of approximately 11.9 km (39,000 ft) altitude and Mach 0.8. Following release from the aircraft and ignition of its first-stage motor, Pegasus follows a nearly vacuum optimized lifting ascent trajectory to orbit, carrying 275 kg (600 lbm) payloads to 480 km (250 nmi) polar orbits as well as proportional payloads to other altitudes and inclinations or suborbital trajectories.

Advanced propulsion, structural, and avionics technologies, coupled with an air-launched lifting trajectory give Pegasus approximately twice the performance of a similarly sized ground-launched vehicle. This increase is the result of a number of factors: potential and kinetic energy imparted by the carrier aircraft, reduced aerodynamic drag from the lower air density flight profile, improved propulsion efficiency from higher motor expansion ratios, reduced gravity loss from the unique S-shaped trajectory and wing-generated lift, and reduced thrust direction losses from the reduced velocity vector turning.

Operational advantages over traditional ground-based vehicles operating from fixed pads include significantly reduced range safety concerns [drop from the carrier aircraft typically takes place 80 km (50 mi) offshore], ability to achieve any launch azimuth without doglegs or out-of-plane maneuvering, and ability to fly over or around launch-constraining weather with the carrier aircraft.

The first Pegasus vehicle was rolled out of Orbital's vehicle assembly building (VAB) at NASA Dryden at Edwards AFB in August 1989. Fully flight-like with the exception of inert propellant in the three SRMs, the vehicle was used for a series of integration tests and three captive carry flights with the NASA B-52 carrier aircraft. The first launch took place on 5 April 1990, and successfully placed into orbit an ARPA–Navy experimental communications satellite and a NASA Goddard bus that remained attached to the Pegasus third stage and included two chemical release experiment canisters and a comprehensive payload environment instrumentation package.

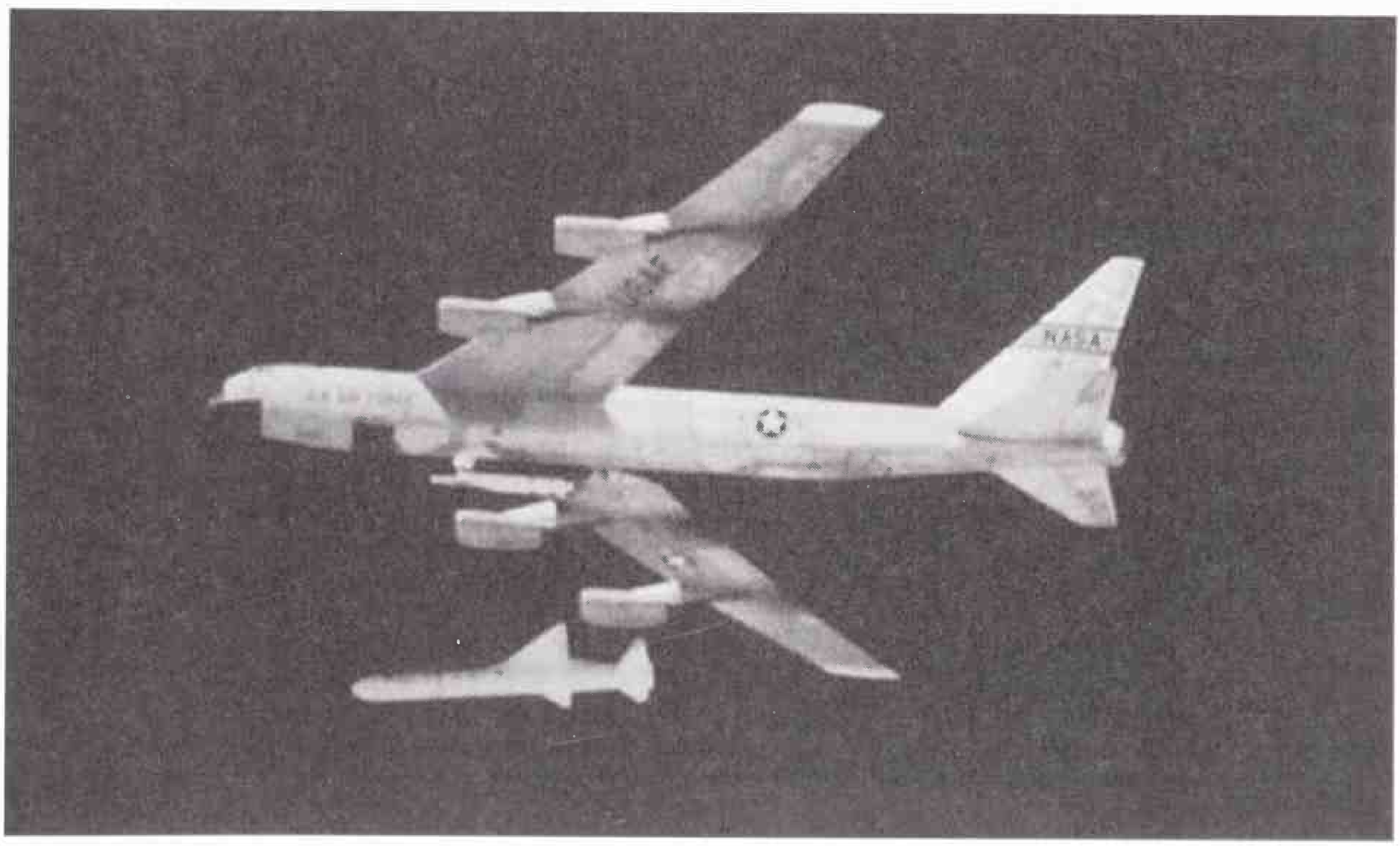
Vehicle History

The need for additional performance and precise injection accuracy led to the development of the HAPS. HAPS is a restartable liquid monopropellant fourth stage for Pegasus, which was first flown on the second mission on 19 July 1991. That mission, also flown for DARPA, placed seven microsatellites in orbit. A staging anomaly during the separation of the spent first stage, unrelated to the HAPS, caused a velocity shortfall and resulted in a lower than desired orbit. The third mission, flown on 9 February 1993, represented the first commercial Pegasus mission. It was also the first Atlantic launch of Pegasus. The vehicle and its Brazilian spacecraft were integrated at the VAB and ferried cross-country on the B-52 to Cape Canaveral Air Force Station for launch operations. Mission control was performed from NASA Wallops Flight Facility. The seventh mission, the last to be flown using the B-52, was the first flight of Orbital's PegaStar spacecraft bus, an integrated design that uses booster avionics and structural components to form the basis of a satellite. Launched 3 August 1994, that PegaStar vehicle carried photovoltaic flight experiments for the U.S. Air Force.

To satisfy the need for even greater payload performance, the Pegasus XL was conceived in 1991. The primary change from the original configuration was the lengthening of first and second stages, allowing for an increase in propellant of 24% and 30%, respectively. Structural and avionics upgrades were also incorporated, however the most visible external modification is the relocation of the two horizontal fins to a downward-canted location as required to clear the landing gear doors on Orbital's L-1011 carrier aircraft. These modifications have also been retrofitted to the shorter standard vehicle for its use from the L-1011, as all subsequent flights used this carrier aircraft. One final change in the Pegasus program is the relocation of the VAB from NASA Dryden to Vandenberg AFB.

The first launch of the Pegasus XL took place on 27 June 1994. Representing the first use of the L-1011 as well as the first flight of the Pegasus XL, all carrier aircraft and initial flight operations went smoothly. Unfortunately, vehicle control was lost at 35 s into flight, followed by loss of telemetry at 38 s and subsequent commanded flight termination. The accident investigation team determined that improperly modeled vehicle aerodynamics resulted in a failure of the autopilot to properly control the vehicle. Aerodynamic models were subsequently revised and validated through wind-tunnel testing.

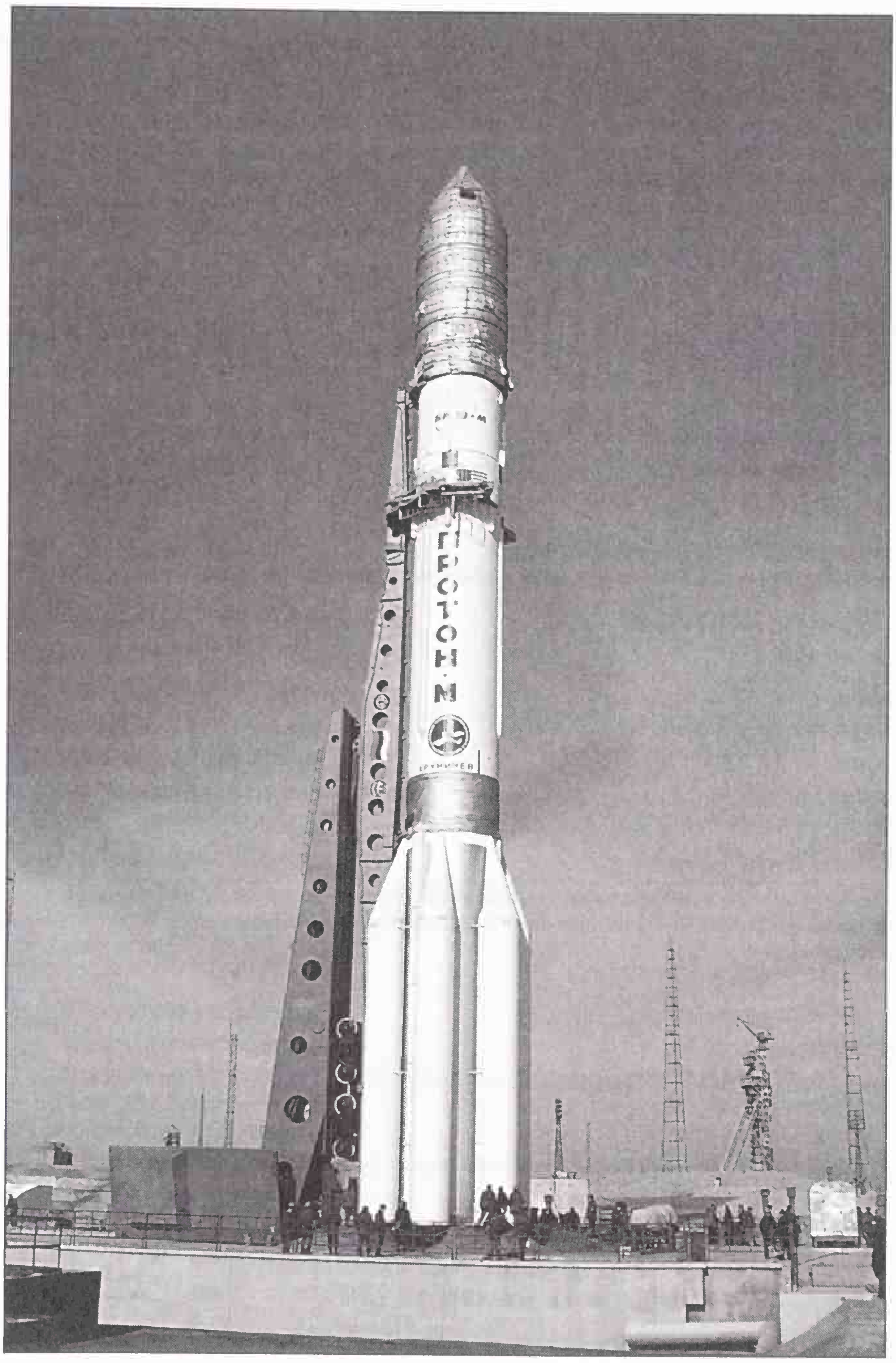
Demand for Pegasus launches boomed in the late 1990s, driven by Orbital's deployment of its ORBCOMM constellation and a number of small NASA spacecraft programs. Since then, Pegasus has suffered from the same shrinkage of the launch market for small spacecraft that has hurt its competition. However, Pegasus has fared better than its competition because of its status as NASA's preferred provider of small launch services.



Courtesy Orbital Sciences Corporation.

An early Pegasus launched from NASA's NB-52B carrier aircraft.

PROTON



Courtesy ILS.

The Proton is the largest Russian launch vehicle. It is capable of launching heavy space station components to LEO or carrying satellites directly to geostationary orbit. Since the mid 1990s it has been marketed by International Launch Services (ILS), a joint venture between Khrunichev Space Center and RSC Energia of Russia and Lockheed Martin of the United States.

Contact Information

Marketing and Sales:
International Launch Services
1660 International Drive
McLean, VA 22102
USA
Phone: +1 (571) 633-7400
Fax: +1 (571) 633-7500
Web site: www.ilslaunch.com

General Description

Proton K/Block DM

Proton K

Summary

Proton was developed between 1961 and 1965 and is the largest operational space launch vehicle in nations of the former Soviet Union. A three-stage version is used to launch space station modules and other heavy payloads to LEO. The more common four-stage version using the Block DM upper stage was used for high-energy missions such as placing communications satellites into GTO, MEO, or directly into GEO or for launching planetary spacecraft on Earth escape trajectories. ILS no longer markets the Proton K/Block DM configuration, but Russia continues to use it for some domestic launches. Commercial launches have been shifted to the new Proton M/Breeze M.

Status

Operational. First launch in 1967 for Proton K and 1974 for Proton K/Block DM.

Origin

Russia

Key Organizations

Marketing Organization	International Launch Services
Launch Service Provider	International Launch Services
Prime Contractor	Khrunichev State Research and Production Space Center

Primary Missions

Three stage: space station deployment

Four stage: Heavy GEO payloads, planetary missions, GLONASS deployment

Estimated Launch Price

Price negotiable

Spaceport

Launch Site	Baikonur LC 81 Pads 23 and 24 and LC 200 Pad 39
Location	46.1° N, 63.0° E
Available Inclinations	51.6, 64.8, and 72.7 deg directly. All others require plane change maneuvers.

Performance Summary

Adapter mass must be subtracted to determine separated spacecraft mass. Performance reflects commercial configuration and flight profile. Maximum performance for Russian government missions is higher. See Performance section for more details.

200 km (108 nmi), 51.6 deg	19,760 kg (43,560 lbm)
185 km (100 nmi), 90 deg	3620 kg (7980 lbm)
Space Station Orbit: 186×222 km (100×120 nmi), 51.6 deg	19,760 kg (43,560 lbm). The three-stage Proton delivers spacecraft to low orbit, then the spacecraft must use onboard propulsion to raise the orbit to space station altitude.
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	3620 kg (7980 lbm)
GTO: 1500 m/sec Delta-V to 650	4350 kg (9590 lbm)—two upper-stage burns 4930 kg (10,846 lbm)—three upper-stage burns
Geostationary Orbit	1880 kg (4145 lbm)

Flight Record (through 31 December 2003)

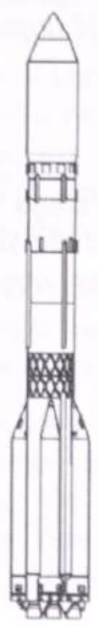
Total Flights	295
Launch Vehicle Successes	262
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	32

Flight Rate

4–14

General Description

Proton M/Breeze M



Proton M

Summary

The Proton M is an enhanced version of the Proton K vehicle. The first three stages are upgraded with improved avionics, lighter structures, and uprated first-stage engines. The residual propellant in jettisoned stages is also reduced to decrease Proton's environmental impact. The Block DM upper stage is replaced by a new Breeze M stage, which is derived from the Breeze K upper stage developed for the Rockot launch vehicle. The Breeze M includes additional tankage, which is jettisoned in flight, enabling increased performance to high-energy orbits such as GEO.

Status

Operational. First launch in 1999 for Breeze M and 2001 for Proton M.

Origin

Russia

Key Organizations

Marketing Organization	International Launch Services
Launch Service Provider	International Launch Services
Prime Contractor	Khrunichev State Research and Production Space Center

Primary Missions

Intermediate to heavy GTO and GEO payloads

Estimated Launch Price

Prices negotiable

Spaceport

Launch Site	Baikonur LC 81 Pads 23 and 24 and LC200 Pad 39
Location	46.1° N, 63.0° E
Available Inclinations	51.6, 64.8, and 72.7 deg directly. All others require plane change maneuvers.

Performance Summary

Adapter mass must be subtracted to determine separated spacecraft mass. Performance reflects commercial configuration and flight profile. Maximum performance for Russian government missions is higher. See Performance section for more details.

200 km (108 nmi), 51.6 deg	21,000 kg (46,300 lbm)
185 km (100 nmi), 90 deg	?
Space Station Orbit: 186×222 km (100×120 nmi), 51.6 deg	21,000 kg (46,300 lbm). The three-stage Proton delivers spacecraft to low orbit, then the spacecraft must use onboard propulsion to raise the orbit to space station altitude.
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	?
GTO: 1500 m/sec Delta-V to 650	5500 kg (12,100 lbm)
Geostationary Orbit	2920 kg (6440 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Launch Vehicle Successes	2
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

Planned at 8–10 per year.

Nomenclature

Following standard practice, the name Proton comes from the name of the vehicle's first payload. Proton was also referred to as UR-500 (series 500 of the Universal Rocket family) by its manufacturer. The initial two-stage version of Proton has been replaced by the three-stage Proton K, which is also available with the Block DM upper stage. The Proton K is now being replaced by the improved Proton M, which was the Breeze (or Briz) M upper stage.

Block DM Nomenclature

The upper stage of the Proton K is the Block DM. The term Block, also transliterated as Blok, translates roughly as stage. Because the stage was first used as the fifth stage of the N-1 lunar launch vehicle, it is given the letter D, the fifth letter of the Cyrillic alphabet. The letter M indicates that it has been modernized. Specifically, the Block DM has its own onboard avionics, while the older Block D, which is still used periodically for planetary missions, is controlled by the spacecraft above it. Several different versions of the Block DM are currently in use.

Name	Article Number	Description
Domestic		
Block D	11S824	Original configuration using 11D58 engine and controlled by payload
Block D-1	11S824M	Upgraded 11D58 engine, stage controlled by payload
Block D-2	11S824F	Similar to Block D-1
Block DM	11S86	Similar to Block D-1 with onboard avionics in toroidal instrument compartment
Block DM-2	11S861	Similar to Block DM with improved guidance
Block DM-2M	11S861-01	Similar to Block DM-2 but modified for heavier payloads through use of "syntin" synthetic hydrocarbon fuel
Block DM-5	17S40	Modified for heavier payloads with different interfaces than Block DM-2M
Commercial		
Block DM1	11S861	Same as Block DM-2 with Saab payload adapter–separation system for commercial payloads
Block DM2	17S40	Variation of Block DM-5 with Iridium dispenser
Block DM3	11S861-01	Variation of Block DM-2M with Saab payload adapter–separation system for commercial BSS601 payloads
Block DM4	11S861-01	Variation of Block DM-2M with Saab payload adapter–separation system for commercial FS-1300 payloads
Block DM-SL	?	Modification of Block DM upper stage used on Sea Launch Zenit; not used with Proton

Cost

The trends in Proton pricing are interesting in part because they may be indicators of the price trends of other Russian launch vehicles. Proton launch prices started out to attract customers to an unfamiliar launch vehicle. As Proton became more accepted by satellite operators, and as services improved to match Western practices, prices matched market conditions. Proton was first offered commercially in the mid 1980s by Glavkosmos, the Soviet agency responsible for marketing space services. Glavkosmos announced prices of $28–35 million for GTO or GEO launch services on Proton, but was unable to sign any launch contracts. Following the collapse of the Soviet Union, a few Western customers such as Inmarsat and Iridium began purchasing Proton launch services. These initial contracts were reportedly priced in the $35–50 million price range. As Proton became more established in the commercial market, and as Khrunichev began marketing Proton in partnership with Lockheed Martin through ILS, prices began to rise to roughly $60–75 million. This may also have been driven by Russia's anti-dumping agreements with the United States, which required that Proton prices not undercut those of other commercial launch services by more than 15%. According to Russian press reports, RSC Energia released prices for several contracts that took place during this period, including the 1997 launches of Telstar 5 and PanAmSat 5 for $47 million each, and the 1998 launch of Astra 2A for $57 million. With increasing demand for launch services, and with Proton now fully established in the commercial launch services market, prices continued to rise. According to filing documents with the U.S. Securities and Exchange Commission, the company Sirius Satellite Radio was offered prices from $90 to 95 million per launch for a multiple launch contract for launches occurring around 2000. ILS reported in 2000 that typical prices for Proton K/Block DM launch services were between $90–98 million, and prices for Proton M/Breeze M launches are typically $100–112 million.

Availability

The Soviet Union attempted unsuccessfully to sell Proton launch services beginning in early 1983 with an offer to Inmarsat. This prompted the United States to forbid U.S.-built payloads from being launched on Soviet vehicles because of technology transfer concerns. Moreover, the Soviets were hesitant to reveal previously secret designs and capabilities, making it difficult to attract customers. The situation improved in the early 1990s after the fall of the Soviet Union, when a few Western customers including Iridium and Inmarsat ordered Proton launches. In 1992 Lockheed formed a joint venture with Khrunichev and Energia, called LKE (for Lockheed–Khrunichev–Energia International), to market Proton launches commercially. Following the merger of Lockheed and Martin Marietta, LKE became a subsidiary of ILS, which markets both the Proton and Lockheed Martin's Atlas. The partnership with a Western aerospace company has helped Proton become a mainstream commercial launch vehicle, and it is now a common choice for satellite operators around the world. A system of international quotas was put in place in 1996 which limited the number of Russian launch vehicles that could be sold commercially and constrained how low they could be priced, in order to protect western launch vehicles from low cost Russian competitors. The quota system expired in 2000, and there are currently no such restrictions on quantity or price of Russian vehicles.

The Proton K/Block DM is no longer being marketed by ILS, which has transitioned to the Proton M/Breeze M. However, the Russian government will continue to use both Proton K and Proton M for some time. During the transition, some launches are using an interim configuration of the Proton K combined with the Breeze M.

Protons are currently produced at a maximum rate of 16 per year. Three to four of these are reserved for Russian government payloads, leaving 12 for commercial customers. In mid 1998, Khrunichev was authorized by the Russian government to increase production to meet growing commercial demand. The turnaround time per pad is approximately 21 days. A total of four pads are available for Proton launches, although only three are currently active. Two pads, numbers 23 and 24 at Launch Complex 81, and a third pad, number 39 at Launch Complex 200, are used for both commercial and Russian government launches.

Performance

Proton is launched from the Baikonur Cosmodrome in Kazakhstan. Because Baikonur is a landlocked launch site, specific drop zones are reserved for the impact of the first stage, second stage, and payload fairing. These drop zones are typically about 310 km (190 mi) downrange for the first stage, and 1985 km (1230 mi) downrange for the second stage and payload fairing. Proton is therefore constrained to fly along one of three launch azimuths that are aligned with these zones. The basic three-stage Proton does not have an upper-stage restart capability, and therefore cannot reach orbits with perigees higher than approximately 200 km (108 nmi). To reach higher orbits or other inclinations, Proton delivers the payload plus an upper stage to a standard 180–200 km (97–108 nmi) circular support orbit. The upper stage or spacecraft propulsion system then performs orbit raising and/or plane change maneuvers. (While the azimuth restrictions limit performance, this approach does have the advantage of standardizing the trajectory and flight software of the lower three stages.)

Historically the Block DM upper stage has been used for most Proton K flights. Performance of the three-stage Proton configurations to typical support orbits is provided subsequently. Performance varies depending on the fairing separation time. Two separation times are available, each of which meets impact point restrictions. Payload fairing separation at 185 s results in higher performance than the 350 s separation time, but also exposes the payload to heating rates much higher than are standard for Western spacecraft. An intermediate time will be used by Proton M to enable full performance for Western satellite missions. Performance for Russian government missions, which may use smaller payload fairings and the early fairing jettison time, can be higher than the commercial advertised performance. A comparison is shown below.

	Proton K/Block DM		**Proton M/Breeze M**	
	Commercial	*Russian Government*	*Commercial*	*Russian Government*
200 km (108 nmi) 51.6°	19,760 kg (43,560 lbm)	20,700 kg (45,450 lbm)	21,000 kg (46,300 lbm)	22,000 kg (48,400 lbm)
GSO	1880 kg (4145 lbm)	2100 kg (4630 lbm)	2920 kg (6440 lbm)	3200 kg (7040 lbm)

The Block DM is a LOX/kerosene-fueled stage with a capability for up to seven engine restarts (five have been demonstrated) and an on-orbit lifetime of 24 h, or longer with additional modifications. This allows the Block DM to deliver payloads directly to GSO or to coast for several revolutions in the support orbit for proper phasing and then deliver a spacecraft to GSO close to its planned longitude. Russian domestic satellites have generally been delivered directly to GSO, but commercial spacecraft are typically delivered to a GTO with the apogee and perigee optimized for the particular mission. For certain GTO orbits, a three-burn trajectory can be used in which the Block DM stage is delivered to a suborbital trajectory and performs its first burn to reach the parking orbit. This results in higher performance as shown subsequently. Performance is increased for certain domestic Russian missions by using the lighter Block D2 stage, which lacks the avionics compartment of the Block DM. This results in a performance improvement of approximately 800 kg (1750 lbm), but requires that the spacecraft be capable of controlling the stage. This configuration is used primarily for Russian planetary spacecraft.

The Block DM is also used for many Proton LEO missions (for example, on three Iridium deployment flights), to deliver the payload to the desired inclination and altitude. However, the Block DM has propellant loading constraints that impact its performance for LEO and other low-energy missions. A minimum of 11,500 kg (25,350 lbm) of propellant must be loaded onto the Block DM, even for missions that do not require this much capability from the stage. As a result of carrying this surplus propellant, the available payload mass is limited and is largely independent of altitude for LEO missions. Missions to higher altitudes simply burn more of the propellant load, leaving a smaller surplus.

The new Breeze M is a modified version of the Breeze KM upper stage developed for the Rockot space launch vehicle. Like the Block DM, the Breeze M is capable of 24 h of on-orbit operations and multiple restarts. However, the Breeze M main engine has a much lower thrust (and therefore a longer burn time) than the Block DM. As a result, the apogee raising maneuver for GTO missions is usually, but not always, split into two burns. The first puts the Breeze M and payload into an intermediate elliptical orbit, and the second raises the apogee to the planned altitude.

Performance values shown are for commercial payloads. The standard commercial payload fairing is assumed for Block DM missions. The short version of the Breeze M fairing is assumed. The larger Breeze M payload fairing reduces payload capacity by roughly 100 kg (220 lbm) for GTO missions. The later of two available fairing separation times is used to keep free molecular heating rates at acceptable levels for commercial payloads. Domestic Russian payloads typically use smaller fairings, and in some cases separate the payload fairing earlier, resulting in higher performance than shown here.

Performance

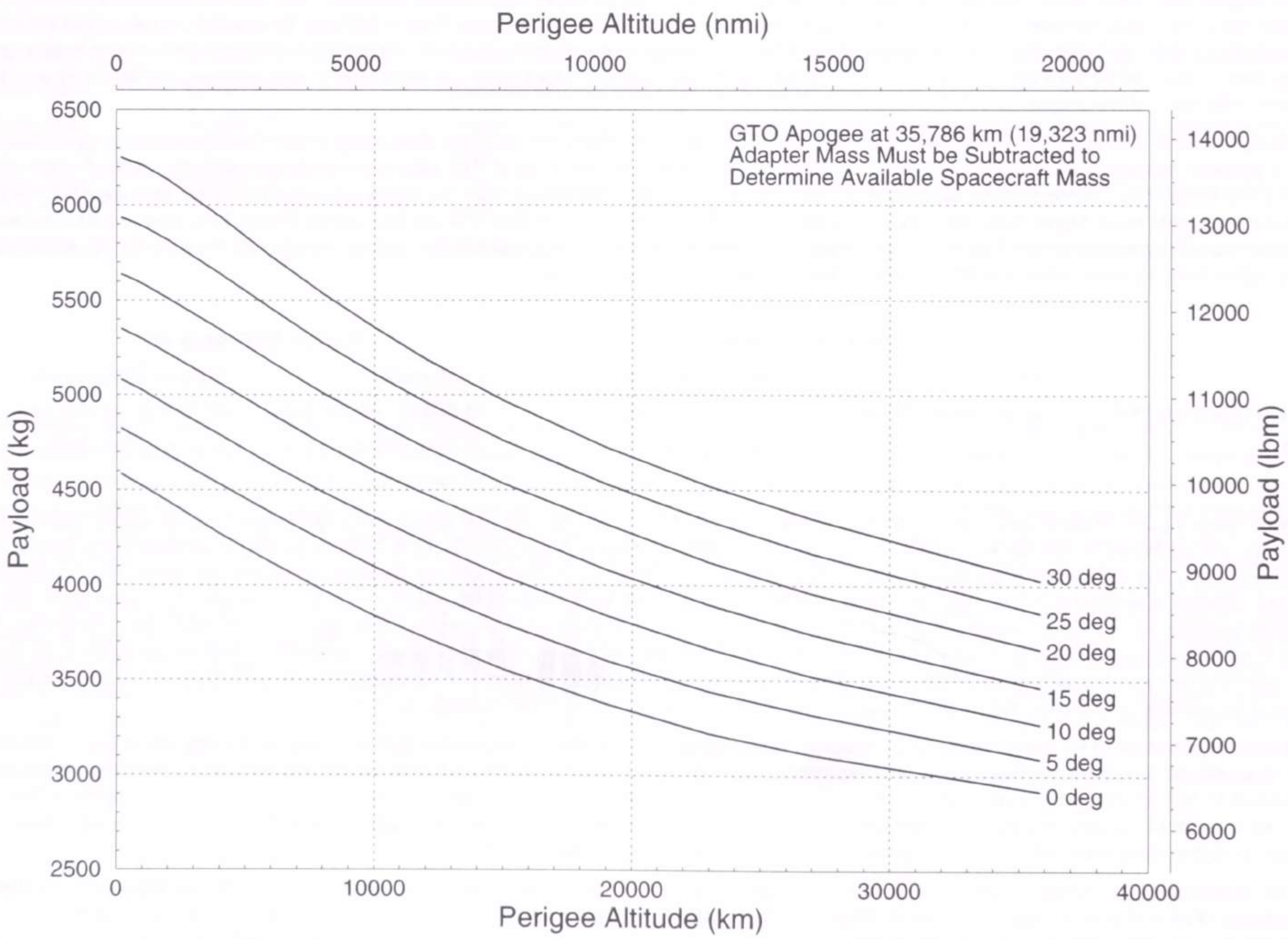
The performance shown is the payload systems mass, which includes both the spacecraft mass and the mass of the payload adapter and any additional payload support hardware such as cable harnesses. Typical mass for Block DM adapters is 120–175 kg (265–385 lbm). Typical mass for Breeze M adapters is 110 kg (240 lbm). Sufficient flight performance reserve is included to provide 3 sigma confidence of performance levels.

Three-Stage Proton Performance

Orbit	Proton K	Proton M
200 km (108 nmi), 51.6 deg	19,760 kg (43,560 lbm)	21,000 kg (46,300 lbm)
190 km (103 nmi), 64.8 deg	19,300 kg (42,550 lbm)	20,610 kg (45,435 lbm)
170 km (92 nmi), 72.7 deg	18,900 kg (41,670 lbm)	19,975 kg (44,035 lbm)

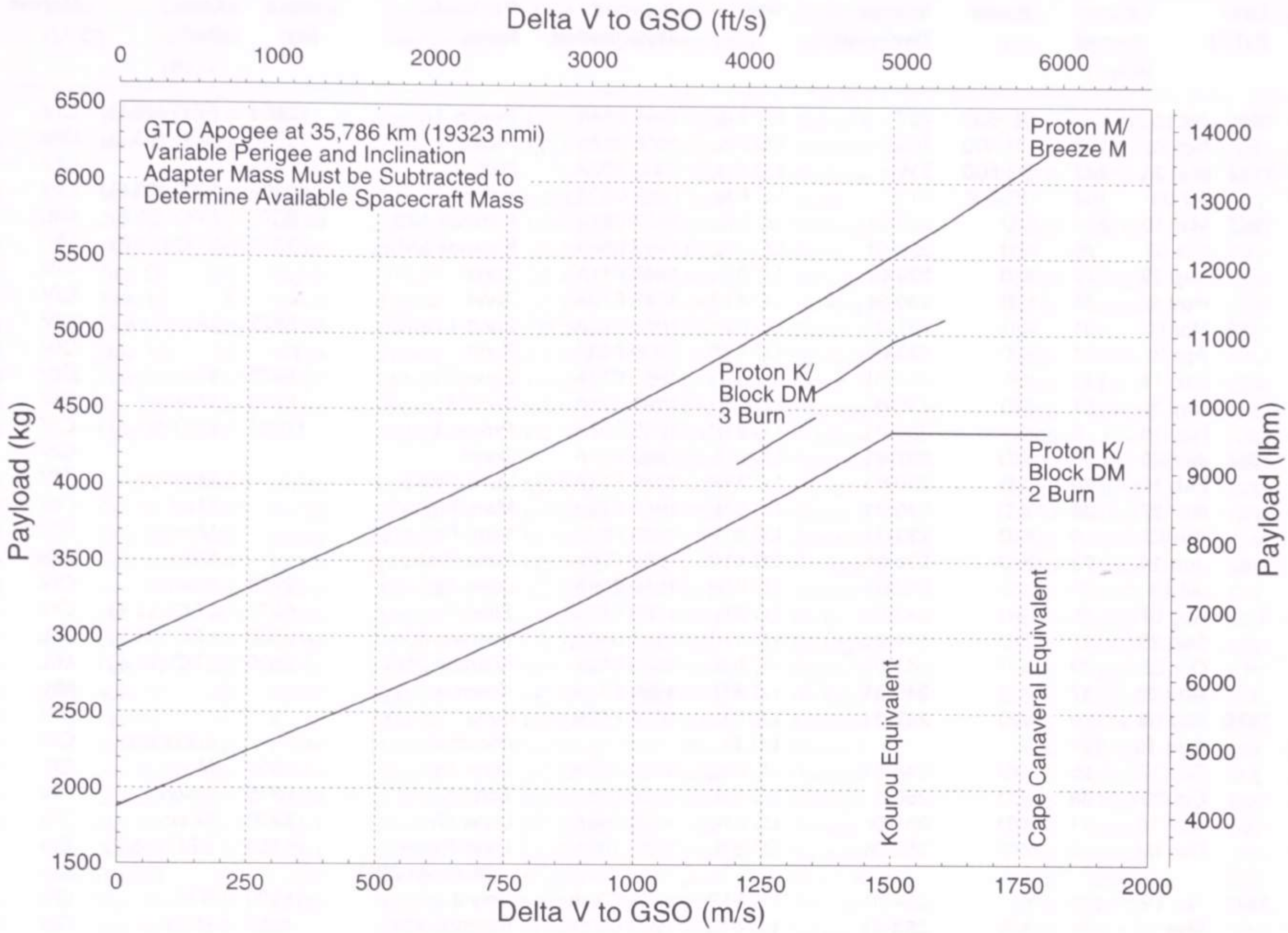
Proton K/Block DM LEO Performance (constant over altitude range listed)

Inclination Range	Payload Capability	Altitude Range for Constant Payload Mass
28.6–63.4 deg	5000 kg (11,000 lbm)	< 5000 km (2700 nmi)
72.7 deg to sun-synchronous orbit	3720 kg (8200 lbm)	< 18,000 km (9700 nmi) for 72.7 deg < 1200 km (650 nmi) for SSO



Proton K/Breeze M: Performance for GTO Orbits (2 upper-stage burns)

PERFORMANCE

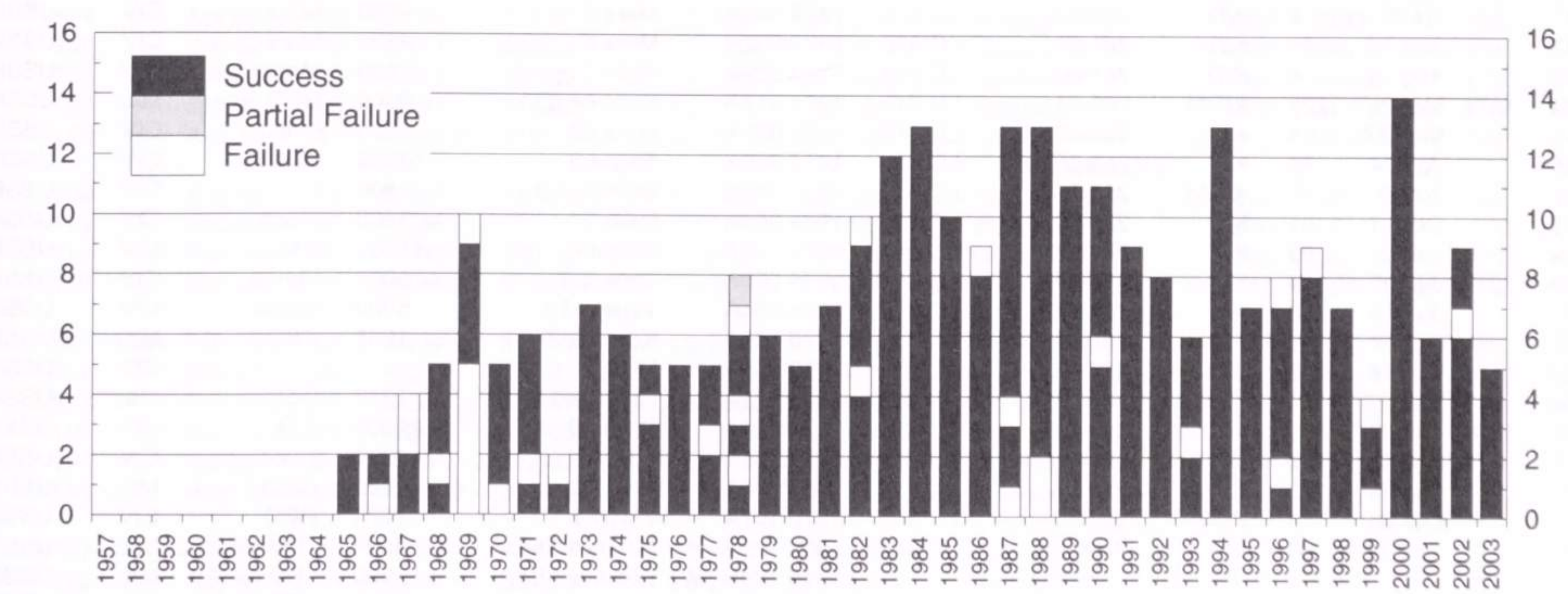


Proton K and Proton M: GTO Performance vs Delta V to GSO

FLIGHT HISTORY

Proton flights before 1970 were considered development and test flights. Note that Proton performed one suborbital launch in 1970, which is listed in the table below for reference, but is not counted in the total number of orbital flights.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Proton K/D or DM	Proton K/Breeze M	Proton K total	Proton M/Breeze M	Combined Proton Family
Total Orbital Flights	263	3	295	2	302
Launch Vehicle Successes	235	2	262	2	268
Launch Vehicle Partial Failures	1	0	1	0	1
Launch Vehicle Failures	27	1	32	0	33

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1	1965 Jul 16	—	UR-500	207	LC 81L	1965 054A	Proton 1	12200	EEO (63.5)	CIV	USSR
	2	Nov 02	109	UR-500	209	LC 81L	1965 087A	Proton 2	12000	EEO (63.5)	CIV	USSR
F	**3**	**1966 Mar 24**	**142**	**UR-500**	**211**	**LC 81L**	1966 F03A	Proton			CIV	USSR
	4	Jul 06	104	UR-500	212	LC 81L	1966 060A	Proton 3	12200	EEO (63.5)	CIV	USSR
	5	1967 Mar 10	247	K/D	227-01	LC 81L	1967 021A	Kosmos 146	5375	LEO (51.5)	MIL	USSR
	6	Apr 08	29	K/D	228-01	LC 81L	1967 032A	Kosmos 154	5375	LEO (51.6)	MIL	USSR
F	**7**	**Sep 27**	**172**	**K/D**	**229-01**	**LC 81L**	1967 F11A	Zond			CIV	USSR
F	**8**	**Nov 22**	**56**	**K/D**	**230-01**	**LC 81P**	1967 F12A	Zond			CIV	USSR
	9	1968 Mar 02	101	K/D	231-01	LC 81L	1968 013A	Zond 4	5375	Moon	CIV	USSR
F	**10**	**Apr 22**	**51**	**K/D**	**232-01**	**LC 81P**	1968 F03A	Zond			CIV	USSR
	11	Sep 14	145	K/D	234-01	LC 81L	1968 076A	Zond 5	5375	Moon	CIV	USSR
	12	Nov 10	57	K/D	235-01	LC 81L	1968 101A	Zond 6	5375	Moon	CIV	USSR
	13	Nov 16	6	K	236-01	LC 81P	1968 103A	Proton 4	17000	LEO (51.5)	CIV	USSR
F	**14**	**1969 Jan 20**	**65**	**K/D**	**237-01**	**LC 81L**	1969 F01A	Zond			CIV	USSR
F	**15**	**Feb 19**	**30**	**K/D**	**239-01**	**LC 81P**	1969 F04A	Luna Probe		Moon	CIV	USSR
F	**16**	**Mar 27**	**36**	**K/D**	**240-01**	**LC 81L**	1969 F06A	Mars Probe		Mars	CIV	USSR
F	**17**	**Apr 02**	**6**	**K/D**	**233-01**	**LC 81P**	1969 F07A	Mars Probe		Mars	CIV	USSR
F	**18**	**Jun 14**	**73**	**K/D**	**238-01**	**LC 81P**	1969 F08A	Luna Probe		Moon	CIV	USSR
	19	Jul 13	29	K/D	242-01	LC 81P	1969 058A	Luna 15	2718	Moon	CIV	USSR
	20	Aug 07	25	K/D	243-01	LC 81L	1969 067A	Zond 7	5979	LEO (51.4)	CIV	USSR
	21	Sep 23	47	K/D	244-01	LC 81P	1969 080A	Kosmos 300	5600	LEO (51.5)	MIL	USSR
	22	Oct 22	29	K/D	241-01	LC 81P	1969 092A	Kosmos 305	5600	LEO (51.5)	MIL	USSR
F	**23**	**Nov 28**	**37**	**K/D**	**245-01**	**LC 81L**	1969 F14A	Kosmos			MIL	USSR
F	**24**	**1970 Feb 06**	**70**	**K/D**	**247-01**	**LC 81L**	1970 F02A	Luna			CIV	USSR
F	**—**	**Aug 18**	**193**	**K**		**LC 81**		unnamed		suborbital	CIV	USSR
	25	Sep 12	25	K/D	248-01	LC 81L	1970 072A	Luna 16	5600	Moon	CIV	USSR
	26	Oct 20	38	K/D	250-01	LC 81L	1970 088A	Zond 8	5375	Moon	CIV	USSR
	27	Nov 10	21	K/D	251-01	LC 81L	1970 095A	Luna 17	5600	Moon	CIV	USSR
	28	Dec 02	8	K/D	252-01	LC 81L	1970 103A	Luna Probe (Kosmos 382)	10380	EEO (55.9)	CIV	USSR
	29	1971 Apr 19	138	K	254-01	LC 81P	1971 032A	Salyut 1	18500	STA	CIV	USSR
F	**30**	**May 10**	**21**	**K/D**	**253-01**	**LC 81L**	1971 042A	Kosmos 419 (Mars Probe)	4650	Mars	CIV	USSR
	31	May 19	9	K/D	255-01	LC 81P	1971 045A	Mars 2	4650	Mars	CIV	USSR
	32	May 28	9	K/D	249-01	LC 81L	1971 049A	Mars 3	4643	Mars	CIV	USSR
	33	Sep 02	97	K/D	256-01	LC 81P	1971 073A	Luna 18	5600	Moon	CIV	USSR
	34	Sep 28	26	K/D	257-01	LC 81P	1971 082A	Luna 19	5600	Moon	CIV	USSR
	35	1972 Feb 14	139	K/D	258-01	LC 81P	1972 007A	Luna 20	5600	Moon	CIV	USSR
F	**36**	**Jul 29**	**166**	**K**	**260-01**	**LC 81L**	1972 F04A	Salyut	18000	STA	CIV	USSR
	37	1973 Jan 08	163	K/D	259-01	LC 81L	1973 001A	Luna 21	4850	Moon	CIV	USSR
P	**38**	**Apr 03**	**85**	**K**	**283-01**	**LC 81L**	1973 017A	Salyut 2	18500	LEO (51.6)	CIV	USSR
S	39	May 11	38	K	284-01	LC 81L	1973 026A	Salyut (Kosmos 557)	19400	STA	MIL	USSR
	40	Jul 21	71	K/D	261-01	LC 81L	1973 047A	Mars 4	4650	Mars	CIV	USSR
	41	Jul 25	4	K/D	262-01	LC 81P	1973 049A	Mars 5	4650	Mars	CIV	USSR
	42	Aug 05	11	K/D	281-01	LC 81L	1973 052A	Mars 6	4650	Mars	CIV	USSR
	43	Aug 09	4	K/D	281-02	LC 81P	1973 053A	Mars 7	4650	Mars	CIV	USSR
	44	1974 Mar 26	229	K/DM	282-01	LC 81L	1974 017A	Kosmos 637	2000	GEO	MIL	USSR
	45	May 29	64	K/D	282-02	LC 81P	1974 037A	Luna 22	4000	Moon	CIV	USSR
	46	Jun 24	26	K	283-02	LC 81L	1974 046A	Salyut 3	18500	STA	CIV	USSR
	47	Jul 29	35	K/DM	287-01	LC 81P	1974 060A	Molniya 1S	1600		CIV	USSR
	48	Oct 28	91	K/D	285-01	LC 81P	1974 084A	Luna 23	5600	Moon	CIV	USSR
	49	Dec 26	59	K	284-02	LC 81P	1974 104A	Salyut 4	18900	STA	CIV	USSR
	50	1975 Jun 08	164	K/D	286-01	LC 81P	1975 050A	Venera 9	5000	Venus	CIV	USSR
	51	Jun 14	6	K/D	285-02	LC 81P	1975 054A	Venera 10	5000	Venus	CIV	USSR
	52	Oct 08	116	K/DM	286-02	LC 81L	1975 097A	Kosmos 775	1965	GEO	MIL	USSR
F	**53**	**Oct 16**	**8**	**K/D**	**287-02**	**LC 81L**	1975 F05A	Luna			CIV	USSR
	54	Dec 22	67	K/DM	288-01	LC 81P	1975 123A	Raduga 1	1965	GEO	MIL	USSR
	55	1976 Jun 22	183	K	290-02	LC 81L	1976 057A	Salyut 5	19000	STA	CIV	USSR
	56	Aug 09	48	K/D-1	288-02	LC 81L	1976 081A	Luna 24	4800	Moon	CIV	USSR
	57	Sep 11	33	K/DM	289-01	LC 81P	1976 092A	Raduga 2	1965	GEO	MIL	USSR
	58	Oct 26	45	K/DM	290-01	LC 81P	1976 107A	Ekran 1	1970	GEO	CIV	USSR
	59	Dec 15	50	K	289-02	LC 81P	1976 121A M	Kosmos 881	6000	LEO (51.6)	MIL	USSR
							1976 121B M	Kosmos 882	6000	LEO (51.6)	MIL	USSR
	60	1977 Jul 17	214	K	293-02	LC 81P	1977 066A	Kosmos 929	19000	LEO (51.5)	MIL	USSR
	61	Jul 23	6	K/DM	291-01	LC 200P	1977 071A	Raduga 3	1965	GEO	MIL	USSR
F	**62**	**Aug 04**	**12**	**K**	**293-01**	**LC 81P**	1977 F02A	Kosmos			MIL	USSR
	63	Sep 20	47	K/DM	291-02	LC 200P	1977 092A	Ekran 2	1970	GEO	CIV	USSR

LC-81 and LC-200 are the Proton launch complexes at Baikonur. LC-81L (left) is Pad 23, LC-81P (right) is Pad 24. LC-200L is Pad 39, LC-200 P is Pad 40.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	64		Sep 29	9	K	295-01	LC 81P	1977 097A		Salyut 6	9000	LEO (51.6)	CIV	USSR
	65	1978	Mar 30	182	K	292-01	LC 81P	1978 032A	M	Kosmos 997	700	LEO (51.6)	MIL	USSR
								1978 032B	M	Kosmos 998	7000	LEO (51.6)	MIL	USSR
F	**66**		**May 27**	**58**	**K/DM**	**294-02**	**LC 200P**	1978 F02A		Ekran			CIV	USSR
	67		Jul 18	52	K/DM	292-02	LC 200P	1978 073A		Raduga 4	1965	GEO	MIL	USSR
F	**68**		**Aug 17**	**30**	**K/DM**	**297-02**	**LC 200P**	1978 F03A		Ekran			CIV	USSR
	69		Sep 09	23	K/D-1	296-01	LC 81L	1978 084A		Venera 11	4940	Venus	CIV	USSR
	70		Sep 14	5	K/D-1	296-02	LC 81P	1978 086A		Venera 12	4940	Venus	CIV	USSR
F	**71**		**Oct 17**	**33**	**K/DM**	**298-01**	**LC 200P**	1978 F04A		Ekran			CIV	USSR
P	**72**		**Dec 19**	**63**	**K/DM**	**295-02**	**LC 200P**	1978 118A		Gorizont 1	2120	GEO	CIV	USSR
	73	1979	Feb 21	64	K/DM	294-01	LC 200P	1979 015A		Ekran 3	1970	GEO	CIV	USSR
	74		Apr 25	63	K/DM	298-02	LC 200P	1979 035A		Raduga 5	1965	GEO	MIL	USSR
	75		May 22	27	K	300-02	LC 81P	1979 042A	M	Kosmos 1100	9000	LEO (51.6)	MIL	USSR
								1979 042B	M	Kosmos 1101	9000	LEO (51.6)	MIL	USSR
	76		Jul 05	44	K/DM	299-01	LC 200P	1979 062A		Gorizont 2	2120	GEO	CIV	USSR
	77		Oct 03	90	K/DM	302-02	LC 200P	1979 087A		Ekran 4	1970	GEO	CIV	USSR
	78		Dec 28	86	K/DM	303-01	LC 200P	1979 105A		Gorizont 3	2120	GEO	CIV	USSR
	79	1980	Feb 20	54	K/DM	297-01	LC 200L	1980 016A		Raduga 6	1965	GEO	MIL	USSR
	80		Jun 14	115	K/DM	303-02	LC 200L	1980 049A		Gorizont 4	2120	GEO	CIV	USSR
	81		Jul 14	30	K/DM	301-01	LC 200P	1980 060A		Ekran 5	1970	GEO	CIV	USSR
	82		Oct 05	83	K/DM	300-01	LC 200L	1980 081A		Raduga 7	1965	GEO	MIL	USSR
	83		Dec 26	82	K/DM	304-01	LC 200P	1980 104A		Ekran 6	1970	GEO	CIV	USSR
	84	1981	Mar 18	82	K/DM	306-01	LC 200P	1981 027A		Raduga 8	1965	GEO	MIL	USSR
	85		Apr 24	37	K	299-02	LC 200L	1981 039A	D	Kosmos 1267	15100	STA	MIL	USSR
	86		Jun 25	62	K/DM	305-01	LC 200P	1981 061A		Ekran 7	1970	GEO	CIV	USSR
	87		Jul 30	35	K/DM	301-02	LC 200L	1981 069A		Raduga 9	1965	GEO	MIL	USSR
	88		Oct 09	71	K/DM	310-01	LC 200L	1981 102A		Raduga 10	1965	GEO	MIL	USSR
	89		Oct 30	21	K/D-1	311-01	LC 200P	1981 106A		Venera 13	2460	Venus	CIV	USSR
	90		Nov 04	5	K/D-1	311-02	LC 200L	1981 110A		Venera 14	2460	Venus	CIV	USSR
	91	1982	Feb 05	93	K/DM	308-01	LC 200P	1982 009A		Ekran 8	1970	GEO	CIV	USSR
	92		Mar 15	38	K/DM	305-02	LC 200L	1982 020A		Gorizont 5	2500	GEO	CIV	USSR
	93		Apr 19	35	K	306-02	LC 200P	1982 033A		Salyut 7	18900	STA	CIV	USSR
								1982 033C	A	Iskra 2			CIV	USSR
	94		May 17	28	K/DM	310-02	LC 200L	1982 044A		Kosmos 1366	1965	GEO	MIL	USSR
F	**95**		**Jul 22**	**66**	**K/DM**	**307-02**	**LC 200P**	1982 F04A		Ekran		GEO	CIV	USSR
	96		Sep 16	56	K/DM	309-01	LC 200P	1982 093A		Ekran 9	1970	GEO	CIV	USSR
	97		Oct 12	26	K/DM-2	315-01	LC 200L	1982 100	M	Kosmos 1413–1415 (Glonass)	3@1260	MEO (64.7)	MIL	USSR
	98		Oct 20	8	K/DM	312-01	LC 200P	1982 103A		Gorizont 6	2500	GEO	CIV	USSR
	99		Nov 26	37	K/DM	313-01	LC 200L	1982 113A		Raduga 11	1965	GEO	MIL	USSR
F	**100**		**Dec 24**	**28**	**K/DM**	**314-01**	**LC 200L**	1982 F08A		Raduga		GEO	MIL	USSR
	101	1983	Mar 02	68	K	309-02	LC 200L	1983 013A	D	Kosmos 1443	20000	STA	MIL	USSR
	102		Mar 12	10	K/DM	304-02	LC 200P	1983 016A		Ekran 10	1970	GEO	CIV	USSR
	103		Mar 23	11	K/D-1	307-01	LC 200L	1983 020A		Astron	3250	EEO (79.8)	CIV	USSR
	104		Apr 08	16	K/DM	315-02	LC 200P	1983 028A		Raduga 12	1965	GEO	MIL	USSR
	105		Jun 02	55	K/D-1	321-01	LC 200L	1983 053A		Venera 15	4000	Venus	CIV	USSR
	106		Jun 07	5	K/D-1	321-02	LC 200P	1983 054A		Venera 16	4000	Venus	CIV	USSR
	107		Jun 30	23	K/DM	314-02	LC 200L	1983 066A		Gorizont 17	2500	GEO	CIV	USSR
	108		Aug 10	41	K/DM-2	317-01	LC 200L	1983 084	M	Kosmos 1490–1492 (Glonass)	3@1260	MEO (64.7)	MIL	USSR
	109		Aug 25	15	K/DM	316-02	LC 200P	1983 088A		Raduga 13	1965	GEO	MIL	USSR
	110		Sep 29	35	K/DM	318-01	LC 200P	1983 100A		Ekran 11	1970	GEO	CIV	USSR
	111		Nov 30	62	K/DM	308-02	LC 200L	1983 118A		Gorizont 8	2500	GEO	CIV	USSR
	112		Dec 29	29	K/DM-2	320-02	LC 200P	1983 127	M	Kosmos 1519–1521 (Glonass)	3@1260	MEO (65.3)	MIL	USSR
	113	1984	Feb 15	48	K/DM	318-02	LC 200L	1984 016A		Raduga 16	1965	GEO	MIL	USSR
	114		Mar 02	16	K/DM	316-01	LC 200P	1984 022A		Kosmos 1540	1965	GEO	MIL	USSR
	115		Mar 16	14	K/DM	322-01	LC 200L	1984 028A		Ekran 12	1970	GEO	CIV	USSR
	116		Mar 29	13	K/DM	319-02	LC 200P	1984 031A		Kosmos 1546	1965	GEO	MIL	USSR
	117		Apr 22	24	K/DM	312-02	LC 200L	1984 041A		Gorizont 9	2500	GEO	CIV	USSR
	118		May 19	27	K/DM-2	323-02	LC 200P	1984 047	M	Kosmos 1554–1556 (Glonass)	3@1260	MEO (65.5)	MIL	USSR
	119		Jun 22	34	K/DM	319-01	LC 200L	1984 063A		Raduga 15	1965	GEO	MIL	USSR
	120		Aug 01	40	K/DM	324-01	LC 200P	1984 078A		Gorizont 10	2500	GEO	CIV	USSR
	121		Aug 24	23	K/DM	324-02	LC 200L	1984 090A		Ekran 13	1970	GEO	CIV	USSR
	122		Sep 04	11	K/DM-2	320-01	LC 200P	1984 095	M	Kosmos 1593–1595 (Glonass)	3@1260	MEO (64.8)	MIL	USSR
	123		Sep 28	24	K/DM-2	327-02	LC 200L	1984 106A		Kosmos 1603	3250	LEO (71)	MIL	USSR

LC-81 and LC-200 are the Proton launch complexes at Baikonur. LC-81L (left) is Pad 23, LC-81P (right) is Pad 24. LC-200L is Pad 39, LC-200 P is Pad 40.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	124		Dec 15	78	K/D-1	329-01	LC 200L	1984 125A		Vega 2	4000	Comet Halley	CIV	USSR
	125		Dec 21	6	K/D-1	325-01	LC 200P	1984 128A		Vega 1	4000	Comet Halley	CIV	USSR
	126	1985	Jan 18	28	K/DM	326-02	LC 200L	1985 007A		Gorizont 11	2500	GEO	CIV	USSR
	127		Feb 21	34	K/DM	327-01	LC 200L	1985 016A		Kosmos 1629	1965	GEO	MIL	USSR
	128		Mar 22	29	K/DM	328-01	LC 200P	1985 024A		Ekran 14	1970	GEO	CIV	USSR
	129		May 17	56	K/DM-2	330-02	LC 200L	1985 037	M	Kosmos 1650–1652 (Glonass)	3@1260	MEO (64.8)	MIL	USSR
	130		May 30	13	K/DM-2	313-02	LC 200P	1985 042A		Kosmos 1656	3250	LEO (71.1)	MIL	USSR
	131		Aug 08	70	K/DM	317-02	LC 200L	1985 070A		Raduga 16	1965	GEO	MIL	USSR
	132		Sep 27	50	K	331-01	LC 200L	1985 086A	D	Kosmos 1686 (Star 4)	20000	STA	MIL	USSR
	133		Oct 25	28	K/DM-2	332-02	LC 200P	1985 102A		Kosmos 1700	1965	GEO	MIL	USSR
	134		Nov 15	21	K/DM	326-01	LC 200L	1985 107A		Raduga 17	1965	GEO	MIL	USSR
	135		Dec 24	39	K/DM-2	334-02	LC 200L	1985 118	M	Kosmos 1710–1712 (Glonass)	3@1260	MEO (65.2)	MIL	USSR
	136	1986	Jan 17	24	K/DM	331-02	LC 200P	1986 007A		Raduga 18	1965	GEO	MIL	USSR
	137		Feb 19	33	K	337-01	LC 200L	1986 017A		Mir	20100	STA	CIV	USSR
	138		Apr 04	44	K/DM	302-01	LC 200P	1986 027A		Kosmos 1738	1965	GEO	MIL	USSR
	139		May 24	50	K/DM	333-01	LC 200L	1986 038A		Ekran 15	1970	GEO	CIV	USSR
	140		Jun 10	17	K/DM	322-02	LC 200P	1986 044A		Gorizont 12	2500	GEO	CIV	USSR
	141		Sep 16	98	K/DM-2	336-01	LC 200P	1986 071	M	Kosmos 1778–1880 (Glonass)	3@1260	MEO (64.8)	MIL	USSR
	142		Oct 25	39	K/DM	335-02	LC 200P	1986 082A		Raduga 19	1965	GEO	MIL	USSR
	143		Nov 18	24	K/DM	334-01	LC 200L	1986 090A		Gorizont 13	2500	GEO	CIV	USSR
F	**144**		**Nov 29**	**11**	**K**	**338-01**	**LC 200P**	1986 F07A		Kosmos			MIL	USSR
F	**145**	**1987**	**Jan 30**	**62**	**K/DM-2**	**341-01**	**LC 200P**	1987 010A		Kosmos 1817	1965	LEO (51.6)	MIL	USSR
	146		Mar 19	48	K/DM	323-01	LC 200P	1987 028A		Raduga 20	1965	GEO	MIL	USSR
	147		Mar 31	12	K	336-02	LC 200L	1987 030A	D	Kvant 1	20000	STA	CIV	USSR
F	**148**		**Apr 24**	**24**	**K/DM-2**	**335-01**	**LC 200P**	1987 036	M	Kosmos 1838–1840 (Glonass)	3@1260	MEO (64.9)	MIL	USSR
	149		May 11	17	K/DM	338-02	LC 200L	1987 040A		Gorizont 14	2500	GEO	CIV	USSR
	150		Jul 25	75	K	347-01	LC 200P	1987 064A		Kosmos 1870	20000	LEO (71.9)	MIL	USSR
	151		Sep 03	40	K/DM	337-02	LC 200L	1987 073A		Ekran 16	1970	GTO	CIV	USSR
	152		Sep 16	13	K/DM-2	339-02	LC 200P	1987 079	M	Kosmos 1883–1885 (Glonass)	3@1260	MEO (64.9)	MIL	USSR
	153		Oct 01	15	K/DM-2	328-02	LC 200L	1987 084A		Kosmos 1888	1965	GEO	MIL	USSR
	154		Oct 28	27	K/DM-2	325-01	LC 200P	1987 091A		Kosmos 1894	1965	GEO	MIL	USSR
	155		Nov 26	29	K/DM-2	330-01	LC 200L	1987 096A		Kosmos 1897	1965	GEO	MIL	USSR
	156		Dec 10	14	K/DM-2	343-01	LC 200P	1987 100A		Raduga 21	1965	GEO	MIL	USSR
	157		Dec 27	17	K/DM-2	345-01	LC 200L	1987 109A		Ekran 17	1970	GEO	CIV	USSR
F	**158**	**1988**	**Jan 18**	**22**	**K/DM-2**	**341-02**	**LC 200P**	1988 F01A		Gorizont		GEO	CIV	USSR
F	**159**		**Feb 17**	**30**	**K/DM-2**	**346-02**	**LC 200L**	1988 009	M	Kosmos 1917–1619 (Glonass)	3@1260	LEO (64.8)	MIL	USSR
	160		Mar 31	43	K/DM	343-02	LC 200P	1988 028A		Gorizont 15	2500	GEO	CIV	USSR
	161		Apr 26	26	K/DM-2	332-01	LC 200L	1988 034A		Kosmos 1940	1065	GEO	MIL	USSR
	162		May 06	10	K/DM	349-01	LC 200L	1988 036A		Ekran 18	1970	GEO	CIV	USSR
	163		May 21	15	K/DM-2	348-01	LC 200L	1988 043	M	Kosmos 1946–1948 (Glonass)	3@1260	MEO (64.9)	MIL	USSR
	164		Jul 07	47	K/D-2	356-02	LC 200L	1988 057A		Fobos 1	6220	Mars	CIV	USSR
	165		Jul 12	5	K/D-2	356-01	LC 200P	1988 059A		Fobos 2	6220	Mars	CIV	USSR
	166		Aug 01	20	K/DM-2	351-01	LC 200L	1988 066A		Kosmos 1961	1965	GEO	MIL	USSR
	167		Aug 18	17	K/DM-2	333-02	LC 200P	1988 071A		Gorizont 16	2500	GEO	CIV	USSR
	168		Sep 16	29	K/DM-2	349-02	LC 200L	1988 085	M	Kosmos 1970 (Glonass)	3@1415	MEO (64.8)	MIL	USSR
	169		Oct 20	34	K/DM-2	339-01	LC 200L	1988 095A		Raduga 22	1965	GEO	MIL	USSR
	170		Dec 10	51	K/DM-2	329-02	LC 200P	1988 108A		Ekran 19	1970	GEO	CIV	USSR
	171	1989	Jan 10	31	K/DM-2	350-02	LC 200L	1989 001	M	Kosmos 1987–1989 (Glonass)	3@1415	MEO (64.9)	MIL	USSR
	172		Jan 26	16	K/DM-2	351-02	LC 200P	1989 004A		Gorizont 17	2500	GEO	CIV	USSR
	173		Apr 14	78	K/DM-2	359-02	LC 200L	1989 030A		Raduga 23	1965	GEO	MIL	USSR
	174		May 31	47	K/DM-2	352-02	LC 200P	1989 039	M	Kosmos 2022–2024 (Glonass)	3@1415	MEO (64.8)	MIL	USSR
	175		Jun 21	21	K/DM-2	355-02	LC 200L	1989 048A		Raduga 1-1	1965	GEO	MIL	USSR
	176		Jul 05	14	K/DM-2	340-02	LC 200P	1989 052A		Gorizont 18	2500	GEO	CIV	USSR
	177		Sep 28	85	K/DM-2	346-01	LC 200P	1989 081A		Gorizont 19	2500	GEO	CIV	USSR
	178		Nov 26	59	K	354-01	LC 200L	1989 093A	D	Kvant 2	19565	STA	CIV	USSR
	179		Dec 01	5	K/D-1	352-01	LC 200P	1989 096A		Granat	3500	EEO (67.1)	CIV	USSR
	180		Dec 15	14	K/DM-2	344-01	LC 81L	1989 098A		Raduga 24	1965	GEO	MIL	USSR

LC-81 and LC-200 are the Proton launch complexes at Baikonur. LC-81L (left) is Pad 23, LC-81P (right) is Pad 24. LC-200L is Pad 39, LC-200 P is Pad 40.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	181		Dec 27	12	K/DM-2	347-02	LC 200L	1989 101A		Kosmos 2054	2150	GEO	MIL	USSR
	182	1990	Feb 15	50	K/DM-2	363-02	LC 81L	1990 016A		Raduga 25	1965	GEO	MIL	USSR
	183		May 19	93	K/DM-2	350-01	LC 200P	1990 045	M	Kosmos 2079–2081 (Glonass)	3@1415	MEO (65)	MIL	USSR
	184		May 31	12	K	360-01	LC 200L	1990 048A	D	Kristall	19640	STA	CIV	USSR
	185		Jun 20	20	K/DM	342-02	LC 200P	1990 054A		Gorizont 20	2500	GEO	CIV	USSR
	186		Jul 18	28	K/DM-2	340-01	LC 200L	1990 061A		Kosmos 2085	1965	GEO	MIL	USSR
F	**187**		**Aug 09**	**22**	**K/DM-2**	**345-02**	**LC 200L**	1990 F04A		Ekran M		GEO	CIV	USSR
	188		Nov 03	86	K/DM-2	370-01	LC 81L	1990 094A		Gorizont 21	2500	GEO	CIV	USSR
	189		Nov 23	20	K/DM-2	348-02	LC 200L	1990 102A		Gorizont 22	2500	GEO	CIV	USSR
	190		Dec 08	15	K/DM-2	361-01	LC 81L	1990 110	M	Kosmos 2109–2111 (Glonass)	3@1415	MEO (64.8)	MIL	USSR
	191		Dec 20	12	K/DM-2	361-01	LC 81L	1990 112A		Raduga 26	1965	GEO	MIL	USSR
	192		Dec 27	7	K/DM-2	342-01	LC 200L	1990 116A		Raduga 1-2	2000	GEO	MIL	USSR
	193	1991	Feb 14	49	K/DM-2	344-02	LC 200L	1991 010A		Kosmos 2133	1965	GEO	MIL	USSR
	194		Feb 28	14	K/DM-2	360-02	LC 81L	1991 014A		Raduga 27	1965	GEO	MIL	USSR
	195		Apr 01	32	K	365-01	LC 200P	1991 024A		Almaz 1	18550	LEO (72.7)	CIV	USSR
	196		Apr 04	3	K/DM-2	354-02	LC 200L	1991 025	M	Kosmos 2139–2141 (Glonass)	3@1415	MEO (64.9)	MIL	USSR
	197		Jul 01	88	K/DM-2	373-01	LC 200L	1991 046A		Gorizont 23	2500	GEO	CIV	USSR
	198		Sep 13	74	K/DM-2	353-01	LC 81L	1991 064A		Kosmos 2155	1965	GEO	MIL	USSR
	199		Oct 23	40	K/DM-2	362-02	LC 200L	1991 074A		Gorizont 24	2500	GEO	CIV	USSR
	200		Nov 22	30	K/DM-2	353-02	LC 81L	1991 079A		Kosmos 2172	1965	GEO	CIV	USSR
	201		Dec 19	27	K/DM-2	355-01	LC 81L	1991 087A		Raduga 28	1965	GEO	MIL	USSR
	202	1992	Jan 29	41	K/DM-2	373-02	LC 81L	1992 005	M	Kosmos 2177–2179 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	203		Apr 02	64	K/DM-2	369-01	LC 81L	1992 017A		Gorizont 25	2500	GEO	CIV	Russia
	204		Jul 14	103	K/DM-2	371-02	LC 81L	1992 043A		Gorizont 26	2500	GEO	CIV	Russia
	205		Jul 30	16	K/DM-2	376-01	LC 81L	1992 047	M	Kosmos 2204–2206 (Glonass)	3@1415	MEO (64.9)	MIL	Russia
	206		Sep 10	42	K/DM-2	363-01	LC 81L	1992 059A		Kosmos 2209	1965	GEO	MIL	Russia
	207		Oct 30	50	K/DM-2	372-01	LC 81L	1992 074A		Ekran 20	1970	GEO	CIV	Russia
	208		Nov 27	28	K/DM-2	364-01	LC 81L	1992 082A		Gorizont 27	2500	GEO	CIV	Russia
	209		Dec 17	20	K/DM-2	357-02	LC 200L	1992 088A		Kosmos 2224	1965	GEO	MIL	Russia
	210	1993	Feb 17	62	K/DM-2	362-01	LC 81L	1993 010	M	Kosmos 2234–2236 (Glonass)	3@1415	MEO (64.9)	MIL	Russia
	211		Mar 25	36	K/DM-2	358-01	LC 81L	1993 013A		Raduga 29	1965	GEO	MIL	Russia
F	**212**		**May 27**	**63**	**K/DM-2**	**364-02**	**LC 81L**	1993 F01A		Gorizont	2500	GEO	CIV	Russia
	213		Sep 30	126	K/DM-2	359-01	LC 81L	1993 062A		Raduga 30	1965	GEO	MIL	Russia
	214		Oct 28	28	K/DM-2	368-01	LC 81L	1993 069A		Gorizont 28	2500	GEO	CIV	Russia
	215		Nov 18	21	K/DM-2	367-01	LC 81L	1993 072A		Gorizont 29	2500	GEO	CIV	Russia
	216	1994	Jan 20	63	K/DM-2M	358-02	LC 81L	1994 002A		Gals 1	2500	GEO	CIV	Russia
	217		Feb 05	16	K/DM-2	375-02	LC 81L	1994 008A		Raduga 1-3	1965	GEO	MIL	Russia
	218		Feb 18	13	K/DM-2	376-02	LC 81L	1994 012A		Raduga 31	1965	GEO	MIL	Russia
	219		Apr 11	52	K/DM-2	377-01	LC 81L	1994 021	M	Kosmos 2275–2277 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	220		May 20	39	K/DM-2	357-01	LC 81L	1994 030A		Gorizont 30	2500	GEO	CIV	Russia
	221		Jul 06	47	K/DM-2	365-02	LC 81L	1994 038A		Kosmos 2282	3800	GEO	MIL	Russia
	222		Aug 11	36	K/DM-2	367-02	LC 81L	1994 050A	M	Kosmos 2287–2289 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	223		Sep 21	41	K/DM-2	381-02	LC 200L	1994 060A		Kosmos 2291	1965	GEO	MIL	Russia
	224		Oct 13	22	K/DM-2M	377-02	LC 200L	1994 067A		Express A	2500	GEO	CIV	Russia
	225		Oct 31	18	K/DM-2	361-02	LC 81L	1994 069A		Elektro 1	2580	GEO	CIV	Russia
	226		Nov 20	20	K/DM-2	371-01	LC 200L	1994 076	M	Kosmos 2294–2296 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	227		Dec 16	26	K/DM-2	373-02	LC 81L	1994 082A		Luch	2300	GEO	CIV	Russia
	228		Dec 28	12	K/DM-2	366-01	LC 81L	1994 087A		Raduga 32	1965	GEO	MIL	Russia
	229	1995	Mar 07	69	K/DM-2	370-02	LC 200L	1995 009	M	Kosmos 2307–2309 (Glonass)	3@1415	MEO (64.7)	MIL	Russia
	230		May 20	74	K	378-02	LC 81L	1995 024A	D	Spektr	19640	STA	CIV	Russia
	231		Jul 24	65	K/DM-2	374-01	LC 200L	1995 037	M	Kosmos 2316–2318 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	232		Aug 30	37	K/DM-2	369-02	LC 200L	1995 045A		Kosmos 2319	1965	GEO	MIL	Russia
	233		Oct 11	42	K/DM-2	386-01	LC 81L	1995 054A		Luch 1 1	2400	GEO	CIV	Russia
	234		Nov 17	37	K/DM-2	384-01	LC 200L	1995 063A		Gals 2	2500	GEO	CIV	Russia
	235		Dec 14	27	K/DM-2	378-01	LC 200L	1995 068	M	Kosmos 2323–2325 (Glonass)	3@1415	MEO (64.8)	MIL	Russia
	236	1996	Jan 25	42	K/DM-2	374-02	LC 200L	1996 005A		Gorizont 31	2500	GEO	CIV	Russia

LC-81 and LC-200 are the Proton launch complexes at Baikonur. LC-81L (left) is Pad 23, LC-81P (right) is Pad 24. LC-200L is Pad 39, LC-200 P is Pad 40.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
F	**237**		**Feb 19**	**25**	**K/DM-2**	**383-02**	**LC 200L**	1996 010A		Raduga 33	1965	GTO	MIL	Russia
	238		Apr 08	49	K/DM3	390-01	LC 81L	1996 021A		Astra 1F	3010	GTO	CML	Lux.
	239		Apr 23	15	K	385-01	LC 81L	1996 023A	D	Priroda	19500	STA	CIV	Russia
	240		May 25	32	K/DM-2	379-01	LC 200L	1996 034A		Gorizont 32	2500	GEO	CIV	Russia
	241		Sep 06	104	K/DM1	375-01	LC 81L	1996 053A		Inmarsat 302	1144	GEO	CML	Int'l
	242		Sep 26	20	K/DM-2M	379-02	LC 200L	1996 058A		Express 2	2500	GEO	CIV	Russia
F	**243**		**Nov 16**	**51**	**K/D-2**	**392-02**	**LC 200L**	1996 064A		Mars 96	6825	LEO (51.4)	CIV	Russia
	244	1997	May 24	189	K/DM4	380-02	LC 81L	1997 026A		Telstar 5	3650	GTO	CML	USA
	245		Jun 06	13	K/DM5	380-01	LC 200L	1997 028A		Kosmos 2344	6000	LEO (63.4)	MIL	Russia
	246		Jun 18	12	K/DM2	390-02	LC 81L	1997 030A	M	Iridium 009	690	LEO (87)	CML	USA
								1997 030B	M	Iridium 010	690	LEO (87)	CML	USA
								1997 030C	M	Iridium 011	690	LEO (87)	CML	USA
								1997 030D	M	Iridium 012	690	LEO (87)	CML	USA
								1997 030E	M	Iridium 013	690	LEO (87)	CML	USA
								1997 030F	M	Iridium 014	690	LEO (87)	CML	USA
								1997 030G	M	Iridium 016	690	LEO (87)	CML	USA
	247		Aug 14	57	K/DM-2	381-01	LC 200L	1997 041A		Kosmos 2345	2500	GEO	MIL	Russia
	248		Aug 28	14	K/DM3	387-02	LC 81L	1997 046A		PAS 5	3600	GTO	CML	USA
	249		Sep 14	17	K/DM2	391-01	LC 81L	1997 051A	M	Iridium 029	690	LEO (87)	CML	USA
								1997 051B	M	Iridium 032	690	LEO (87)	CML	USA
								1997 051C	M	Iridium 033	690	LEO (87)	CML	USA
								1997 051D	M	Iridium 027	690	LEO (87)	CML	USA
								1997 051E	M	Iridium 028	690	LEO (87)	CML	USA
								1997 051F	M	Iridium 030	690	LEO (87)	CML	USA
								1997 051G	M	Iridium 031	690	LEO (87)	CML	USA
	250		Nov 12	59	K/DM-2M	382-01	LC 200L	1997 070A		Kupon 1	2300	GEO	CML	Russia
	251		Dec 02	20	K/DM3	382-02	LC 81L	1997 076A		Astra 1G	3300	GTO	CML	Lux.
F	**252**		**Dec 24**	**22**	**K/DM3**	**394-01**	**LC 81L**	1997 086A		HGS 1 (AsiaSat 3)	3410	EEO	CML	China
	253	1998	Apr 07	104	K/DM2	391-02	LC 81L	1998 021A	M	Iridium 062	690	LEO (87)	CML	USA
								1998 021B	M	Iridium 063	690	LEO (87)	CML	USA
								1998 021C	M	Iridium 064	690	LEO (87)	CML	USA
								1998 021D	M	Iridium 065	690	LEO (87)	CML	USA
								1998 021E	M	Iridium 066	690	LEO (87)	CML	USA
								1998 021F	M	Iridium 067	690	LEO (87)	CML	USA
								1998 021G	M	Iridium 068	690	LEO (87)	CML	USA
	254		Apr 29	22	K/DM2	384-02	LC 200L	1998 025A		Kosmos 2350	2500	GEO	MIL	Russia
	255		May 07	8	K/DM3	393-02	LC 81L	1998 028A		Echostar 4	3000	GTO	CML	USA
	256		Aug 30	115	K/DM3	383-01	LC 81L	1998 050A		Astra 2A	3492	GTO	CML	Lux.
	257		Nov 04	66	K/DM3	395-02	LC 81L	1998 065A		PAS 8	3800	GTO	CML	USA
	258		Nov 20	16	K	395-01	LC 81L	1998 067A		Zarya	19964	STA	CIV	USA
	259		Dec 30	40	K/DM2	385-02	LC 81L	1998 077A	M	Kosmos 2362–2364 (Glonass)	3@1300	MEO (64.8)	MIL	Russia
	260	1999	Feb 15	47	K/DM3	396-01	LC 81L	1999 005A		Telstar 6	3674	GTO	CML	USA
	261		Feb 28	13	K/DM2	387-01	LC 81R	1996 010A		Raduga 1-34	1965	GEO	MIL	Russia
	262		Mar 21	21	K/DM3		LC 81L	1999 013A		Asiasat 3S	3463	GTO	CML	China
	263		May 20	60	K/DM3		LC 81L	1999 027A		Nimiq 1	3600	GTO	CML	Canada
	264		Jun 18	29	K/DM3		LC81L	1999 033A		Astra 1H	3690	GTO	CML	Lux.
T, F	**265**		**Jul 05**	**17**	**K/Br. M**	**389-01**	**LC 81**	**1999 F02**		**Raduga 1-5**	**1965**	**GTO**	**MIL**	**Russia**
S	266		Sep 06	63	K/DM-2M	388-02	LC 81L	1999 047A	M	Yamal 101	1360	GEO	CML	Russia
								1999 047B	M	Yamal 102	1360	GEO	CML	Russia
	267		Sep 27	21	K/DM3	398-02	LC 81L	1999 053A		LMI 1	3740	GTO	CML	Russia
F	**268**		**Oct 27**	**30**	**K/DM-2**	**386-02**	**LC 200L**	**1999 F03A**		**Express 1A**	**2600**	**GEO**	**MIL**	**Russia**
	269	2000	Feb 12	108	K/DM3	399-02	LC 81L	2000 011A		Garuda 1	4300	GTO	CML	Russia
	270		Mar 12	29	K/DM-2M	399-01	LC 200L	2000 013A		Express 6A	2642	GEO	CML	Russia
	271		Apr 17	36	K/DM-2M	397-01	LC 200L	2000 019A		SESAT	2500	GEO	CML	Russia
	272		Jun 06	50	K/Br. M	392-01	LC 81P	2000 029A		Gorizont 45	2200	GEO	CML	Russia
	273		Jun 24	18	K/DM-2M	394-02	LC 200L	2000 031A		Express 3A	2642	GEO	CML	Russia
	274		Jun 30	6	K/DM3	400-01	LC 81P	2000 035A		Sirius Radio 1	3765	EEO (63.3)	CML	USA
	275		Jul 04	4	K/DM-2	389-02	LC 200L	2000 036A		Kosmos 2371	2150	GEO	MIL	Russia
	276		Jul 12	8	K	398-01	LC 81L	2000 037A		Zvezda Service Module	20295	STA (ISS)	CIV	Russia
	277		Aug 28	47	K/DM-2	401-02	LC 81P	2000 049A		Raduga 1-5R	2000	GEO	MIL	Russia
	278		Sep 05	8	K/DM3	400-02	LC 81L	2000 051A		Sirius Radio 2	3765	EEO (63.4)	CML	USA
	279		Oct 01	26	K/DM3	401-01	LC 81L	2000 059A		GE 1A	3593	GTO	CML	USA
	280		Oct 13	12	K/DM-2	393-01	LC 81P	2000 063A	M	Kosmos 2374 - 2376 (GLONASS)	3@1415	MEO (64.8)	MIL	Russia
	281		Oct 21	8	K/DM3	402-01	LC 81L	2000 067A		GE 6 (AMC-6)	3552	GTO	CML	USA
	282		Nov 30	40	K/DM3	402-02	LC 81L	2000 077A		Sirius Radio 3	3765	EEO (63.4)	CML	USA
	283	2001	Apr 07	128	M/Br. M	535-01	LC 81P	2001 014A		Ekran M-18	2100	GEO	CML	Russia
	284		May 15	38	K/DM3	403-01	LC 81L	2001 019A		PAS 10	3739	GTO	CML	USA

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)		Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	285		Jun 16	32	K/DM3	403-02	LC 81L	2001 025A		Astra 2C	3643	GTO	CML	Luxembourg
	286		Aug 24	69	K/DM3M	404-01	LC 81P	2001 037A		Kosmos 2379	2500	GEO	MIL	Russia
	287		Oct 06	43	K/DM-2		LC 81L	2001 045A		Raduga 1-6	2000	GEO	MIL	Russia
	288		Dec 01	56	K/DM-2		LC 81P	2001 053	M	Kosmos 2380 - 2382 (GLONASS)	3@1415	MEO (64.8)	MIL	Russia
	289	2002	Mar 30	119	K/DM3	406-01	LC 81L	2002 016A		Intelsat 903	4726	GTO	CML	USA
	290		May 07	38	K/DM3	404-02	LC 81P	2002 023A		DirecTV 5	3641	GTO	CML	USA
	291		Jun 10	34	K/DM-2M	407-01	LC 200L	2002 029A		Express A1R	2586	GEO	CML	Russia
	292		Jul 25	45	K/DM5		LC 81P	2002 037A		Kosmos 2392	2600	LEO (63.5)	MIL	Russia
	293		Aug 22	28	K/DM3	406-02	LC 81L	2002 039A		EchoStar 8	4685	GTO	CML	USA
	294		Oct 17	56	K/DM-2		LC 200L	2002 048A		INTEGRAL	3951	EEO (85)	CIV	France
F	**295**		**Nov 26**	**40**	**K/DM3**		**LC 81L**	**2002 053A**		**Astra 1K**	**5250**	**GTO**	**CML**	**Luxembourg**
	296		Dec 25	29	K/DM-2		LC 81L	2002 060A	M	Kosmos 2394 - 2396 (GLONASS)	3@1415	MEO (64.8)	MIL	Russia
	297		Dec 29	4	M/Br. M		LC 81P	2002 062A		Nimiq 2	3600	GTO	CML	Canada
S	298	2003	Apr 24	116	K/DM-2		LC 81P	2003 015A		Kosmos 2397	2155	GEO	MIL	Russia
	299		Jun 06	43	M/Br. M		LC 200L	2003 024A		AMC 9	4100	GTO	CML	Luxembourg
	300		Nov 24	171	K/DM-2M	407-02	LC 81L	2003 053A	M	Yamal 200 KA-1	1360	GEO	CIV	Russia
								2003 053B	M	Yamal 200 KA-2	1320	GEO	CIV	Russia
	301		Dec 10	16	K/Br. M	410-03	LC 81P	2003 056A	M	Kosmos 2402 (GLONASS 32-1)	1415	MEO (65.1)	MIL	Russia
								2003 056B	M	Kosmos 2403 (GLONASS 32-3)	1415	MEO (65.1)	MIL	Russia
								2003 056C	M	Kosmos 2404 (GLONASS 32-4)	1415	MEO (65.1)	MIL	Russia
	302		Dec 28	18	K/DM-2M	410-04	LC 200L	2003 060A		Express AM-22	2542	GEO	CIV	Russia

LC-81 and LC-200 are the Proton launch complexes at Baikonur. LC-81L (left) is Pad 23, LC-81P (right) is Pad 24. LC-200L is Pad 39, LC-200 P is Pad 40.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

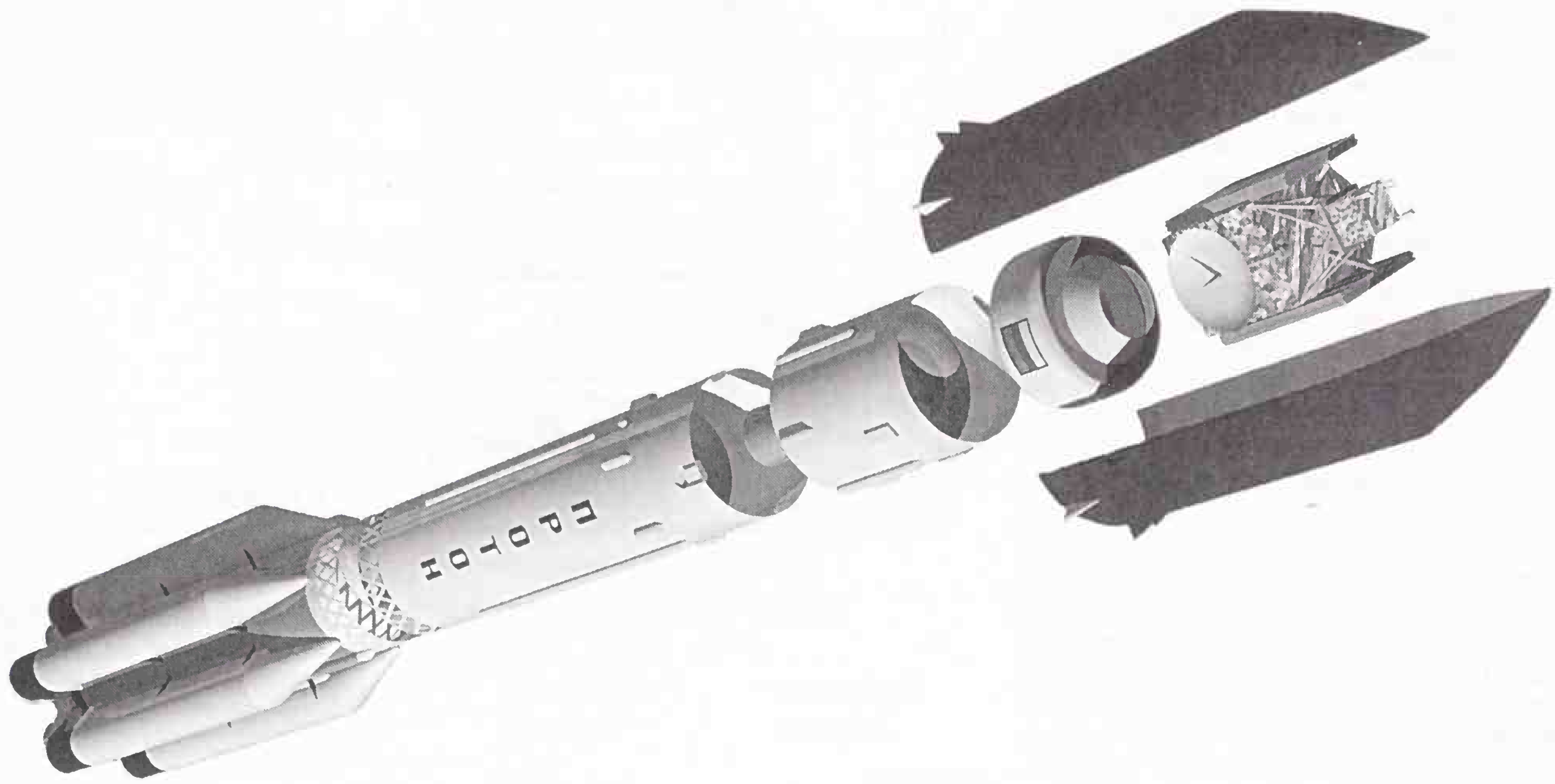
Flight History

Failure Descriptions:

F	1966 Mar 24	211	1966 F03	Failure to orbit.
F	1967 Sep 27	229-01	1967 F11	Failure to orbit.
F	1967 Nov 22	230-01	1967 F12	Second-stage failure.
F	1968 Apr 22	232-01	1968 F03	Failure to orbit.
F	1969 Jan 20	237-01	1969 F01	Second-stage failure.
F	1969 Feb 19	239-01	1969 F04	First-stage failure.
F	1969 Mar 27	240-01	1969 F06	Failure to orbit.
F	1969 Apr 02	233-01	1969 F07	Failure to orbit.
F	1969 Jun 14	238-01	1969 F08	Failure to orbit.
F	1969 Nov 28	245-01	1969 F14	Failure to orbit.
F	1970 Feb 06	247-01	1970 F02	A T+128.3 s, flight safety system automatically shut off first-stage engine because of false alarm from pressure gauge.
F	1971 May 10	253-01	1971 042A	Block D flight sequencer programmed improperly, resulting in failure to perform second burn or payload separation. Reportedly the coast time between burns was set to 1.5 years instead of 1.5 h.
F	1972 Jul 29	260-01	1972 F04	At T+181.9 s second-stage stabilization system failed because of short circuit in pitch and yaw channels of the automated stabilization system.
P	1973 Apr 03	Salyut 2	1973 017A	In mid-April the Salyut developed a severe tumbling motion, breaking up into 25 pieces and subsequently decaying on 28 May. Most sources agree that it is was likely that an attitude control thruster stuck on and caused the station to tumble until it broke up.
S	1973 May 11	Salyut	1973 026A	During or shortly after the launch phase the Salyut failed and began tumbling.
F	1975 Oct 16	287-02	1975 F05	Failure of fourth-stage oxidizer booster pump.
F	1977 Aug 04	293-01	1977 F02	At T+40.1 s, a first-stage engine steering unit failed, causing loss of stability and automatic thrust termination at T+53.7 s.
F	1978 May 27	294-02	1978 F02	Vehicle stability lost at T+87 s because of an error in first-stage No. 2 engine steering unit. Fault attributed to fuel leak in second-stage engine compartment, which caused control cables to overheat.
F	1978 Aug 17	297-02	1978 F03	Loss of stability at T–259.1 s caused flight termination. Hot gas leak from second-stage engine because of faulty seal on pressure gauge led to failure of electrical unit for automatic stabilization.
F	1978 Oct 17	298-01	1978 F04	At T+235.62 s, second-stage engine shut off with resultant loss of stability caused by a turbine part igniting in turbo pump gas tract followed by gas inlet destruction and hot air ejection into second rear section.
P	1978 Dec 19	295-02	1978 118A	Block DM was misaligned for GEO injection burn, resulting in non-circular orbit with 11 deg inclination.
F	1982 Jul 22	307-02	1982 F04	First-stage engine No. 5 suffered failure of hydraulic gimbal actuator because of dynamic excitation at T+45 s. Automatic flight shutdown commanded.
F	1982 Dec 24	314-01	1982 F08	Second-stage failure due to high frequency vibration.
F	1986 Nov 29	338-01	1986 F07	Second-stage failure.
F	1987 Jan 30	341-01	1987 010A	Fourth stage failed to start because of control system component failure.
F	1987 Apr 24	335-01	1987 036	Fourth stage shut down early and failed to restart. Failure occurred in control system because of manufacturing defect in instrument.
F	1988 Jan 18	341-02	1988 F01	Third-stage engine failure caused by destruction of fuel line leading to mixer.
F	1988 Feb 17	346-02	1988 009	Fourth-stage engine failure because of high combustion chamber temperatures caused by foreign particles from propellant tank.
F	1990 Aug 09	345-02	1990 F04	Second-stage engine shutoff because of termination of oxidizer supply due to fuel line being clogged by a wiping rag.
F	1993 May 27	364-02	1993 F01	Second- and third-stage engines suffered multiple burn-throughs of combustion chambers because of propellant contamination.
F	1996 Feb 19	383-02	1996 010A	Block DM-2 stage failed at ignition for second burn. Suspected causes were failure of a tube joint, which could cause a propellant leak, or possible contamination of hypergolic start system.
F	1996 Nov 16	392-02	1996 064A	Block D-2 fourth-stage engine failed to reignite to boost spacecraft into desired transfer orbit; injection burn did not propel spacecraft out of Earth orbit. Spacecraft and upper stage reentered after a few hours. Root cause could not be determined because of lack of telemetry coverage, but suspected cause was failure of Mars 96 spacecraft, which was controlling Block D stage, or poor integration between spacecraft and stage.
F	1997 Dec 24	394-01	1997 086A	Block DM shut down early because of improperly coated turbopump seal, leaving spacecraft in high-inclination geosynchronous transfer orbit. Customer declared spacecraft a total loss and collected insurance payment. However, Hughes salvaged the spacecraft using spacecraft thrusters to raise orbit apogee to perform two lunar swingbys, which lowered inclination and raised perigee. Apogee was then lowered to achieve a geosynchronous orbit inclined 8 deg. Spacecraft is now operational for limited use.
F	1999 Jul 05	389-01		Contaminants from a welding defect in the turbopump caused the second-stage engine no. 3 to catch fire, destroying the rear section of the stage.
S	1999 Sep 06	Yamal 102		
F	1999 Oct. 27	386-02		articulate contamination caused the turbine exhaust duct of second stage engine no. 1 to catch fire at T+223 s, resulting in rapid shutdown of the stage. This and the previous failure in July were attributed to poor workmanship at the Voronezh engine plant. Both engines were part of the same batch built in 1993, during a period when production decreased significantly.
F	2002 Nov 26			A failed valve caused excess fuel to collect in the Block DM main engine during the parking orbit coast phase after the first burn. The engine was destroyed.

Vehicle Design

Overall Vehicle



Courtesy ILS.

Proton M/Breeze M

	Proton K/Block DM	Proton M/Breeze M
Height	57.2 m (187.8 ft)	Approx. 60 m (200 ft)
Gross Liftoff Mass	691.5 t (1524 klbm)	702 t (1545 klbm)
Thrust at Liftoff	9.5 MN (2100 klbf)	9.5 MN (2100 klbf)

Stages

The Proton launch vehicle consists of three core stages for injection into low orbit. An optional fourth stage can be added for higher orbits or to change inclination. The first three stages are fueled by N_2O_4/UDMH. Proton was originally designed to serve as either a space launch vehicle or a heavy ICBM, and therefore storable propellants were chosen to permit Proton to be held on standby for a long period of time and still launch rapidly.

Proton is constructed of an aluminum alloy and is covered by an external thermal finishing paint. Propellant tanks are internally coated with an anti-corrosive coating. The oxidizer also has a passive abator mixed into it to reduce its corrosiveness.

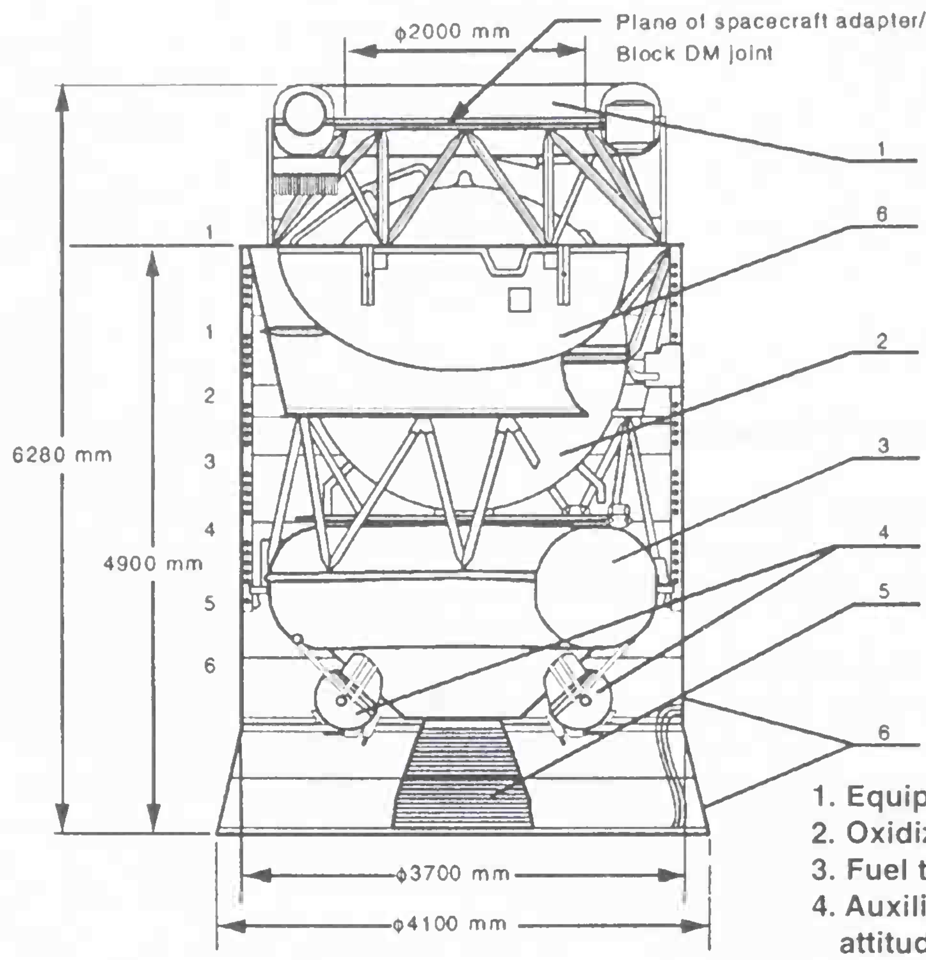
The Proton first stage consists of a large central core oxidizer tank surrounded by six smaller fuel tanks. These are easily mistaken for strap-on boosters, but in fact they are an integral part of the first stage and do not separate. This distinctive design stems from early requirements for transporting Proton. The vehicle stages had to be manufactured at a central factory, then shipped by rail to the launch sites. The maximum length and diameter of the components were therefore limited by existing railway tunnels and other rail infrastructure. However, performance requirements dictated that first stage would exceed these dimensions, thus requiring that it be manufactured as more than one element. Two designs were considered. The first consisted of two cylindrical assemblies with similar length and diameter. The fuel tank and integrated engine compartment could be shipped as one element, and the separate oxidizer tank would be stacked on top of it at the launch site to form the first stage. The second design consisted of a large central oxidizer tank, with several smaller fuel tanks that could be strapped on at the launch site. While this design was somewhat heavier and more complex, it resulted in a much stiffer structure. The total height of the rocket would also be shorter, thus simplifying launch infrastructure. These considerations led to the selection of the second configuration. In the final design, six strap-on tanks were chosen, each with one RD-253 engine. The strap-on assemblies are integrated with the core tank on a special rotating assembly fixture at the launch site.

The first-stage RD-253 engines, developed between 1961 and 1965, use a staged-combustion cycle. After the fuel goes through the pumps, all of the oxidizer and a small portion of the fuel enters the gas generator attached to the turbine housing. The remaining fuel is delivered to the regenerative cooling channels of the chamber. The turbine exhaust gas is delivered through to the combustion chamber, where it is burned with the fuel that has passed through the chamber cooling channels. For reliable cooling of the chamber, its fire wall is protected by a refractory ceramic coating and a gas–liquid film formed by delivery of a portion of the fuel component from the cooling channel to the wall through openings. The maximum fuel pressure in the main lines of the engine reaches 400 atm. The use of self-igniting hypergolic propellant eliminates the need for an ignition system. The engine is fired and cut off by nine pyrotechnic valves. A regulator and choke operating from electric drives are installed in the main lines to control the thrust and ratio of fuel and oxidizer flow rates during flight of the engine. Supercharging assemblies produce gases for pressurizing the rocket fuel tanks. The extensive application of welding ensures the structural integrity of the rocket engine. Heat shielding around the engines protect the engine assemblies against the reaction gas jet influence. The gimbal assemblies, which attach

VEHICLE DESIGN

the engine to the rocket, provide for the rotation of the main thrust chambers in a plane parallel to the rocket's longitudinal axis so that flight direction and attitude of the booster can be controlled. The RD-253 engines have been uprated first to 102% and then later to 107% of rated thrust by modifying the propellant flow control valves, and are now designated by the model number RD-259. This modification was first used as a mission-specific upgrade on the Proton K that launched the Mir space station in 1986. It was later incorporated into standard production vehicles. Through 1998, 65 flights used the 102% thrust level. The 107% uprated engines were introduced in 1995 and are standard on Proton K and M vehicles.

The second and third stages both have tanks using common bulkhead domes and are powered by different versions of the RD-0210 engine. Four single high-pressure combustion chamber RD-0210 engines are mounted on the second stage. Three of these are the RD-465/RD-0210 variant, while the fourth is an RD-468/RD-0211 which includes a gas generator for tank pressurization. The third stage uses a single RD-473/RD-0212 variant of the basic RD-0210 engine, which includes a fixed central RD-0213 version of the RD-0210 engine plus four conventional vernier thrust chambers designated RD-0214. The third stage cannot be restarted or throttled, but, like all stages on Proton, the burn time can be varied according to the mission. In response to two failures, the RD-0210 engines were upgraded in 2000 to reduce the risk of fire and to filter potential contaminants in the propellant feed lines.



1. Equipment bay
2. Oxidizer tank
3. Fuel tank
4. Auxiliary propulsion for attitude control ("SOZ" units)
5. Main engine
6. Missile and lower adapters (shrouds)

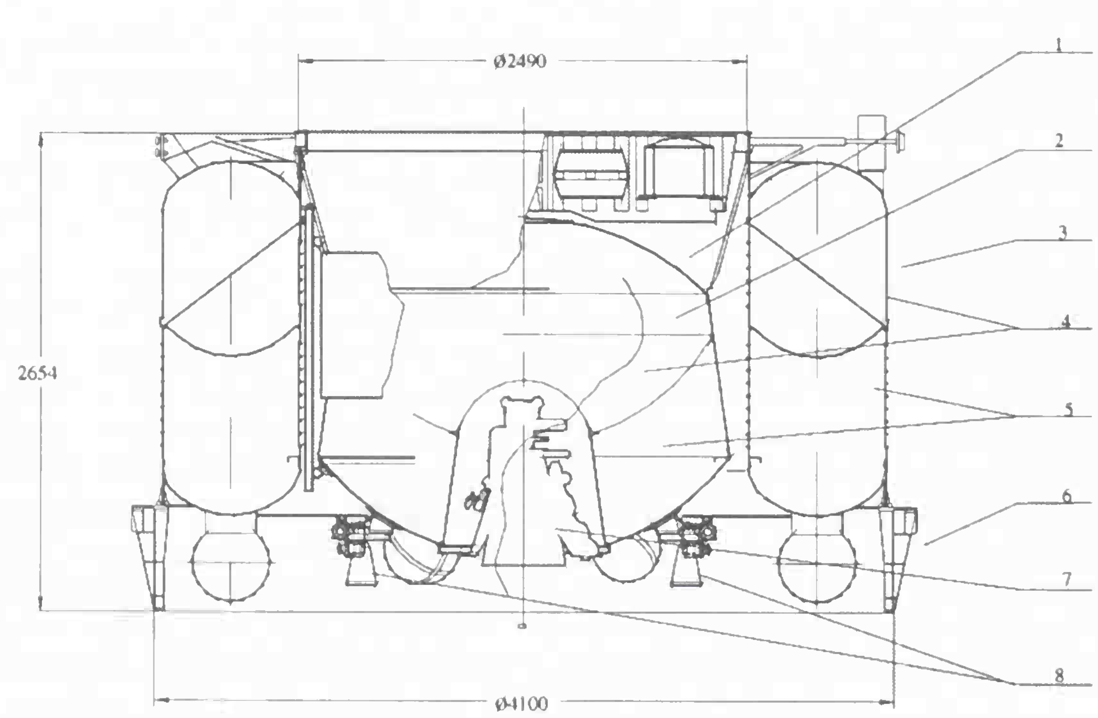
Courtesy ILS.

Block DM

The Block DM fourth stage is used to transfer payloads from LEO into GEO or interplanetary trajectories. It includes a spherical LOX tank and a toroidal kerosene tank that circles the single 11D58M main engine. The tanks are connected with a truss structure. To provide long operational lifetime in orbit before the LOX boils away, the oxygen tank is covered with ML1 "superinsulation" and thermally isolated from the rest of the stage using inserts in the structural connections that have low thermal conductivity. The standard Block DM avionics are housed in a pressurized, toroidal instrumentation compartment at the top of the stage. The alternative Block D stage lacks the instrumentation compartment and is instead controlled by its payload. The Block DM stage can function for 24 h or more on orbit, and is capable of five to seven restarts, allowing complex orbital maneuvers. The main engine can burn either conventional kerosene or a synthetic hydrocarbon fuel called syntin. Syntin is slightly denser and provides slightly better performance, but is more expensive than kerosene. When integrated on the launch vehicle, the Block DM is contained inside a cylindrical skin-stringer load-bearing shroud that is jettisoned before Block DM ignition. The shroud is divided into two sections, referred to as the lower adapter and middle adapter. Combined, the mass of these elements is about 850–900 kg (1900–2000 lbm), which is not included in the mass of the stage reported in the table below.

Changes for the Proton M consist of a number of minor structural enhancements with no fundamental changes to the first three stages plus the replacement of the original Proton K analog avionics system with a state-of-the-art digital GNC system. Stage inert weights are reduced using improved manufacturing techniques, minor design changes, and a reduction in avionics mass. The forward section of the second stage is strengthened to accommodate greater loads from heavier payloads and larger fairings. The propellant lines for the first and second stages are redesigned to reduce propellant residuals by 50%. This provides a slight performance improvement but is primarily intended to reduce the amount of toxic propellant that remains in the tanks when the jettisoned stages impact in the designated drop zones. To this end, a propellant purge system has also been added to expel propellants into the upper atmosphere before impact.

The Breeze M upper stage is a modified version of the Breeze KM upper stage used on Rockot. It consists of two separate components. The cylindrical core section, which is very similar to the Breeze KM, includes a single main engine, reaction control system engines, propellant tanks, and an avionics bay. Surrounding the core section is a toroidal auxiliary propellant tank that carries additional propellant. This tank carries the majority of the Breeze M propellant, and can be jettisoned in flight once the propellant is exhausted.



Courtesy ILS.

Breeze M Upper Stage

1. Equipment bay
2. Central fuel tank
3. Additional propellant tank (toroidal tank)
4. Oxidizer tanks
5. Fuel tanks
6. Lower spacer
7. Main engine
8. Auxiliary (RCS) propulsion system

Vehicle Design

	Stage 1	Stage 2	Stage 3	Stage 4 Block DM	Stage 4 Breeze-M
Dimensions					
Length	21.2 m (69.5 ft)	with interstage: 17.1 m (55.9 ft)	6.9 m (22.6 ft)	6.3 m (20.7 ft) without payload adapter	2.8 m (9.2 ft)
Diameter	Core: 4.1 m (13.4 ft) External tanks: 1.7 m (5.6 ft) Total: 7.4 m (24.3 ft)	4.1 m (13.4 ft)	4.1 m (13.5 ft)	3.7 m (12.1 ft)	4 m (13 ft)
Mass					
Propellant Mass	419.4 t (924.4 klbm)	156.1 t (344.2 klbm)	46.6 t (102.6 klbm)	15,050 kg (33,180 lbm)	Core tank: 5200 kg (11,500 lbm) External tank: 14,600 kg (32,200 lbm)
Inert Mass	Proton K: 31.0 t (68.3 klbm) Proton M: 30.6 t (67.4 klbm)	Proton K: 11.7 t (25.8 klbm) Proton M: 11.4 t (25.1 klbm)	Proton K: 4185 kg (9225 lbm) Proton M: 3700 kg (8150 lbm)	With equipment bay: 2440 kg (5380 lbm) Without equipment bay: 1700 kg (3750 lbm)	2370 kg (5225 lbm)
Gross Mass	450 t (993 klbm)	Proton K: 167.8 t (369.9 klbm) Proton M: 167.5 t (369.3 klbm)	50.6 t (112 klbm)	16,300–17,490 kg (35,935–38,560 lbm)	22,170 kg (48,865 lbm)
Propellant Mass Fraction	0.93	0.93	0.92	0.86–0.90	0.89
Structure					
Type	Skin-stringer	Skin-stringer	Skin-stringer	Monocoque, truss	Monocoque
Material	Aluminum	Aluminum	Aluminum	Aluminum	Aluminum composite
Propulsion					
Engine Designation	RD-259	RD-0210 (3 RD-465 and 1 RD-468 version)	RD-0210 (RD-473 version)	11D58M / RD-58M	S5.98M main engine (Khimavtomatiki Design Bureau) 11D458 verniers
Number of Engines	6	4	1 core engine + 1 vernier engine with 4 nozzles	1	1 main engine + 4 vernier engines
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	LOX/kerosene or syntin	N_2O_4/UDMH
Average Thrust (total)	Sea level: 9.5 MN (2.1 Mlbf) Vacuum: 10.5 MN (2.3 Mlbf) klbf)	Vacuum: 2.33 MN (524 klbf)	Vacuum: Main engine: 583 kN (131 klbf) Vernier total: 31 kN (7 klbf)	Syntin: 86.3 kN (19.4 klbf) Kerosene: 83.4 kN (18.7 klbf)	Main engine: 19.62 kN (4410 lbf) Verniers: 396 N (89 lbf)
Isp	Sea level: 285 s Vacuum: 316 s	Vacuum: 326.5 s	Vacuum: 326.5 s	Syntin: 361 s Kerosene: 352 s	Main engine: 325.5 s Verniers: 252 s
Chamber Pressure	147 bar (2130 psi)	147 bar (2130 psi)	147 bar (2130 psi)	Syntin: 79.4 bar (1152 psi) Kerosene: 77.4 bar (1123 psi)	97 bar (1405 psi)
Nozzle Expansion Ratio	26:1	81.3:1	81.3:1	189.1:1	153.8:1
Propellant Feed System	Staged-combustion turbopump	Staged-combustion turbopump	Staged-combustion turbopump	Staged-combustion turbopump	Main engine: closed-cycle turbopump Verniers: pressure fed
Mixture Ratio (O/F)	2.69:1	2:0:1	2:0:1	2.6:1	2:0:1
Throttling Capability	40% before liftoff, 100–107% during flight	100% only	100% only	100% only	100% only
Restart Capability	No	No	No	Up to 7 restarts	Up to 8 total starts (5 demonstrated)
Tank Pressurization	Gas Generator	Gas Generator	Gas Generator	Stored pressurized gas	Nitrogen
Attitude Control					
Pitch, Yaw,	Gimbal 6 nozzles	Gimbal 4 engine nozzles	4 verniers	Nozzle gimbal ± 3 deg	Nozzle gimbal
Roll	Gimbal 6 nozzles	Gimbal 4 engine nozzles	4 verniers	Rotating nozzle using turbine exhaust gas	12 ACS thrusters, 13.3 N (3.0 lbf) each
Staging					
Nominal Burn Time	120 s	210 s	230 s	680 s	3000 s
Shutdown Process	Burn to depletion	Burn to depletion	Predetermined velocity	Predetermined velocity	Command shutdown
Stage Separation	Stage 2 ignition	6 solid retro-rockets	Retro-rockets	ACS/Spring ejection of payload	ACS/Spring ejection of payload

Vehicle Design

Attitude Control System

On the first stage, each of the six RD-259 engines can gimbal in a single plane tangential to the stage's circumference. Thus, all six engines together provide full yaw, pitch, and roll control. Second-stage control is achieved by gimbaling the four RD-0210 engines. The third stage is controlled by four vernier rocket engines spaced around the single fixed RD-0210.

During Block DM stage burns, the 11D58M main engine gimbal provides pitch and yaw control, and turbopump bleed gas is used for roll control. The Block DM stage also has two sets of independent SOZ (micro) thruster units for three-axis attitude control during coast phases and for propellant settling before main engine ignition. These are fueled by small tanks of N_2O_4/UDMH and contain four thrusters each, ranging in thrust from about 25 N (5.5 lbf) for the axial propellant settling thrusters to 100 N (22.5 lbf) for the yaw thrusters. The SOZ units can be jettisoned after the main engine ignites for its final burn.

The Breeze M stage has a gimbaled main engine for pitch and yaw control, and 12 ACS thrusters with a thrust level of 13.3 N (3.0 lbf) each for roll control during main engine burns and three-axis attitude control during nonpropulsive phases of flight.

Avionics

The Proton third stage is equipped with a triple-string, voting, closed-loop guidance system for the control of the first three stages. Navigation data are collected from distributed inertial reference platforms. The Proton K control system is analog and based on 1960s-era electronics. The Proton M has modern digital flight control systems, which will simplify testing and flight algorithm loading and will allow more complex ascent flight profiles.

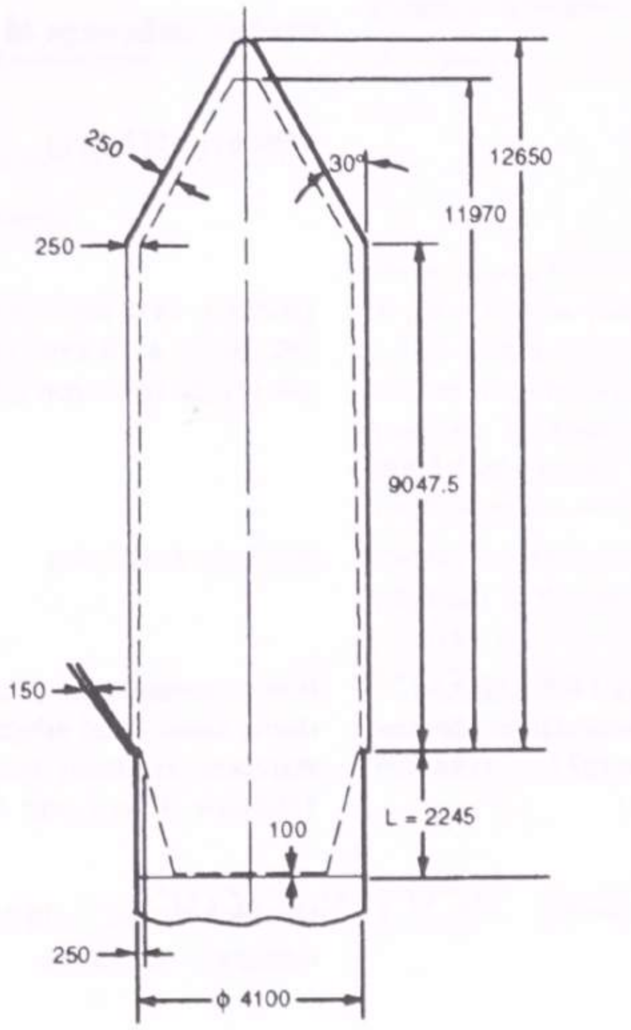
The Block DM has an independent guidance, navigation, and control system, which is contained in a pressurized toroidal compartment above the oxygen tank. The guidance system is a triple-string, voting, digital system that can be commanded from the ground. Limitations of the Block DM gyro platform prevent the stage from rotating more than 180 deg from a given reference orientation.

The Breeze M has an onboard digital computer, three-axis IMU and GLONASS and GPS navigation systems. It is capable of either three-axis-stabilized orientation or spin-stabilized mode. The GNC system was developed using modern 1990s-era hardware and software and architectures. It is a multiple redundancy system, which can be commanded from the ground and can retarget and optimize its trajectory in flight. It can be programmed to complete complex maneuvers to meet spacecraft pointing, thermal control, and separation requirements.

Payload Fairing

Multiple payload fairings are available for the Proton. All fairings are made up of two half shells. Additional acoustic blanket layers can be attached to the inside of the payload shroud if required. Proton payload fairings incorporate active air and liquid cooling and heating systems to maintain spacecraft prelaunch thermal control, regardless of ambient conditions at the launch site. The payload fairing is attached to the third stage in the three-stage or Breeze M configurations or the Block DM cylindrical shroud. Because of hardware impact zone constraints, the Proton payload fairing must be separated either approximately 180 s into flight at an altitude of 110 km (59 nmi), or approximately 350 s into flight, immediately following second-stage separation, at an altitude of 200 km (108 nmi).

Vehicle Design



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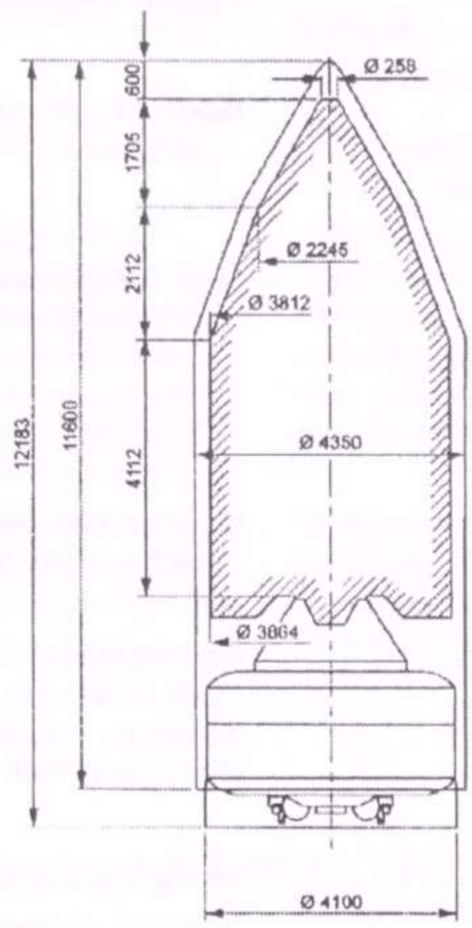

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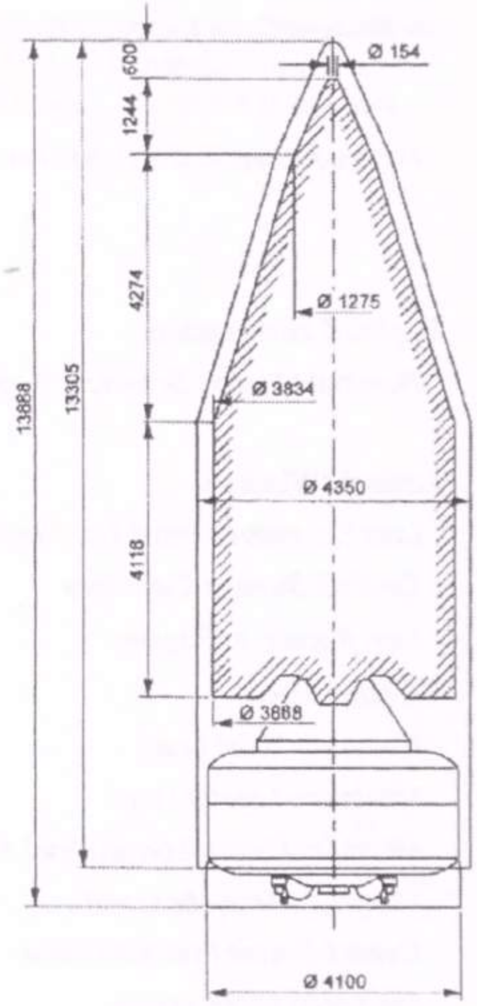

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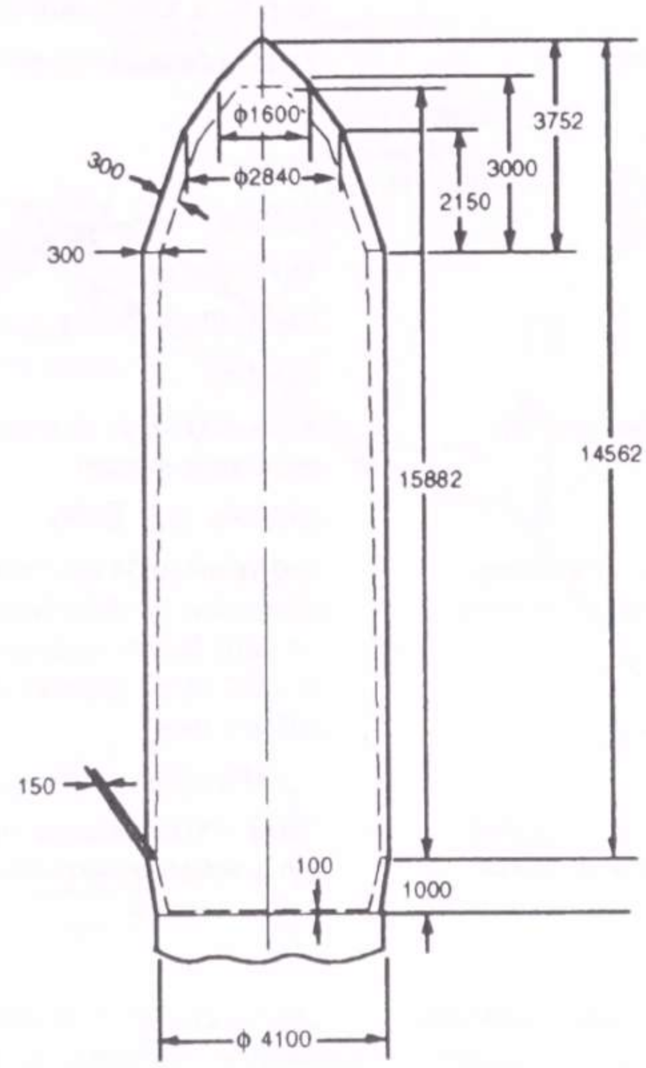
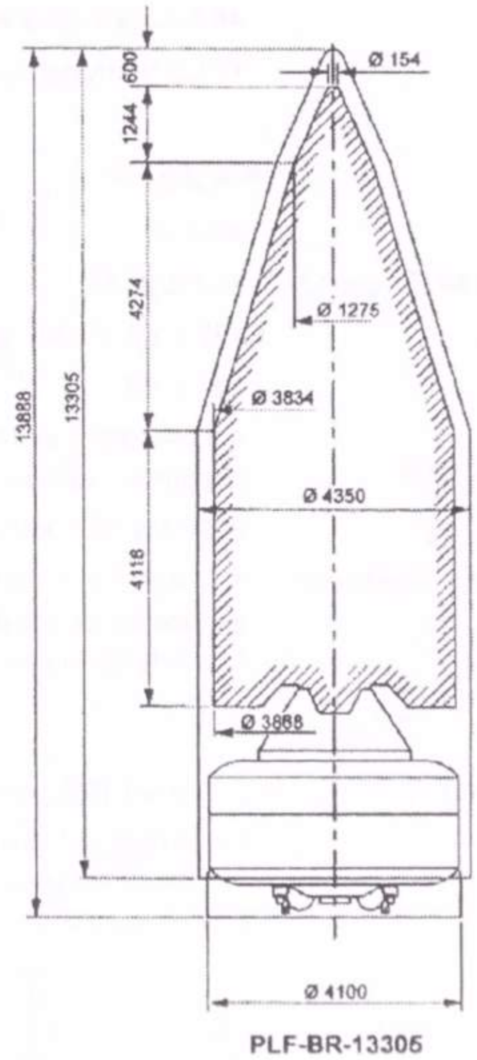


Model 2



PLF-BR-13305

Courtesy ILS.

	Models 1 and 2	**Breeze M Commercial Fairing**
Length	14.9 m (48.9 ft) 15.6 m (51.2 ft)	11.6 m (38.1 ft)—BR-11600 13.3 m (43.3 ft)—BR-13305 15.2 m (49.9 ft)—BR-15255
Primary Diameter	4.4 m (14.4 ft)	4.35 m (14.3 ft)
Mass	?	?
Sections	2	2
Structure	Skin-stringer	Monocoque sandwich
Material	Aluminum	Composite

Payload Accommodations

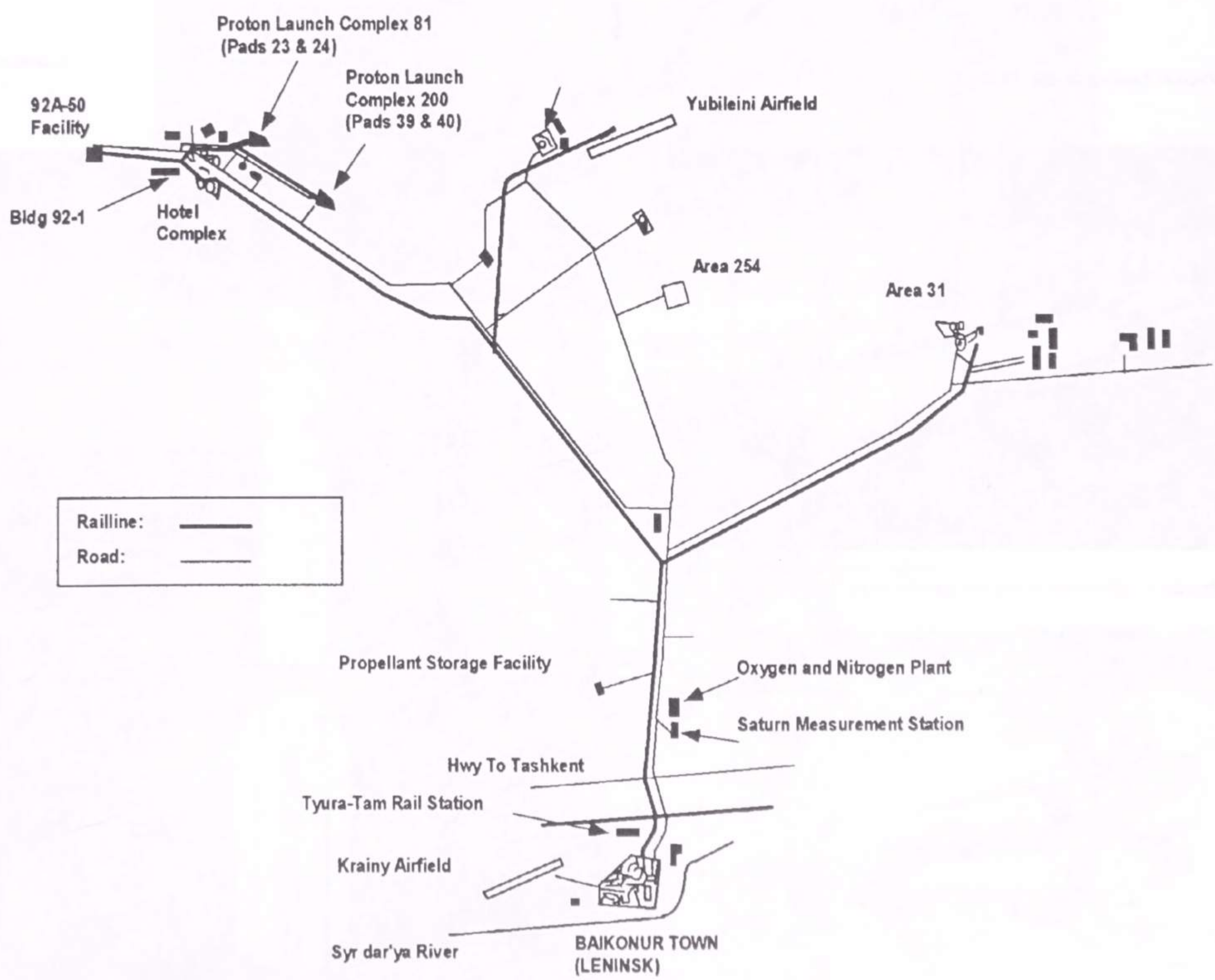
	Proton K/Block DM	Proton M/Breeze M
Payload Compartment		
Maximum Payload Diameter (Note: Cross section of available volume is not quite circular. Available diameter is approximate.)	Block DM commercial fairing: 3970 mm (156 in.)	3864 mm (152.1 in.)
Maximum Cylinder Length	Block DM commercial fairing: 3505 mm (138.0 in.)	BR-1160: 4112 mm (161.9 in.) BR-13305: 4116 mm (162.0 in.) BR-15255: 6064 mm (238.8 in.)
Maximum Cone Length	Block DM commercial fairing: 2817 mm (110.9 in.)	3800 mm (149.6 in.)
Payload Adapter Interface Diameter	Both separation nut (truss type) and Marmon clamp (shell type) adapters available in standard diameters, including 937 mm (37 in.), 1194 mm (47 in.), and 1666 mm (66 in.)	Both separation nut (truss type) and Marmon clamp (shell type) adapters available in standard diameters, including 937 mm (37 in.), 1194 mm (47 in.), and 1666 mm (66 in.)
Payload Integration		
Nominal Mission Schedule Begins	New spacecraft: T–18 to 24 months Reflight: T–6 months	New spacecraft: T–18 to 24 months Reflight: T–6 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–70 min	T–70 min
On-Pad Storage Capability	48 h for a fueled vehicle	48 h for a fueled vehicle
Last Access to Payload	T–1.5 h through access doors	T–1.5 h through access doors
Environment		
Maximum Axial Load	+4.3 *g*	+4.3 *g*
Maximum Lateral Load	±2.3 *g*	±1.35 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz/25 Hz	10 Hz/25 Hz
Maximum Acoustic Level	132.4 dB at 100 Hz	132.4 dB at 100 Hz
Overall Sound Pressure Level	141.4 dB	141.4 dB
Maximum Flight Shock	4000–8000 *g* at 10 kHz, depending on separation system	4000–8000 *g* at 10 kHz, depending on separation system
Maximum Dynamic Pressure on Fairing	3890 Pa (800 lbf/ft^2)	3890 Pa (800 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	130 W/m^2 (0.01 BTU/ft^2/s) for late fairing separation or 9600 W/m^2 (0.85 BTU/ft^2/s) for early fairing separation	130 W/m^2 (0.01 BTU/ft^2/s) for late fairing separation or 9600 W/m^2 (0.85 BTU/ft^2/s) for early fairing separation; optimum jettison at 1135 W/m^2 planned with use of additional jettison zone
Maximum Pressure Change in Fairing	1.5 kPa/s (0.2 psi/s)	1.5 kPa/s (0.2 psi/s)
Cleanliness Level in Fairing	Class 100,000 (no air conditioning during 4-h erection period onto pad and 70 min before launch)	Class 100,000 (no air conditioning during 4-h erection period onto pad and 70 min before launch)
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	LEO at 200 km (108 nm) circular at 51.6 deg: Apogee: ±15 km (8.1 nmi) Perigee: ±6 km (3.2 nmi) Inclination: ±0.03 deg GTO: Apogee: ±160 km (86 nmi) Perigee: 5500 km (2700 nmi) ±400 km (215 nmi) Inclination: ± 0.25 deg	LEO at 200 km (108 nm) circular at 51.6 deg Apogee: ±4.0 km (2.5 nmi) Perigee: ±2.0 km (1.0 nmi) Inclination: ±0.03 deg GTO: Apogee: ±100 km (62.5 nmi) Perigee: 5500 km (2700 nmi) ±300 km (187.5 nmi) Inclination: ± 0.15 deg
Attitude Accuracy (3 sigma)	±5 deg, ±0.7–1.8 deg/s	±5 deg, ±0.7–1.8 deg/s
Nominal Payload Separation Rate	0.15–0.5 m/s (0.5–1.5 ft/s)	0.15–0.5 m/s (0.5–1.5 ft/s)

Payload Accommodations

	Proton K/Block DM	Proton M
Deployment Rotation Rate Available	6 deg/sec	6 deg/sec
Loiter Duration in Orbit	24 h	More than 24 h
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes
Multiple/Auxiliary Payloads		
Multiple/Comanifest	Proton has carried multiple spacecraft on mission unique dispensers. For example, seven spacecraft per flight were carried for Iridium. Proton has not carried comanifested payloads for separate customers in the past, but a dual-payload fairing structure has been considered as an option for future missions.	Proton has carried multiple spacecraft on mission unique dispensers. For example, seven spacecraft per flight were carried for Iridium. Proton has not carried comanifested payloads for separate customers in the past, but a dual-payload fairing structure has been considered as an option for future missions.
Auxiliary Payloads	Proton has not generally carried auxiliary payloads in the past. Availability of secondary payload opportunities is unknown. Contact ILS for more information.	Proton has not generally carried auxiliary payloads in the past. Availability of secondary payload opportunities is unknown. Contact ILS for more information.

Production and Launch Operations

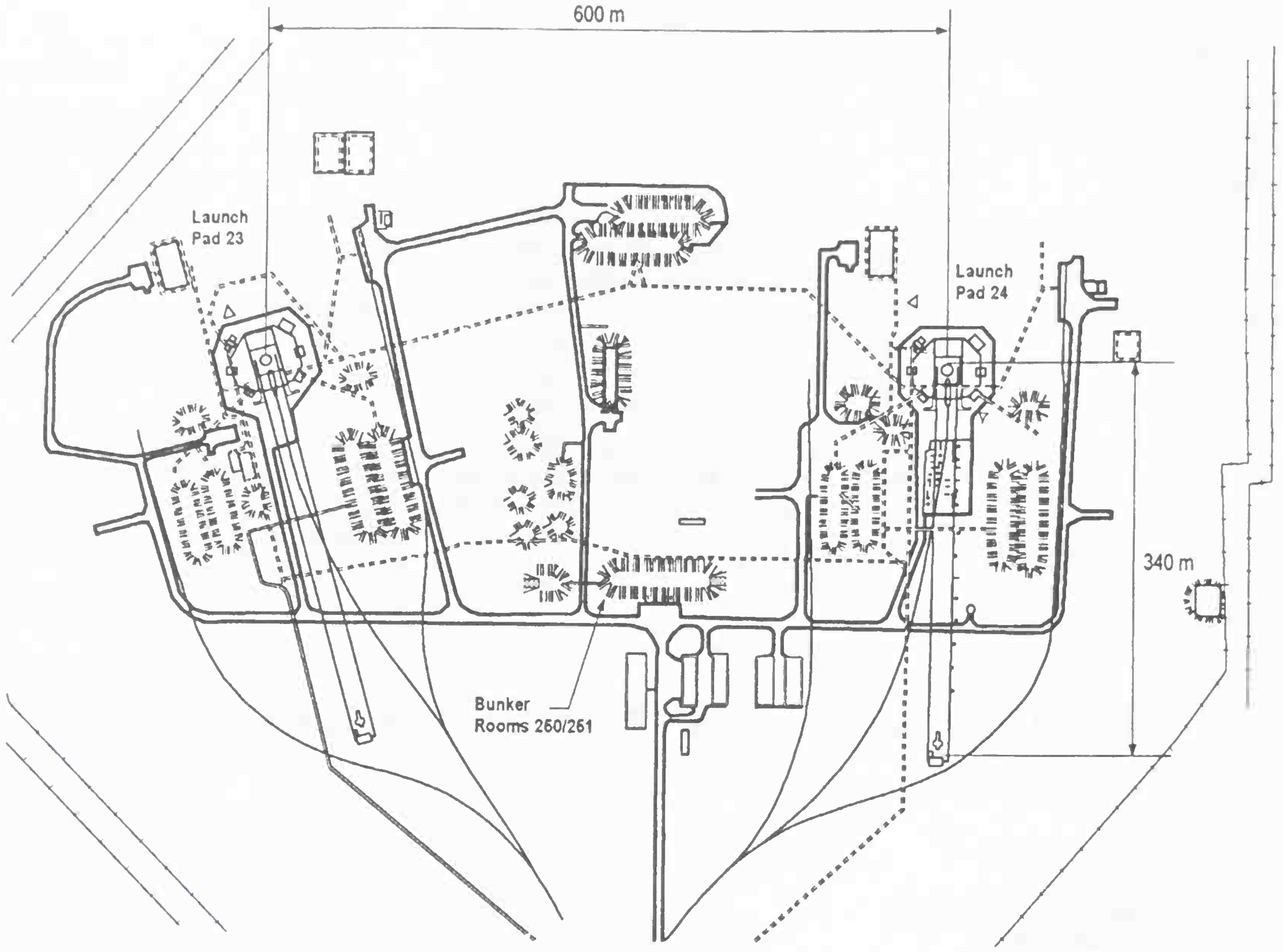
Launch Facilities



Courtesy ILS.

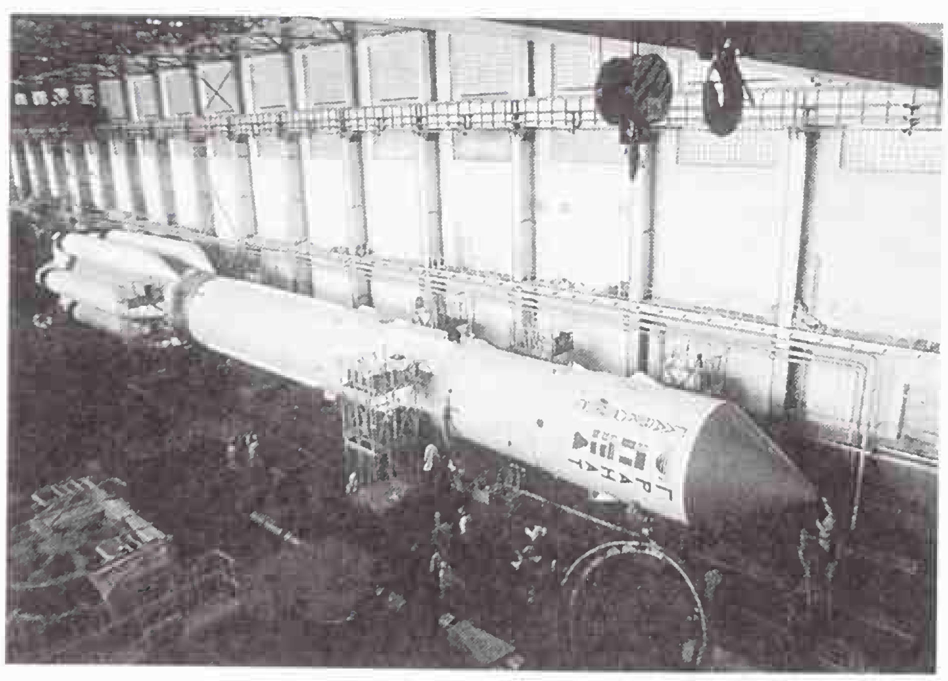
Baikonur Launch Site Layout

Production and Launch Operations



Proton Launch Zone, Area 81

Courtesy ILS.

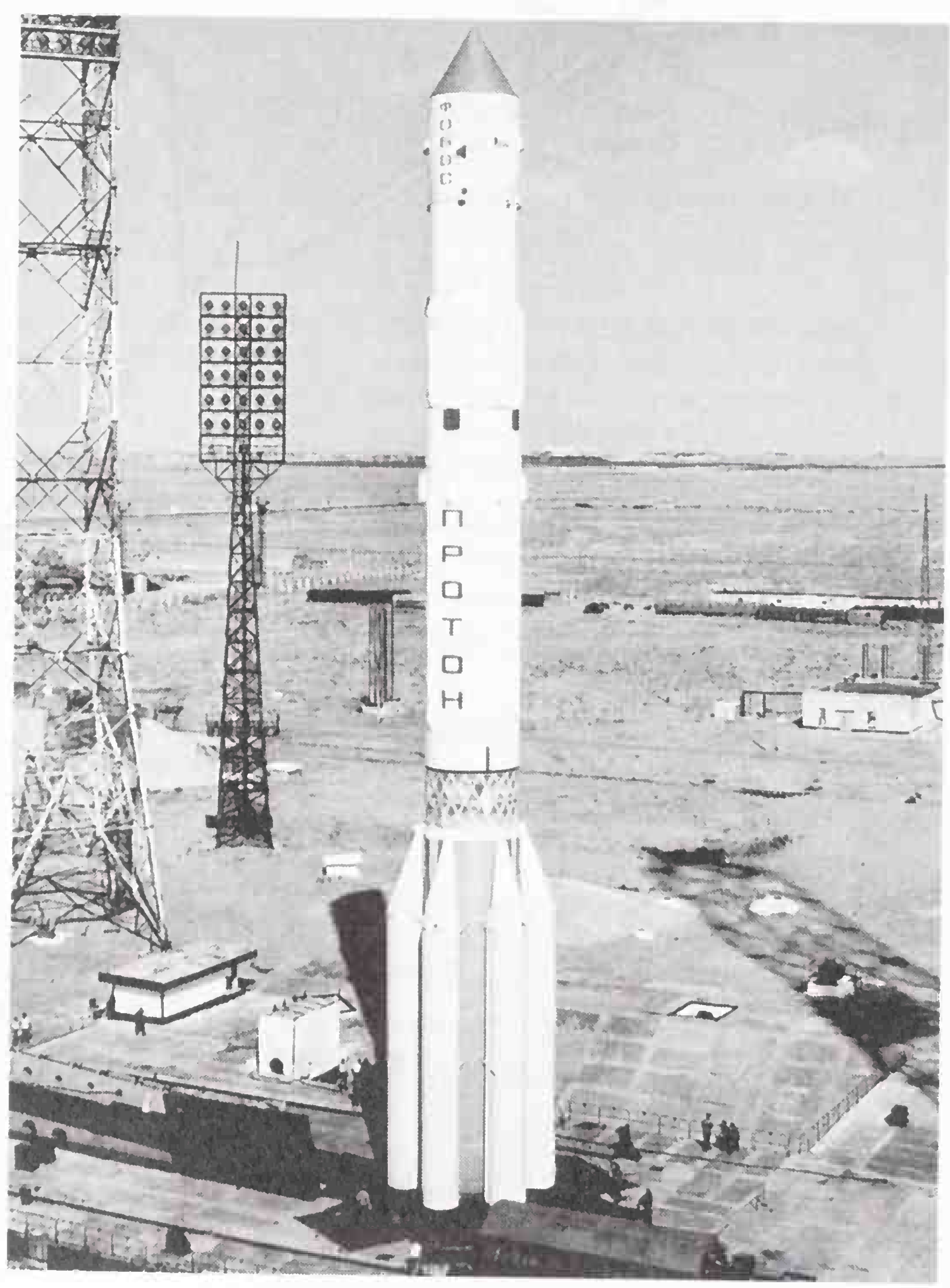


Courtesy Space Commerce Corporation.

Proton Preparation at the Horizontal Assembly Facility

Courtesy Space Commerce Corporation.

Rail Transport to the Launch Pad



Courtesy Space Commerce Corporation.

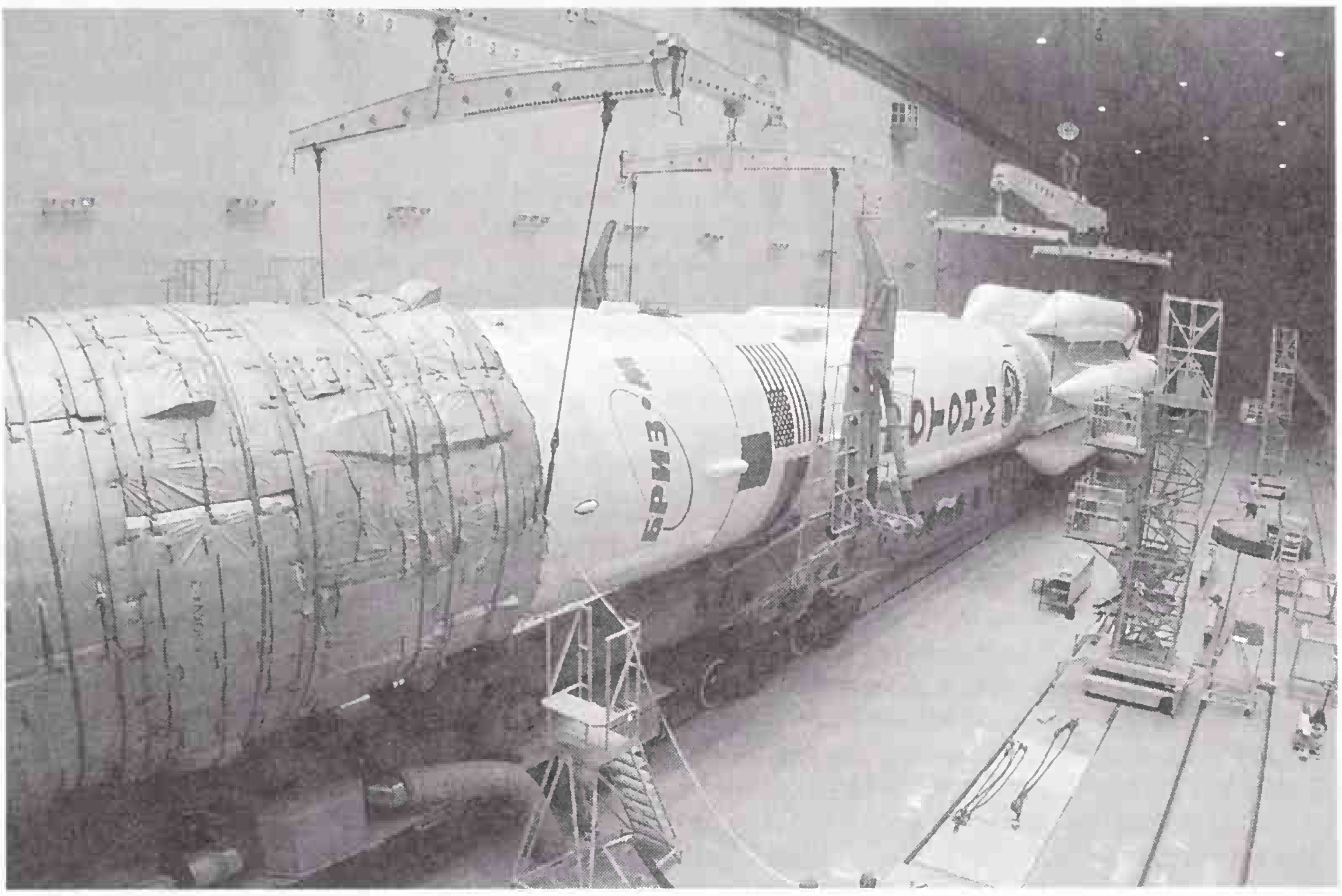
View of Proton Launch Site Layout

Production and Launch Operations

Production

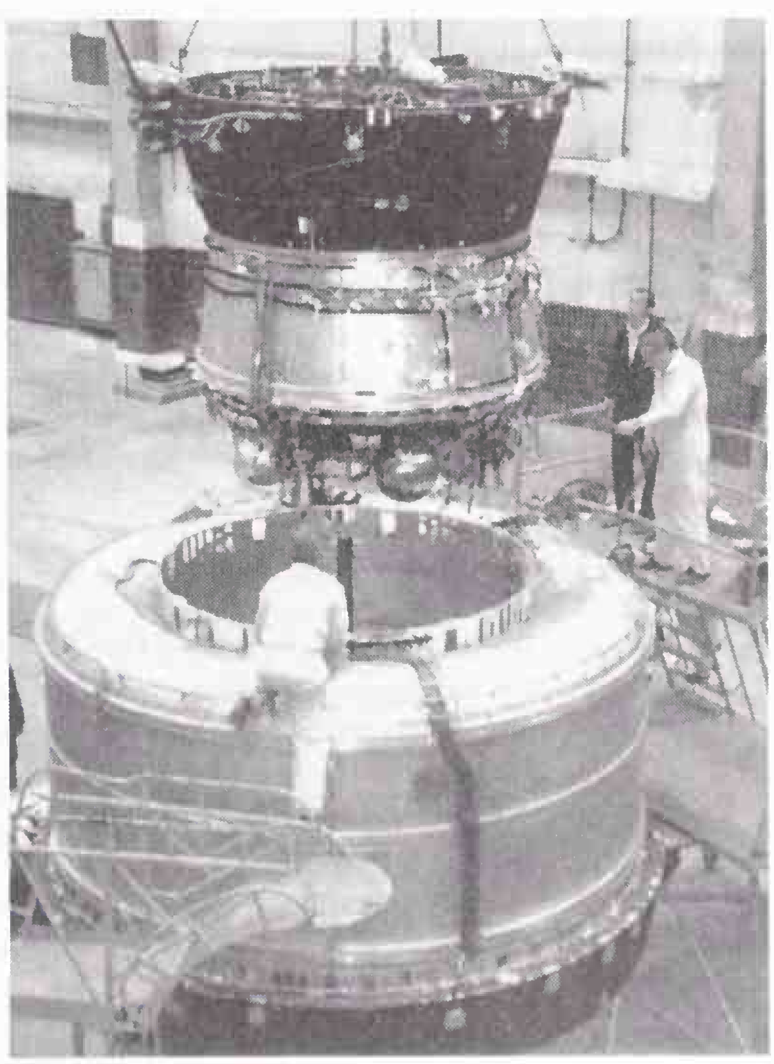
Proton is produced at the Khrunichev factory in Moscow. The Proton stages and Block DM upper stage are manufactured and assembled at the factory, then shipped by rail to Baikonur for integration. The Khrunichev factory has suffered significant budgetary and personnel cutbacks as a result of the end of the SS-19 missile program, which had been one of the major products of the facility. Fortunately, commercialization of the Proton has been successful and the Proton production line is the only missile or launch vehicle line in the former Soviet Union to be operating near full capacity. Unlike other launch vehicles, Proton launch rates are still comparable to their peak during the Soviet era. The following organizations produce elements of the Proton.

RSC Energia	Block DM fourth stage
NPO Energomash	First-stage engines
KB Khimavtomatiki	Second- and third-stage engines
IKP Mars/NIIAP	Guidance and control systems
SAAB Aerospace	Commercial payload separation systems



Courtesy ILS.

A Proton M is assembled in one of the Khrunichev factory buildings.

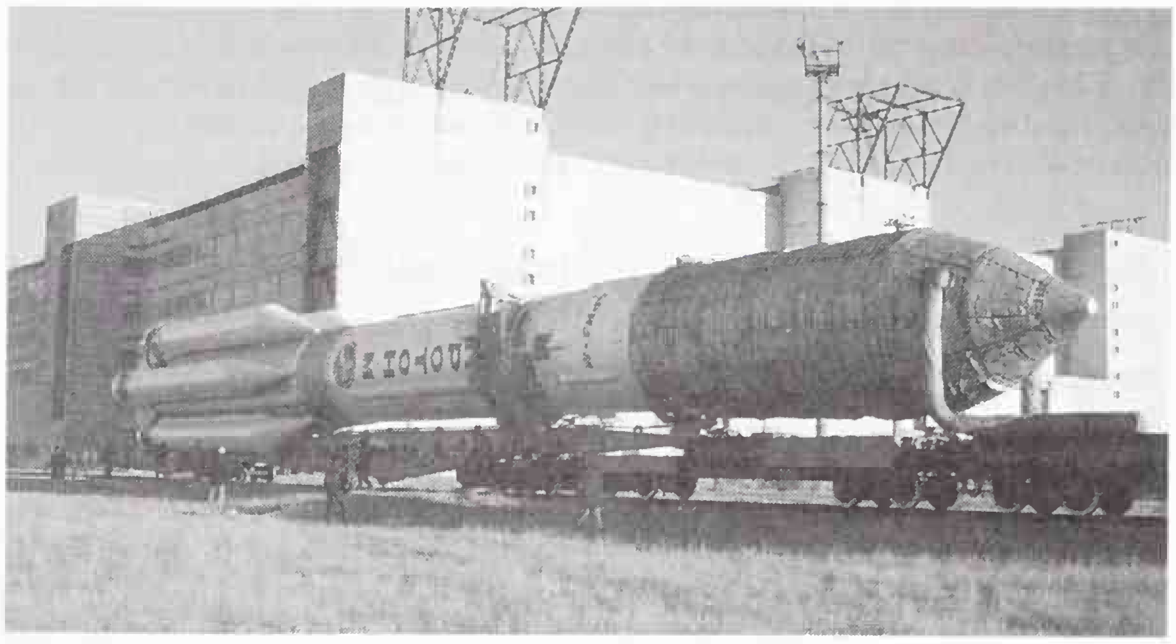


Courtesy ILS.

Breeze M Upper Stage

Production and Launch Operations

Launch Operations—Baikonur



Courtesy ILS.

Proton is launched from the Baikonur Cosmodrome, in the Republic of Kazakhstan east of the Aral Sea. For general information on Baikonur, please see the Spaceports chapter. Spacecraft can arrive at the Yubileini Airfield in the north-central section of the cosmodrome. The airfield is a Category 3 airfield, capable of handling very large aircraft. In fact it was built for landings of the Soviet space shuttle, and of its An-225 carrier aircraft—the largest aircraft in the world. Spacecraft are then transported to Area 92 on the western arm of the Cosmodrome, where the spacecraft processing facilities, Proton launch pads, and customer housing are located.

Proton stage assembly, vehicle integration, and prelaunch testing are performed in Buildings 92-1 and 92A-50. In the long assembly hall, first-stage external fuel tanks are integrated onto the first-stage core on a special handling fixture that rotates the entire stage for easy access. The vehicle is fully integrated in a horizontal position on rail trolleys. Spacecraft processing takes place in one of two areas. Early commercial launches used Area 31, Building 40D for nonhazardous processing, and Building 44 for hazardous processing. Area 31 is on the eastern arm of the cosmodrome and is also used for Soyuz and Zenit processing. Current and future commercial payloads will use Building 92A-50, which is much closer to the Proton launch complexes and customer hotels and has been upgraded to Western standards. Area 31 is still available if required, and is also used for fueling the Block DM upper stage. The space vehicle processing facility (SVPF) is in one small section of Building 92A-50, and consists of a long row of halls connected end to end. Overhead cranes are available, and a set of rail tracks running the length of the building can be used to move spacecraft. The spacecraft in its container is delivered to Hall 102 for reception, unpacking, and storage. Once unpacked, it is moved through Hall 101 to Hall 103 at the opposite end of the building. Hall 103A is used for hazardous operations, including fuel loading. Once fueled and checked out, the spacecraft is returned to Hall 101, for integration onto the payload adapter, mating with the fueled upper stage, and encapsulation into the payload fairing. It is then rotated into a horizontal position and mounted on a rail car for delivery either to Building 92-1, or to Hall 111 of Building 92A-50.

The Proton Launch Vehicle Processing Rooms in Buildings 92-1 and 92A-50 are approximately 30 m wide×19 m long×23 m high (100×390×75 ft), and contain three rail tracks upon which Protons can be transported. Assembly and integration of the Proton launch vehicle stages are carried out with the vehicle in a horizontal position. The first stage's core oxidizer tank and six fuel tank–engine assemblies are transported to Baikonur separately. Assembly of the first stage is accomplished by placing the core tank in a fixture that rotates the tank about its longitudinal axis. After the first fuel tank–engine assembly is attached from underneath the horizontal oxidizer tank, the whole assembly is rotated one-sixth of a revolution to allow for the integration of the next fuel tank–engine module. The process is repeated until all six fuel tank–engine assemblies are attached. Following completion of its assembly, the first stage is moved by crane to the assembly integration erector trolley, where it is mated to the second and third stages. The integrated vehicle is then mated to the upper stage and payload assembly. The assembled Proton then undergoes integration tests. Up to three Proton launch vehicles can be processed simultaneously in each building, and the assembly buildings can accommodate as many as six boosters at one time. Each vehicle requires two weeks for assembly, followed by one week of end-to-end tests. However, the process has been done in as little as nine days for especially urgent missions. Once fully prepared, the Proton is lifted onto a transport-erection rail car using overhead bridge cranes, and transported to the launch pad. A thermal control rail car is available to maintain required environments inside the fairing. Once at the launch pad, the erector lifts the Proton into the vertical position and mounts it on the launch pad.

Protons are launched from two launch complexes at Baikonur, Launch Complex 81 and Launch Complex 200. Each complex has two pads. Launch Complex 81 was built first, in the 1960s and then deactivated in the late 1970s when the pads reached the end of their initial service life. One of the pads, Pad 23 (LC 81 Left) was refurbished and reactivated in 1989. It also became the primary commercial launch pad in the 1990s. The second pad, Pad 24 (LC 81 Right) was repaired and refurbished in the late 1990s in response to increasing commercial demand for Proton K, and the need for a pad designed to launch the new Proton M. Krunichev spent 260 million rubles ($8 million) renovating Pad 24 using funding from its commercial revenues. It will be the primary pad for Proton M. Launch Complex 200 was built in the late 1970s, apparently to handle planned versions of the Proton with a new fluorine/ammonia upper stage powered by the RD-301 engine. The advanced upper stage never flew, probably a result, in part, of engineering and safety problems associated with the propellants. LC-200 includes Pad 39 (LC 200 Left) and Pad 40 (LC-200 Right). Pad 39 is used for Russian government and commercial missions, and Pad 40 is inactive.

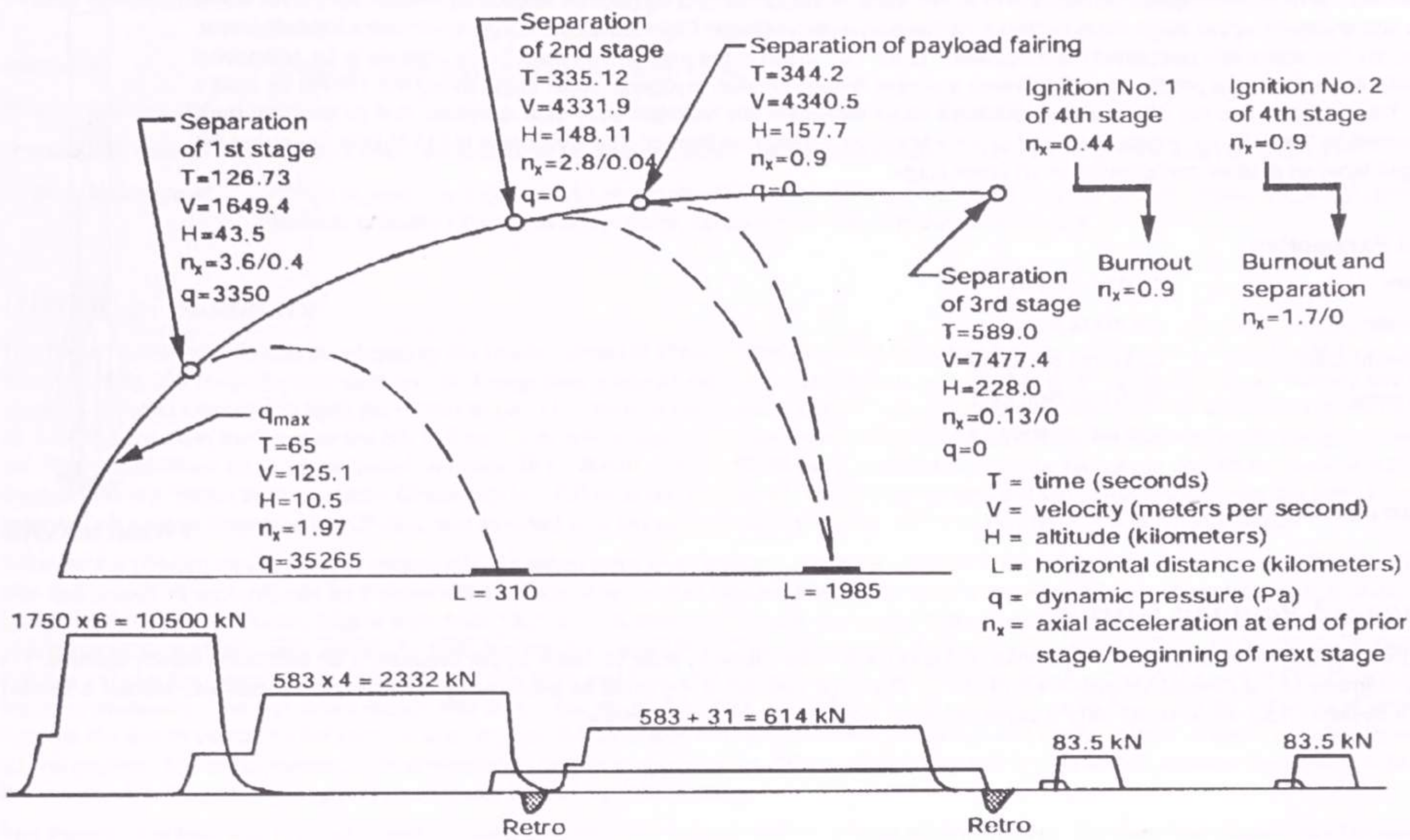
Each pad includes a launch mount and a mobile service tower mounted on railway tracks for access to the vehicle and payload once the rocket is erected. In surge mode, two Protons can be launched from the same pad in about 21 days (Protons have been launched from the same pad in as little as 14 days). As a result of having four available pads, Protons have occasionally been launched as little as four days apart. It takes 4 h to erect and install Proton on the launch pad; typically, this takes place three to five days before launch. Preliftoff maintenance and service operations are carried out with the aid of the mobile service tower. No cable or umbilical tower is incorporated into the launch complex. Many of the services that would be provided by such an umbilical tower are provided by the mobile service tower. Other functions, including fuel loading, are performed by a special servicing mechanism that connects to the Proton at the base of the first stage. All fuel and compressed gas loading operations, including the attaching and removing of connections, are fully automated. The propellant for the vehicle is stored near the launch pad in underground bunkers. The Proton launch complex has undergone evolutionary modifications brought about by changes in the booster's applications. These upgrades include a cryogenic LOX handling system to support the Block DM, a new compressed gas delivery system, and improved on-pad payload thermal control systems.

Approximately 8 h before liftoff, the payload's onboard systems are checked out, and the satellite's readiness for launch is confirmed by checking its telemetry data. Individual safety devices are removed no less than 30 min before launch. Recharging of the onboard chemical batteries is stopped 1.5 h before launch, although trickle charging can continue until 5 min before liftoff. Launch pad personnel are evacuated from the pad area 1.5 h before launch. Shortly thereafter, the mobile service tower is rolled back to a safe distance from the launch pad. Launch go-ahead must be provided by the payload customer no less than 2 min before launch. If a launch scrub occurs, it requires 6–8 h to defuel the booster and an additional 4 h to lower it to a horizontal position for rollback to the horizontal assembly building.

Launch is controlled from a command center located about 1.5 km (1 mi) from the launch complex. At the moment of liftoff, the launch pad servicing mechanism rises with the launch vehicle, tracking the vehicle's movement for the first few fractions of a second. After approximately 20mm of motion, this mechanism is separated and withdrawn by a pneumatic actuator and secured behind an armored steel fire wall cover. This steel cover then helps form part of the launch pad flame deflector.

Production and Launch Operations

Flight Sequence



Courtesy ILS.

Event sequence is shown for Proton K/Block DM. The event sequence for Proton M/Breeze M is essentially similar through third stage shutdown. The timeline for Breeze M flight depends on mission requirements.

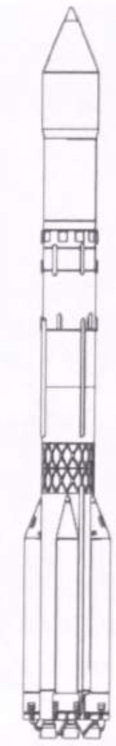
Event Sequence for Proton Launch to GTO with Block DM Upper Stage

Event	Time, s
Stage 1 ignition - 0 up to 50% thrust	*-1.60*
Begin Stage 1 throttle up to 100%	*-0.00*
Stage 1 thrust at 100%	*0.10*
Lift-off	*0.17*
Stage 2 ignition	*122.40*
Stage 1 / 2 separation	*126.60*
Stage 3 vernier engine ignition	*332.0*
Stage 2 engine shutdown	*334.40*
Stage 2 / 3 separation	*335.10*
Stage 3 main engine ignition	*337.50*
Payload fairing jettison	*344.20*
Stage 3 main engine shutdown	*577.10*
Stage 3 vernier engine shutdown	*589.00*
Stage 3 / upper stage separation	*589.10*
Block DM upper adapter jettison	637.00
SOZ unit first settling burn	5437.00
Block DM main-engine first burn	5732.00
SOZ unit shutdown	5737.00
SOZ unit second burn	24,800.00
Block DM main-engine second burn	25,105.00
SOZ unit shutdown	25,105.00
Payload deployment	25,324.00

Vehicle Upgrade Plans

Cryogenic Upper Stage

Since at least the early 1990s, Khrunichev has planned to develop a cryogenic upper stage for Proton, to replace the current Block DM and Breeze M upper stages. The new KVRB upper stage would increase GTO payload to 6600 kg (14,550 lbm). This would be the first cryogenic upper stage implemented on a Russian launch vehicle. Flight tests could begin as soon as 2006, however, Khrunichev has repeatedly postponed the introduction of the new stage in the past (at one point it was planned to be operational by 1996). Khrunichev has produced and delivered a similar, though smaller, cryogenic upper stage called the 12KRB for India's GSLV. This suggests that the difficulties in providing a stage for Proton are financial rather than technical. The stage would likely be powered by the KVD-1M engine produced by KhimMash, or a derivative thereof. The layout puts the LOX tank forward of the hydrogen tank, an unusual configuration in an upper stage.



Proton N/KVRB

KVRB Parameters

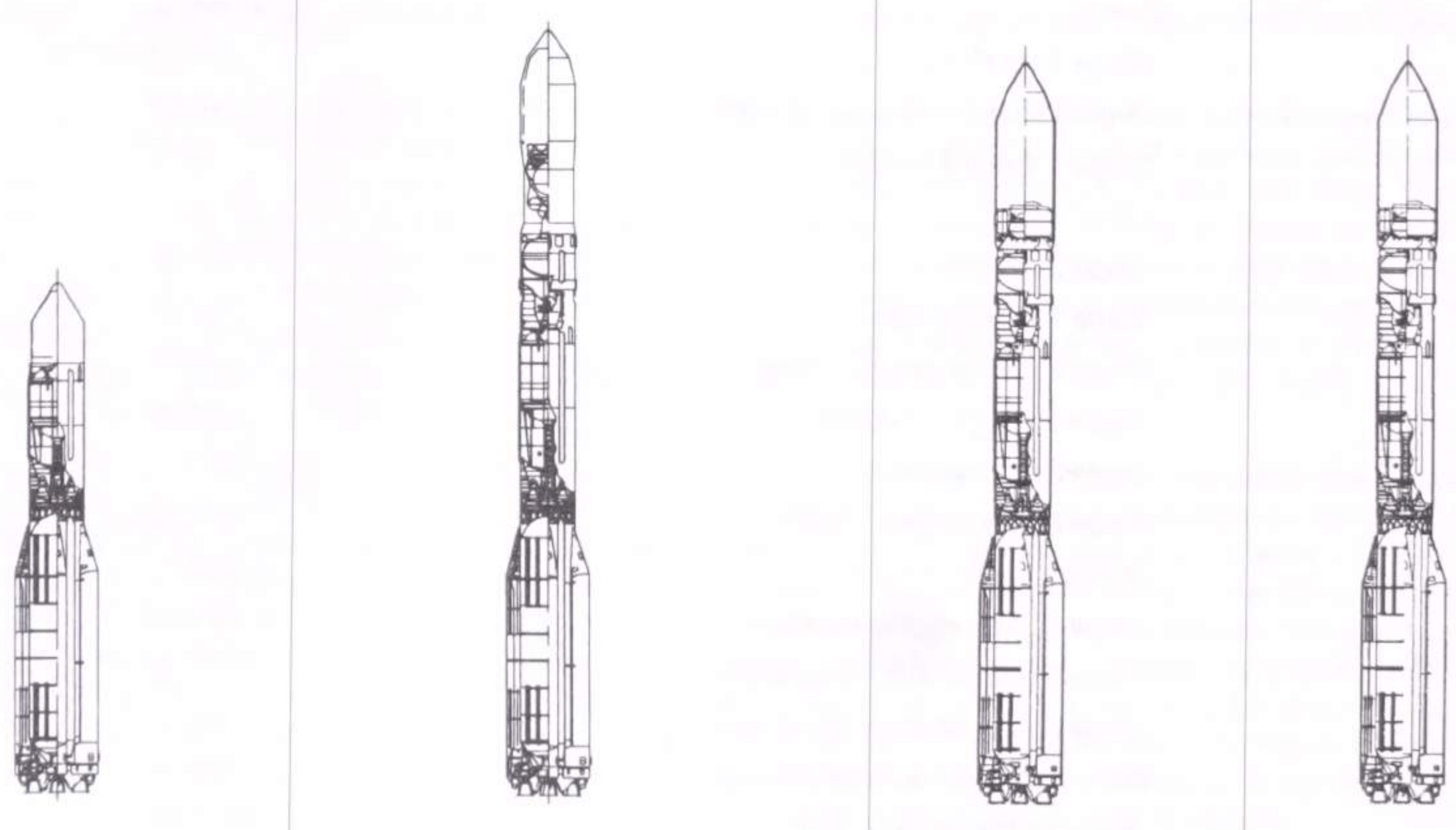
Length	10.0 m (32.8 ft)
Diameter	3.8 m (12.5 ft)
Propellant mass	up to 19 t (42 klbm)
Inert mass	3600 kg (7900 lbm)
Thrust	103 kN (23 klbf)
Isp	461 s
Restart capability	Up to 5 starts

Advanced Payload Fairings

Krunichev has begun development of a payload fairing in the 5 meter class, in order to match similar capabilities on competing launch systems. The usable diameter of the payload volume is 4.6 m (15 ft). The large payload fairing could be available by 2006. Krunichev has also studied a Tandem Launch System (TLS), which could carry multiple satellites. Contact ILS for further details.

Vehicle History

Vehicle Evolution



	Out of Production	In Production		
Vehicle	**Proton**	**Proton K and Proton K/Block DM**	**Proton K/ Breeze M**	**Proton M and Proton M/Breeze M**
Period of Service	1965–1966	Block D: 1967–Present Proton K three stage: 1968–Present Block DM: 1974–Present	1999–Present	2001–Present
LEO Payload	8400 kg (18,500 lbm)	19,760 kg (43,560 lbm)	19,760 kg (43,520 lbm)	21,000 kg (46,300 lbm)
GTO Payload	No capability	4910 kg (10,825 lbm)	4820 kg (10,625 lbm)	5500 kg (12,100 lbm)

Vehicle History

Vehicle Description

•Proton	Two-stage Soviet heavy-lift booster powered by storable propellants
•Proton K/Block D	Four-stage version of Proton. First-stage engines upgraded, first and second stage propellant increased, new third stage based on second stage added, new fourth-stage Block D copied from N-1.
•Proton K	Similar to Proton K/Block D without fourth stage
•Proton K/Block DM	Similar to Proton K/Block D with added instrument compartment on Block DM stage.
•Proton K/Breeze M	Proton K with new Breeze M storable propellant stage. Interim step before introduction of Proton M/ Breeze M.
•Proton M/Breeze M	Proton M similar to Proton K with modernized avionics, upgraded first-stage engines and structures. Block DM replaced by storable propellant Breeze M upper stage, consisting of core unit plus external tank.

Historical Summary

The Proton launch vehicle was developed by the design bureau of Vladimir Chelomei. The Proton was designed to serve as both a heavy missile capable of carrying 100 megaton warheads and as a large space launch vehicle. In competition with his rival chief designers, Sergei Korolev and Mikhail Yangel, Chelomei proposed to build the Proton as part of a family of Universal Rockets of varying sizes and functions. The small UR-100 was deployed as an ICBM known in the West as the SS-11 Sego. In its later evolution it would become the RS-18 (SS-19 Stiletto), which is now being converted into the Rockot and Strela small space launch vehicles. The UR-200 medium ICBM design was beaten out by Yangel's R-36 (SS-9 Scarp), which formed the basis for the Tsiklon launch vehicle. Chelomei's UR-700 ultraheavy-lift launch vehicle design also lost out to Korolev's N-1 for the role of a manned lunar launch vehicle. However, the UR-500 was selected as a heavy-lift launch vehicle in 1961 and was given article number 8K82.

Because the UR-500 was to serve a military role, it needed storable propellants and large engines to burn them. Chelomei turned to Valentin Glushko, who had proposed such engines for Korolev's N-1 booster. Korolev had rejected them, preferring to use less toxic oxygen/kerosene propulsion, but the design was suitable for Proton. Engine tests from 1961 to 1965 demonstrated the propulsion system, and Chelomei's designers considered a number of configurations for the launch vehicle. By 1965 the first two-stage UR-500 was completed. By this time its military role had been dropped. The cost of building silos for the gigantic rocket would have been high, and it is likely that improvements in missile targeting began to make the Proton's huge warheads unnecessary. The first space launch was conducted on 16 July 1965 carrying a Proton physics satellite based on the structure of what would become the launch vehicle's third stage in later designs. A total of four test launches were conducted with the two-stage vehicle now called Proton after its first payload. The performance of this configuration has been quoted at 12,200 kg (26,900 lbm) but this apparently included the burnout mass of the final stage. The true payload capability was closer to 8400 kg (18,500 lbm).

The Proton's first task was to launch manned circumlunar missions as precursors to a lunar landing attempt. The lower two stages were improved, and stretched, the third stage was developed, and a new fourth stage, the Block D was added. The new configuration was called the UR-500K, or 8K82K. Ironically, the Block D was produced by Chelomei's rival, Korolev, as the upper stage for the N-1. Korolev saw the opportunity to use his upper stage in conjunction with the Proton to support his lunar missions, and with his support the lunar program would become the major user of Protons in the late 1960s. The plan was for Proton to send Zond spacecraft—first unmanned test missions, then later manned flights—out to swing around the moon and return, similar to the flight of Apollo 8. This would pave the way for Korolev's N-1 to carry a lunar landing crew to the surface of the moon later. A handful of unmanned missions were conducted carrying either Zonds or Luna robotic probes to the moon. However, Proton (and Zond) could not demonstrate the reliability needed to send a crew to the moon. The initial success rate of the Proton was very poor, resulting in a long period of test flights. Following the success of the U.S. Apollo program, the Russian lunar program was abandoned.

By the mid 1970s the Proton achieved a reliability level similar to other launch vehicles, and became the primary heavy launch vehicle in the Soviet Union. It was used for robotic missions to the planets (primarily Venus and Mars), and for carrying satellites to medium orbits (for the GLONASS system) and to GSO. The three-stage version of the Proton was used to launch all modules of the Salyut and Mir space stations and has also launched elements of the new International Space Station.

In 1983, the Soviet Union unsuccessfully offered Proton commercial launch services to the international Inmarsat organization. One result of this action was that the United States restricted U.S.-built payloads from being flown from the Soviet Union for reasons of technology transfer despite assurances from the Soviets that they would not access the payload. As a result of political conditions, commercial sales of Proton launch services did not take place until after the fall of the Soviet Union, when both Inmarsat and Iridium purchased launch services. In December 1992 a joint venture company was formed between Lockheed Commercial Space Company, a subsidiary of Lockheed Missiles and Space Company of the United States, Khrunichev State Research and Production Space Center of Russia, and Rocket Space Corporation Energia of Russia. This joint venture was approved by both the U.S. and Russian governments, and its incorporation was completed in early 1993. Lockheed–Khrunichev–Energia International (LKE) was later merged into a new organization, International Launch Services (ILS), following the merger of Lockheed with Martin Marietta. ILS serves as the international marketing organization for both Proton and Atlas launch services.

Commercialization of the Proton was made more complex by both internal and international politics. Beginning in 1974, the Chelomei bureau fragmented into three organizations. The design groups split into two companies—KB Salyut, which had some authority over Proton development, and NPO Mashinostroyeniya, which handled the RS-18 ICBM, but has had little involvement with Proton. Both groups shared the use of the Khrunichev factory in Moscow. By the early 1990s, relations between KB Salyut and Khrunichev were openly adversarial. Khrunichev ignored KB Salyut, conducting its own marketing efforts and forming the LKE partnership, while Salyut withdrew its engineers and threatened to move production of Proton to another plant. The Khrunichev plant was more successful at adapting to the changing economic conditions to become more commercial. In 1993, Russian president Boris Yeltsin intervened and ordered KB Salyut and the Khrunichev plant to merge, creating the Khrunichev State Research and Production Space Center in which the Khrunichev plant effectively absorbed KB Salyut. This ended the organizational disagreements.

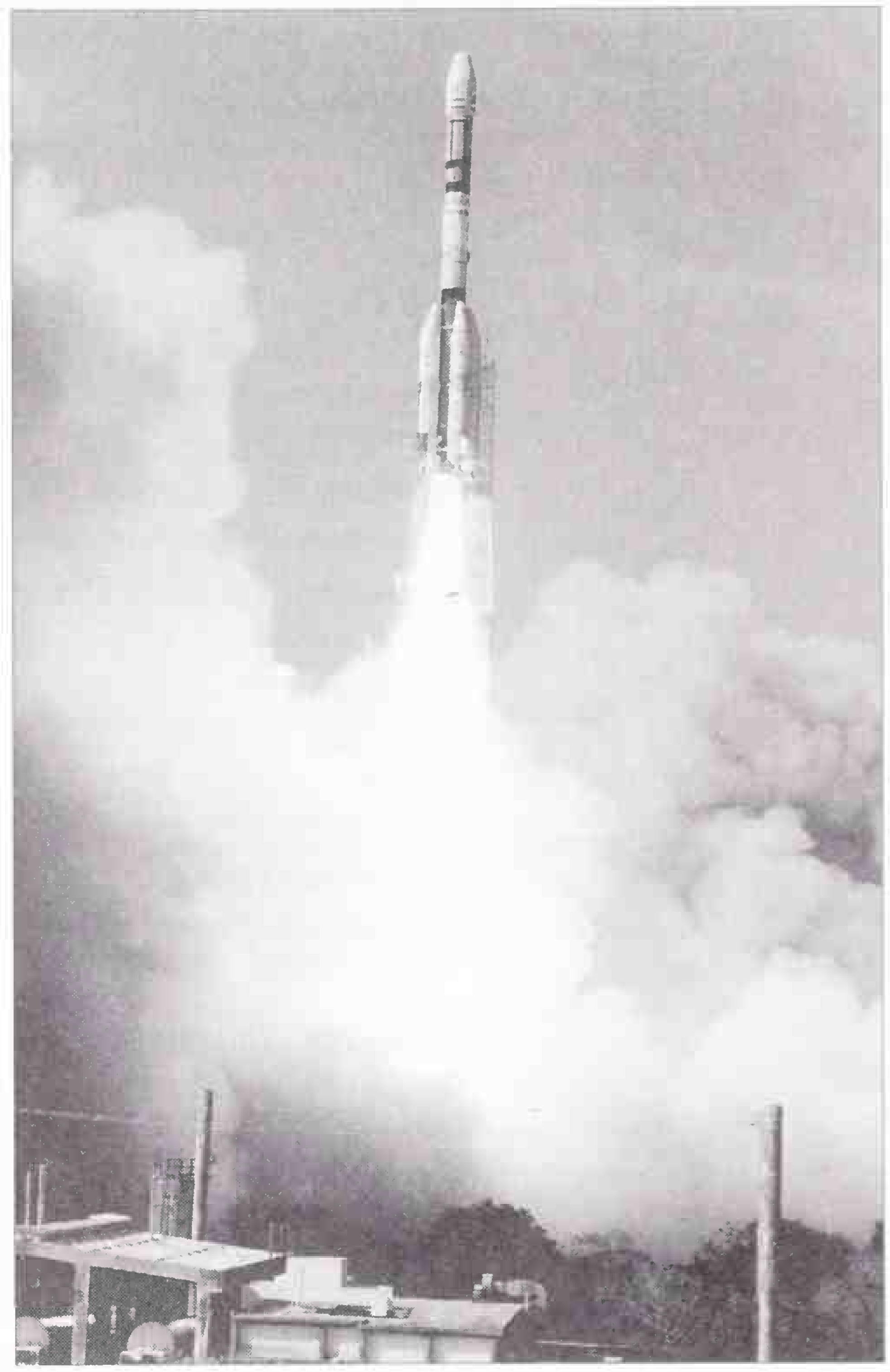
At the same time, U.S. organizations were debating whether to allow Russian launch vehicles to fully enter the commercial launch services market, fearing that a nation with a nonmarket economy would sell launch services far below prices charged by Western companies, driving them out of business. To resolve this, the United States and Russia reached an agreement limiting Russia to 16 launches of commercial GTO payloads (which effectively applies only to Proton) by 2000. Furthermore, Russian launch vehicles would not be allowed to undercut their competitors' prices by more than 15%.

Following resolution of these issues, Proton went on to become the most commercially successful launch vehicle in the former Soviet Union. It has become a mainstream option in the launch services market. The first commercial launch took place in 1996, and by 1998 more than half of Proton launches were for commercial customers. Commercial success has helped the Proton flight rate remain relatively steady, whereas other launch vehicles from the former Soviet Union saw their production decline significantly, or even end altogether. In the long term, Proton is to be replaced by the new Angara launch system, but in the mean time upgrades to Proton continue. The first improved Proton M launched in April 2001, and commercial launches have transitioned to this vehicle.

PSLV AND GSLV

Courtesy ISRO.

India's Polar Satellite Launch Vehicle (PSLV) is designed for delivering India's Earth observation satellites to polar orbit.



Courtesy ISRO.

The Geosynchronous Satellite Launch Vehicle (GSLV) with a cryogenic upper stage is used to place communications satellites into GTO.

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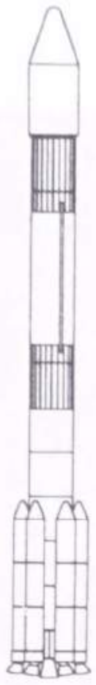
E-mail: kitta@isro.org
Web site: www.isro.org

PSLV

General Description



PSLV

Summary

The Polar Satellite Launch Vehicle (PSLV) is India's third generation launch vehicle to carry applications satellites. PSLV is built domestically to provide independent capability to launch India's one-ton remote sensing satellites into sun-synchronous orbits. India is marketing PSLV launch services internationally.

Status

Operational. First launch in 1993.

Origin

India

Key Organizations

Marketing Organization	Antrix Corporation, Ltd.
Launch Service Provider	Indian Space Research Organization (ISRO)
Prime Contractor	ISRO

Primary Missions

Small sun-synchronous orbit payloads

Estimated Launch Price

\$15–17 million (ISRO, 2002)

Spaceport

Launch Site	PSLV pad at Satish Dhawan Space Centre SHAR
Location	13.7° N, 80.2° E
Available Inclinations	18–100 deg (dogleg required for polar orbits)

Performance Summary

The mass of a mission-specific payload adapter must be subtracted from the performance below to determine available spacecraft mass.

200 km (108 nmi), 49.5 deg	3700 kg (8150 lbm)
200 km (108 nmi), 90 deg	2050 kg (4500 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	3500 kg (7715 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	Standard, 2002: 1350 kg (2975 lbm) Without strap-ons: 900 kg (1985 lbm)
GTO: 200×35,786 km (108×19,323 nmi), 18 deg	1050 kg (2315 lbs) in 2002 increasing to 1200 kg (2645 lbm)
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

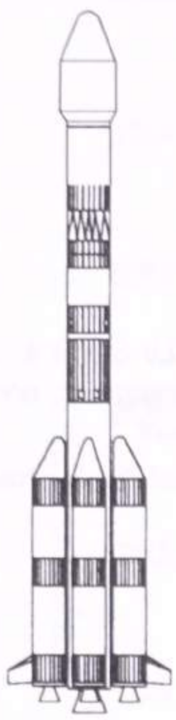
Total Orbital Flights	7
Launch Vehicle Successes	4
Launch Vehicle Partial Failures	2
Launch Vehicle Failures	1

Flight Rate

1–2 per year

GSLV

General Description



GSLV MARK I and MARK II

Summary

The Geosynchronous Launch Vehicle (GSLV) is a derivative of the PSLV, which was developed to provide a domestic launch capability for India's geosynchronous satellites such as the Indian National Satellite (INSAT) series. It improves upon the capability of the PSLV by replacing the solid strap-on boosters with liquid propulsion and adding a cryogenic upper stage to replace the upper two stages of PSLV. For initial flights of the GSLV Mark I, ISRO has used a cryogenic propulsion system purchased from Russia. This will be replaced by a domestically built cryogenic upper stage for the GSLV Mark II.

Status

GSLV Mark I: Operational. First launch in 2001.

GSLV Mark II: In development. First launch planned for 2005–2006.

Origin

India

Key Organizations

Marketing Organization	Antrix Corporation, Ltd.
Launch Service Provider	ISRO
Prime Contractor	ISRO

Primary Missions

GTO payloads

Estimated Launch Price

$35 million (ISRO, 2002)

Spaceport

Launch Site	PSLV pad at Satish Dhawan Space Centre (SHAR)
Location	13.7° N, 80.2° E
Available Inclinations	18–100 deg (dogleg required for polar orbits)

Performance Summary

200 km (108 nmi), 49.5 deg	5000 kg (11,000 lbm)
200 km (108 nmi), 90 deg	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	5000 kg (11,000 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	2000 kg (4400 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 18 deg	GSLV Mark I: 1900 kg (4200 lbm) GSLV Mark II: 2100 kg (4660 lbm) increasing to 2350 kg (5220 lbm)
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Launch Vehicle Successes	1
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	0

Flight Rate

Planned at 1–2 per year

Nomenclature

Indian launch systems are designated Satelllite Launch Vehicles. Each new generation has been given a descriptor, indicating the type or function of the vehicle. For example, the currently operational PSLV is the Polar Satellite Launch Vehicle, specifically designed to deploy remote-sensing satellites into sun-synchronous polar orbits. The PSLV can also be launched without strap-on boosters in a configuration called PSLV-CORE.

SLV-3	Satellite Launch Vehicle
ASLV	Augmented Satellite Launch Vehicle
PSLV	Polar Satellite Launch Vehicle
GSLV	Geosynchronous Satellite Launch Vehicle

Each launch vehicle stage is given two designations. The first indicates the application of the stage. For example PS3 indicates PSLV Stage 3, while GS2 indicates GSLV Stage 2. The second name indicates the propellant type, using S for solid, L for hypergolic liquid, and C for cryogenic, and the approximate propellant mass in metric tons. For example, the PS3 stage is also designated S7 because it has 7 tons of solid propellant.

Flight numbers include a prefix indicating the programmatic status of the vehicle. The letter "D" indicates a development program test flight, while "C" or "F" indicates an operational flight in the continuation program.

Cost

According to ISRO typical launch costs for the PSLV are around $15–17 million, while GSLV costs around $35 million per launch. As of the end of 2002, PSLV has carried four auxiliary commercial payloads. Prices for 100-kg class satellites have been approximately $1 million.

Launch vehicle development and production accounts for 30 to 40 percent of India's space budget. In the 2000–2001 budget, launch vehicles amounted to 6 billion rupees ($125 million), while the estimated 2002–2003 budget includes 8.5 billion rupees ($175 million) for launch vehicles. The current supply of PSLV vehicles are funded in a 6.6 billion rupee ($135 million) multiyear budget allocation for six vehicles.

The development of the GSLV Mark I cost more than 11 billion rupees ($230 million). ISRO is also spending 2.9 billion rupees ($60 milllion) on a second launch pad for GSLV.

Availability

After three developmental flights, the PSLV was declared operational with the flight of PSLV-C1 in September 1997. The GSLV had its first development launch in 2001. The GSLV Mark II is scheduled to be introduced for the third GSLV development flight, expected in 2005 or 2006. Operational flights of GSLV are to begin with the fifth launch in 2004 or 2005. Current planning calls for two PSLV launches and one or two GSLV launches each year. The GSLV is not intended to replace the PSLV. Instead, both launch systems will continue to operate side by side.

PSLV and GSLV are marketed commercially by Antrix, the commercial arm of ISRO. Antrix performed its first commercial launch in May 1999, when a PSLV carried two microsatellites as auxiliary payloads for Germany and Korea followed by another pair of microsatellites in 2001. ISRO and Antrix are interested in continuing to serve this market.

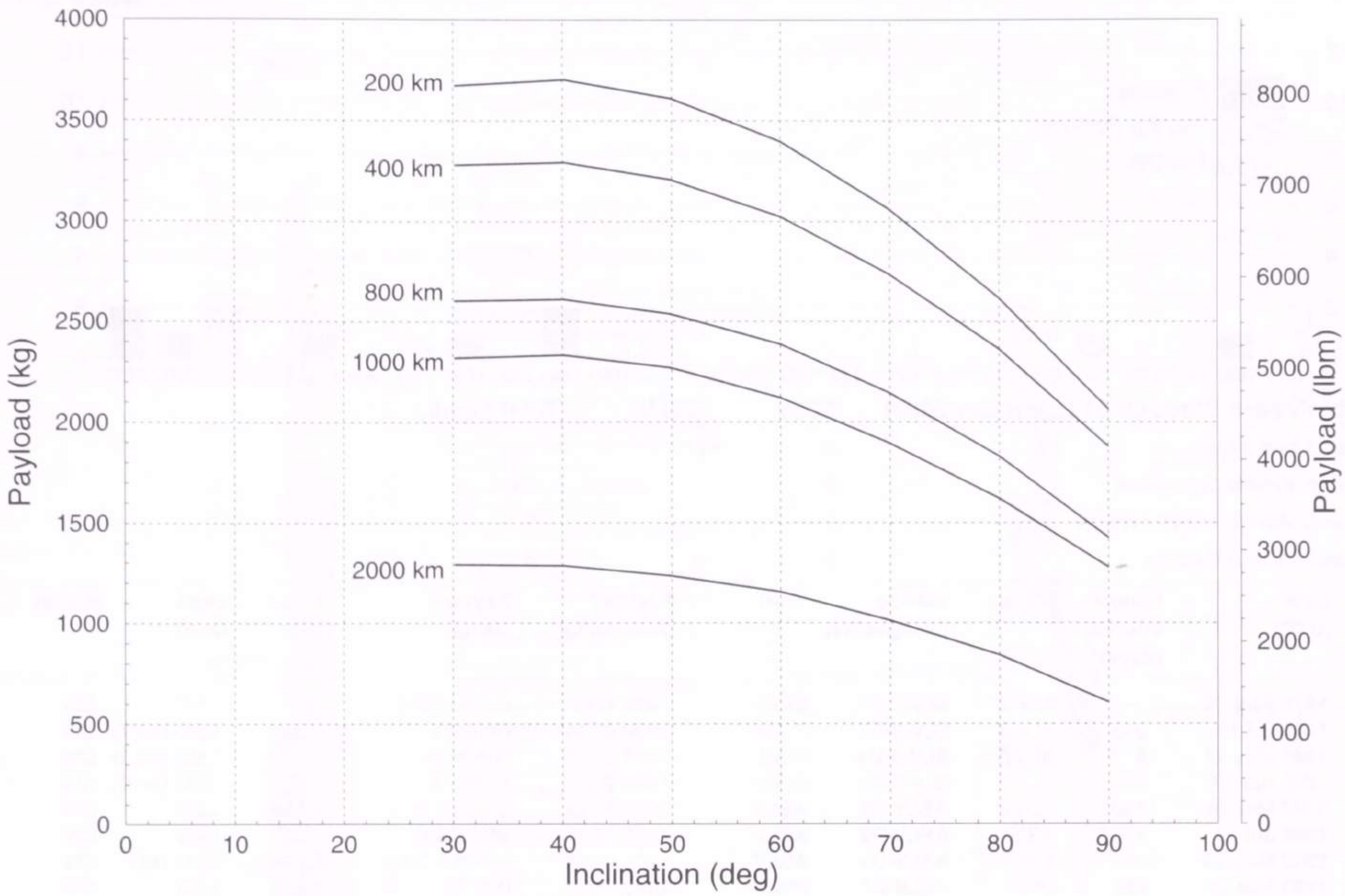
Performance

The ISRO launch vehicle program has followed a strategy of continually upgrading its launch vehicles with carefully planned and qualified improvments. For example, the first PSLV-D1 could carry 820 kg (1800 lbm) to the reference sun-synchronous orbit of 817 km (441 nmi). By the final development flight, PSLV-D3, this had increased to 922 kg (2033 lbm). The first operational flight, PSLV-C1, could carry 1200 kg (2650), further upgraded to 1350 kg (2975 lbm) by PSLV-C4. Performance shown reflects C5 configuration. PSLV can deliver about 540 kg (1190 lbm) to a lunar injection trajectory, or 300–350 kg (660–772 lbm) to Mars. ISRO is planning to take advantage of this capability to launch a lunar probe late in the decade. PSLV can also be flown with fewer than six strap-on motors. A core-only vehicle with no strap-ons has about two-thirds of the performance of a vehicle with all six strap-ons.

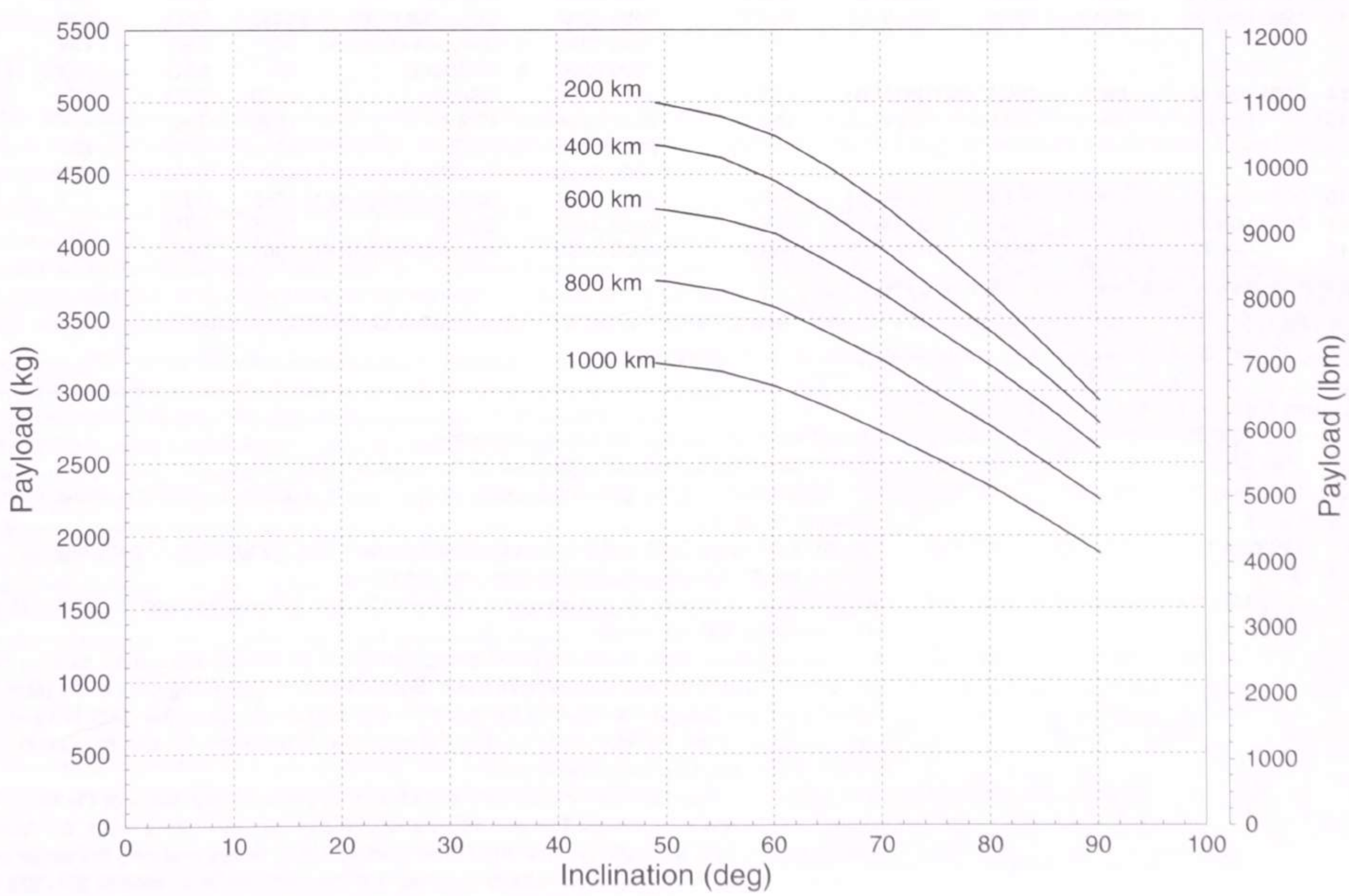
GSLV-D1 had a capacity of 1540 kg (3395 lbm) into GTO. GSLV-D2, launched in May 2003, carried 1825 kg (4025 lbm) using an uprated first stage solid motor and higher pressure engines in the liquid strap-ons. GSLV-D3 will be launched in 2005 or 2006, with a capacity of 2100 kg (4620 lbm) using an Indian-built upper stage in place of the Russian-built stage used initially. Performance is expected to reach 2350 kg (5220 lbm) in later flights.

The Satish Dhawan Space Centre is on the island of Sriharikota, off the eastern coast of India. For Sun-synchronous missions, the launch azimuth of 140 deg is available.The vehicle performs a dog-leg maneuver to reach the final orbit. For GTO missions, the launch corridor at an azimuth of 102 deg is available, resulting in an inclination of 18 deg.

Performance



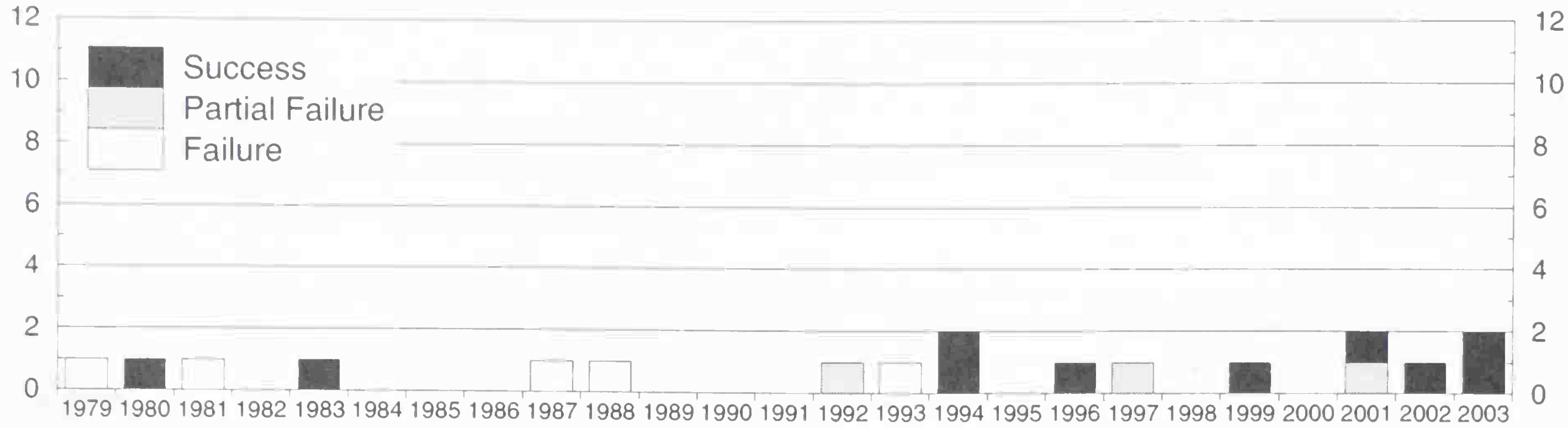
PSLV: Capability for LEO Missions



GSLV: Capability for LEO Missions

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	**PSLV**	**GSLV**	**Total Family**
Total Orbital Flights	7	2	18
Launch Vehicle Successes	4	1	10
Launch Vehicle Partial Failures	2	1	5
Launch Vehicle Failures	1	0	3

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T,F	**1**	**1979 Aug 10**	—	**SLV 3**	**SLV-3-E1**	**SLV3**	1979 F03A	Rohini RS-1	30	LEO	CIV	India
T	2	1980 Jul 18	343	SLV 3	SLV-3-E2	SLV3	1980 062A	Rohini 1	35	LEO (44.7)	CIV	India
T,F	**3**	**1981 May 31**	**317**	**SLV 3**	**SLV-3-D1**	**SLV3**	1981 051A	Rohini 2	38	LEO (46.3)	CIV	India
T	4	1983 Apr 17	686	SLV 3	SLV-3-D2	SLV3	1983 033A	Rohini 3	42	LEO (46.6)	CIV	India
T,F	**5**	**1987 Mar 24**	**1498**	**ASLV**	**ASLV-D1**	**ASLV**	1987 F01A	SROSS A	150	LEO	CIV	India
T,F	**6**	**1988 Jul 13**	**415**	**ASLV**	**ASLV-D2**	**ASLV**	1988 F03A	SROSS B	150	LEO	CIV	India
T,P	**7**	**1992 May 20**	**1408**	**ASLV**	**ASLV-D3**	**ASLV**	1992 028A	SROSS C1	106	LEO (46)	CIV	India
T,F	**8**	**1993 Sep 20**	**488**	**PSLV**	**PSLV-D1**	**PSLV**	1993 F03A	IRS 1E	846	LEO	CIV	India
T	9	1994 May 04	226	ASLV	ASLV-D4	ASLV	1994 027A	SROSS C2	113	LEO (46)	CIV	India
T	10	Oct 15	164	PSLV	PSLV-D2	PSLV	1994 068A	IRS P2	804	SSO	CIV	India
T	11	1996 Mar 21	523	PSLV	PSLV-D3	PSLV	1996 017A	IRS P3	922	SSO	CIV	India
P	**12**	**1997 Sep 29**	**557**	**PSLV**	**PSLV-C1**	**PSLV**	1997 057A	IRS 1D		SSO	CIV	India
	13	1999 May 26	604	PSLV	PSLV-C2	PSLV	1999 029A	OceanSat 1 (IRS P4)	1050	SSO	CIV	India
							1999 029B A	Uribyol 3 (Kitsat 3)	107	SSO	CIV	South Korea
							1999 029C A	TUBSat C	45	SSO	NGO	Germany
T,P,S	**14**	**2001 Apr 18**	**693**	**GSLV Mk1**	**GSLV-D1**	**PSLV**	2001 015A	GSat 1	1540	GTO	CIV	India
	15	Oct 22	187	PSLV	PSLV-C3	PSLV	2001 049A	TES 1	1108	SSO	CIV	India
							2001 049B A	BIRD	85	SSO	CIV	Germany
							2001 049C A	PROBA	94	SSO	CIV	Belgium
	16	2002 Sep 12	325	PSLV	PSLV-C4	PSLV	2002 043A	MetSat 1 (Kalpana 1)	1055	GTO	CIV	India
T	17	2003 May 08	238	GSLV	GSLV-D2	PSLV	2003 018A	Gsat 2	1825	GTO	CIV	India
	18	Oct 17	162	PSLV	PSLV-C5	PSLV	2003 046A	ResourceSat (IRS P6)	1360	SSO	CIV	India

Separate pads are used for the SLV, ASLV, and PSLV.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

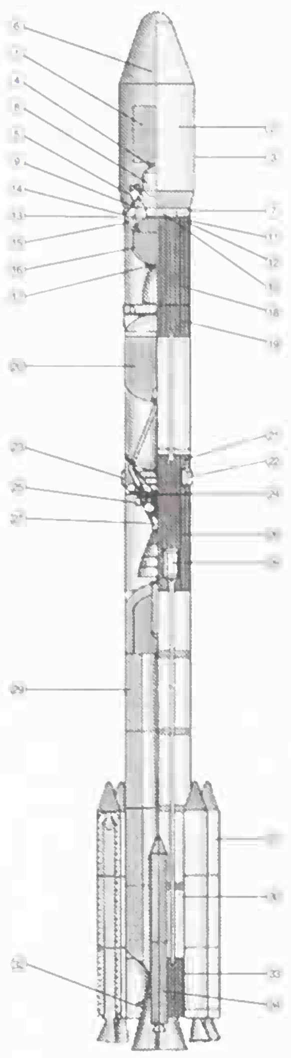
Failure Descriptions:

F	1979 Aug 10	SLV-3-E1	1979 F03	Second-stage control system valve malfunction.
F	1981 May 31	SLV-3-D1	1981 051	Satellite delivered to lower than planned orbit, reentered after 9 days.
F	1987 May 24	ASLV-D1	1987 F01	First stage failed to ignite following booster burnout, due to a failure in either the igniter wiring or the safe/arm device.
F	1988 Jul 12	ASLV-D2	1988 F03	Vehicle broke up at T+50 s due to loss of control caused by a combination of early booster burnout, insufficient autopilot control gain, and wind shear.
P	1992 May 20	ASLV-D3	1992 028A	Stage 4 spun up to only 80 rpm instead of planned 180 rpm, causing injection into 256×435 km orbit instead of 400 km circular.
F	1993 Sep 20	PSLV-D1	1993 F03	A significant attitude disturbance occurred during second- to third-stage separation, causing the attitude control command to exceed its maximum value. Because of a programming error in the pitch control loop of the digital autopilot software in the guidance and control processor, the required reversal of command polarity did not take place, causing the pitch loop to become unstable, resulted in loss of attitude control and failure to acheive orbit.
P	1997 Sep 29	PSLV-C1	1997 057A	Anomalous interaction between the primary and secondary pressure regulators of the fourth stage caused a reduction in propellant flow and thrust after 250 s of burn time. As a result, the fourth stage was shut down by a software override timer after burning 435 s, before reaching the target orbit or depleting propellant. The injection velocity was 140 m/s low, resulting in an orbit of 301×823 km instead of the planned 817 km circular SSO. The spacecraft reached a useable orbit using on-board thrusters.
P	2001 Apr 18	GLSV-D1	2001-015	The cryogenic upper stage shut down about 12 s early for unspecified reasons, leaving the spacecraft with an apogee 4000 km low. ISRO considered this a success given the limited objectives of a test flight. The spacecraft should have been able to reach GEO, but suffered an additional failure.
S	2001 Apr 18	GSAT-1	2001-015	The test spacecraft, which was assembled from spare parts, had two dissimilar propellant tanks. During orbit raising the fuel load in the tanks became unbalanced causing a loss of stability that required additional propellant to correct. The spacecraft could not acheive GEO and was left in a 23-h orbit.

Vehicle Design

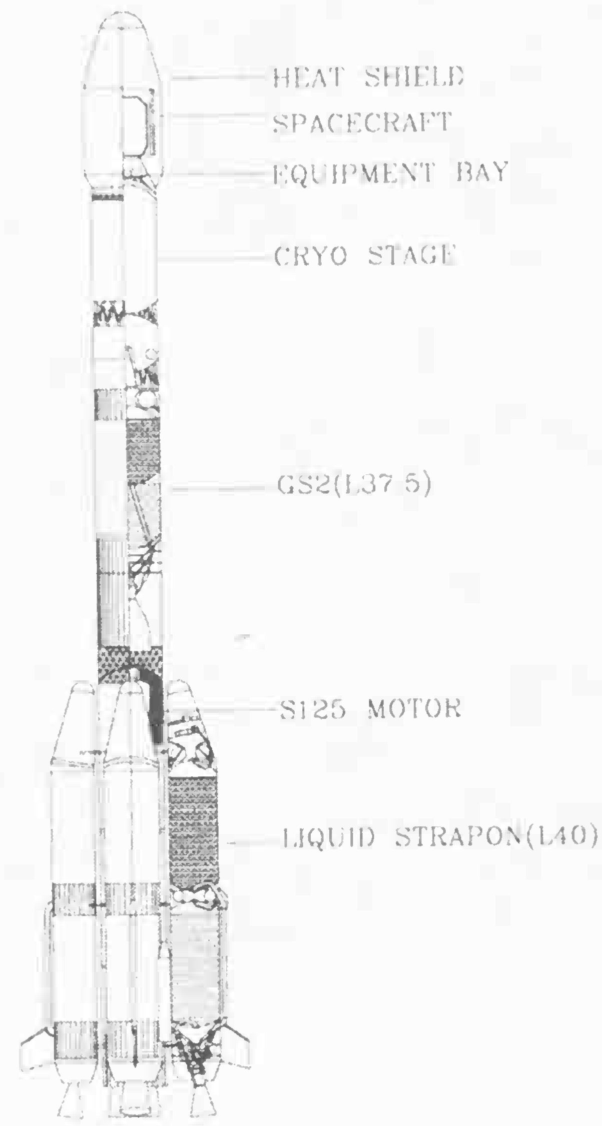
Overall Vehicle

PSLV



1 PAYLOAD
2 HEATSHIELD
3 PAYLOAD SEPARATION PLANE
4 PAYLOAD ADAPTOR
5 EQUIPMENT BAY
6 HEATSHIELD SEPARATION PLANE VERTICAL
7 HEATSHIELD SEPARATION PLANE HORIZONTAL
8 FOURTH STAGE PROPELLANT TANK
9 FOURTH STAGE ENGINE (2)
10 ANTENNAE
11 REACTION CONTROL THRUSTER (6)
12 THIRD STAGE SEPARATION PLANE
13 SECOND STAGE SEPARATION PLANE
14 INTER STAGE 3/4
15 THIRD STAGE ADAPATOR
16 THIRD STAGE MOTOR
17 FLEX NOZZLE CONTROL SYSTEM
18 INTER STAGE 2/3U
19 INTER STAGE 2/3L
20 SECOND STAGE PROPELLANT TANK
21 INTER STAGE 1/2U
22 RETRO ROCKET (4)
23 ULLAGE ROCKET (4)
24 FIRST STAGE SEPARATION PLANE
25 GIMBAL CONTROL SYSTEM
26 INTER STAGE 1/2L
27 SECOND STAGE ENGINE
28 RETRO ROCKET (8)
29 FIRST STAGE MOTOR
30 SITVC INJECTANT TANK (2)
31 STRAP-ON MOTOR (6)
32 SITVC SYSTEM
33 BASESHROUD
34 ROLL CONTROL ENGINE (2)

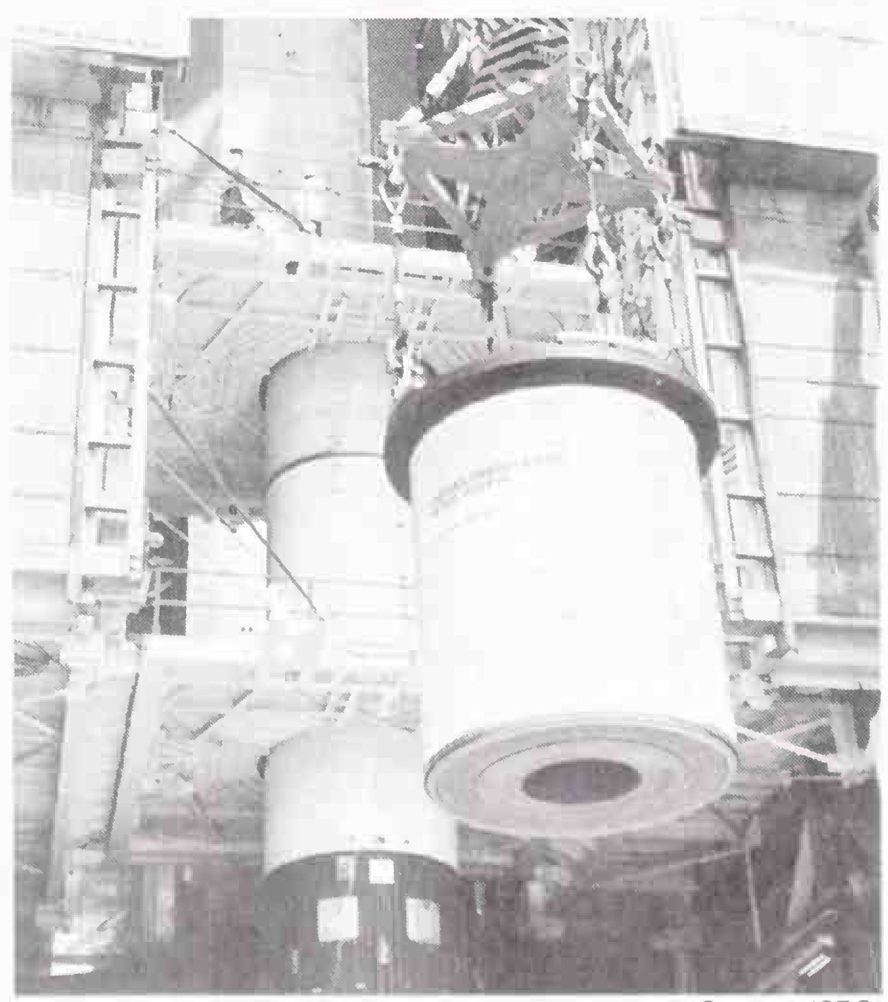
GSLV



	PSLV	GSLV
Height	44.4 m (145.6 ft)	49 m (160 ft)
Gross Liftoff Mass	294 t (648 klbm)	402 t (886 klbm)
Thrust at Liftoff	5290 kN (1190 klbf)	6750 kN (1530 klbf)

PSLV Stages

The PSLV has a unique configuration of alternating solid and liquid stages. The first stage is a large, segmented solid motor, augmented by six solid PSLV strap-on motors. The second stage is powered by a liquid engine, fueled by N_2O_4 and UH25. The third stage is based on a solid motor, and is topped by a small storable-propellant fourth stage for circularization and precision injection.



Courtesy ISRO.

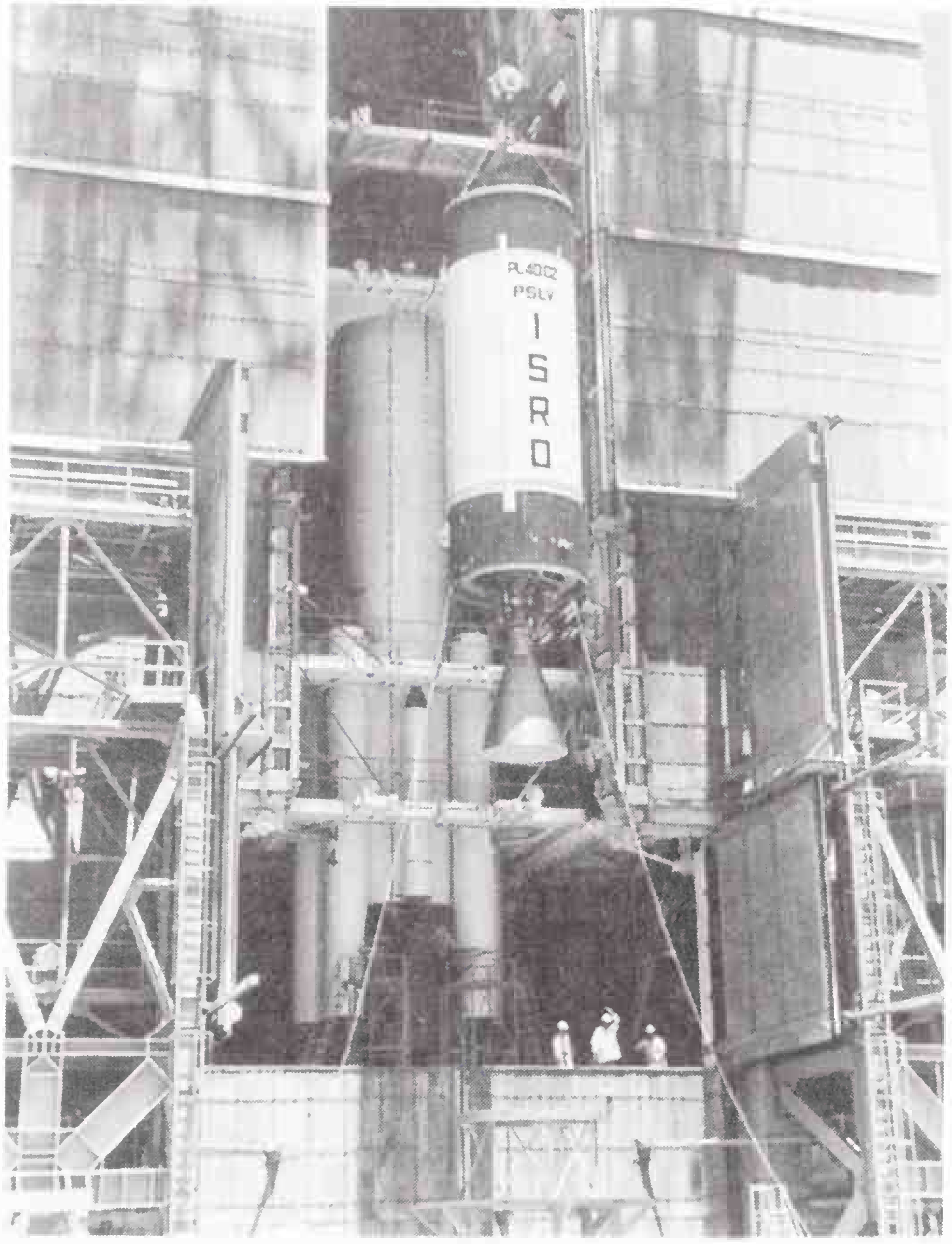
First-Stage Integration

The six PSLV strap-on motor boosters are very similar to the booster and first stage of the previous ASLV, and the first stage of the SLV-3. Four motors are ignited on the ground, while the remaining motors are air lit. To provide improved roll control, one ground-lit and one air-lit motor have secondary injection thrust vector control (SITVC)—also known as liquid injection thrust vector control (LITVC).

The first stage (PS1) of the PSLV is a five-segment solid-rocket motor with HTPB propellant and a composite nozzle. Each segment is 2.8 m in diameter by 3.4 m long (9.2 ×11 ft) and joined by 144 pins. The central segments are interchangeable. At the time of the first PSLV launch in 1993, the first stage was the third largest solid-rocket motor in the world. (The newer Ariane 5 boosters are larger.) Pitch and yaw control of the PSLV during the thrust phase of the solid motor is achieved by injection of an aqueous solution of strontium perchlorate in the nozzle, divergent at 35% of the length of the nozzle from throat to exit. The injectant is stored in two aluminum tanks strapped to the solid-rocket motor and pressurized with nitrogen. The total propellant has been increased from 129 t (284 klbm) in the first PSLV to 138 t (304 klbm) in the operational PSLV vehicles by changing the propellant grain configuration.

The second stage (PS2) is powered by a Vikas engine, burning N_2O_4 and UH25. The Vikas engine is built by ISRO, but is closely based on the Viking IV engine built by Snecma of France. The primary difference is that it is rated for a longer burn time. The engine is powered by a gas generator turbopump. The turbine is driven by hot gases produced in the gas generator by the combustion of UH25 and N_2O_4 and cooled by water spray. The turbine powers two pumps mounted on a single shaft rotating at 9400 rpm, which feed propellants into the combustion chamber. The engine can be gimbaled in two planes and has a radiatively cooled nozzle. The Vikas engine previously used UDMH fuel, but was upgraded in 2003 to use UH25 for an 11% increase in chamber pressure. The stage also includes a pressurization system to maintain minimum specified pressure in the tanks and a pogo suppression system that damps low-frequency longitudinal oscillations during flight. Small solid ullage rockets at the bottom of the stage ensure propellant remains at the bottom of the tanks at ignition. Retro-rockets are used for separation. The propellant tanks share a common bulkhead, and each has slosh suppressors. The thrust frame and interstage skirts are of skin-stringer

Vehicle Design



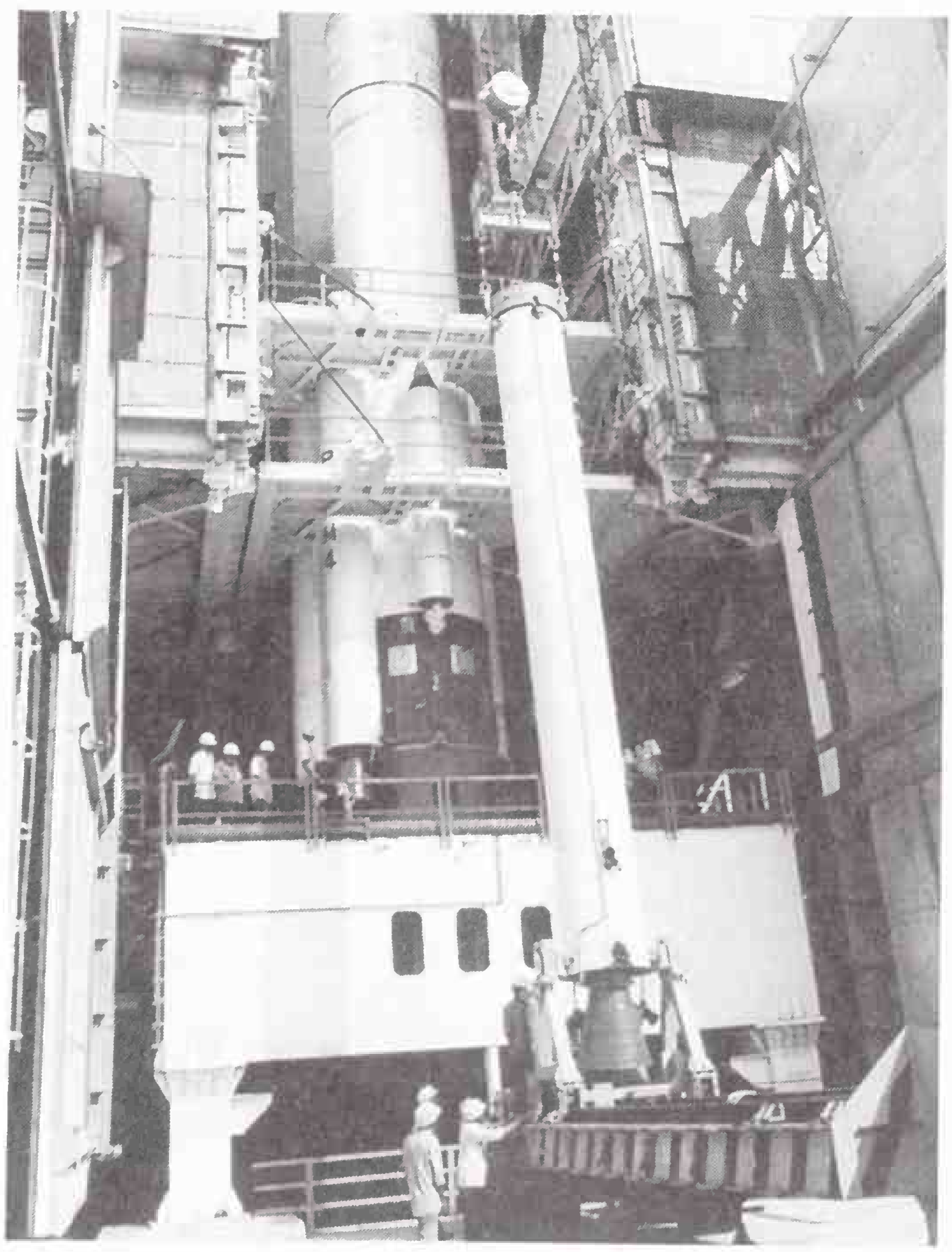
Courtesy ISRO.

Second-Stage Integration

construction. The second stage has been stretched to increase the propellant load from 37.5 t (82.7 klbm) to 40.5 t (89.3 klbm).

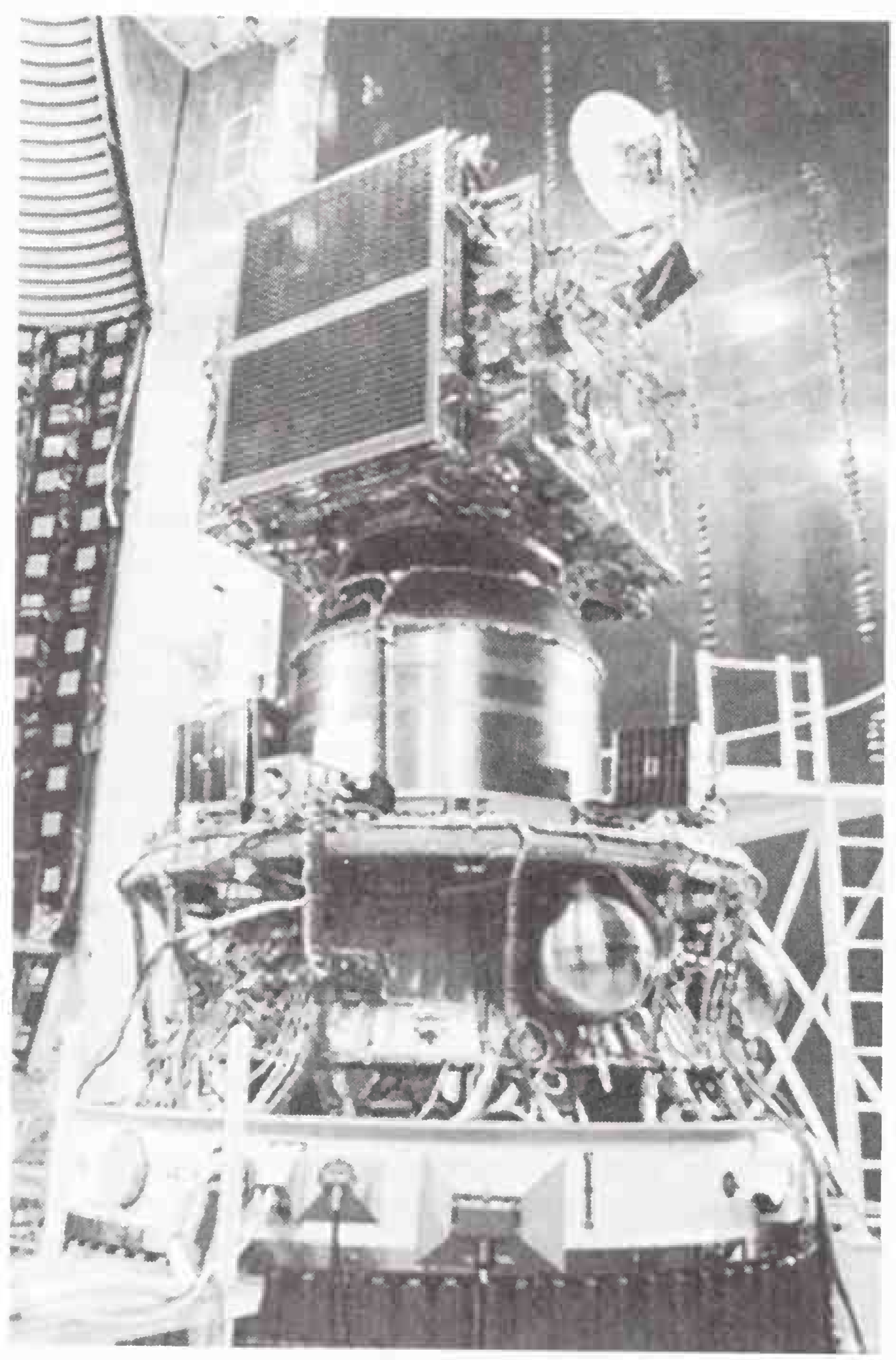
The third stage (PS3) is a solid motor that is located inside an aluminum skin-stringer skirt. It has a polyaramid fiber case and a submerged nozzle equipped with a flex-bearing-seal thrust-vector control system. The older PS3 third stage has been replaced with a high-performance version, HPS3, beginning with PSLV-C4. The adoption of a slower-burning propellant formulation allowed the propellant load to be increased by 300 kg (660 lbm) and the throat area to be reduced, thereby increasing the expansion ratio. Other design changes reduced the inert weight by 100 kg (220 lbs).

The fourth stage (PS4) is used as the terminal stage of the PSLV to provide precision injection. This stage is powered by two identical pressure-fed engines using monomethyl hydrazine (MMH) and MON-3 (3% nitric acid/97% nitrogen tetroxide) as propellants. The propellants are stored in the two compartments of a titanium alloy tank separated by common bulkhead. The tank is designed for an operating pressure of 19.5 bar (282 psi). The tank has a mesh propellant acquisition system (PAS) with catch tanks to ensure sufficient propellant supply for starting the engine under microgravity conditions. The vehicle equipment bay (VEB) is a circular shelf around the propellant tank, supporting the navigation, guidance, and control systems. For the PSLV-C3 mission, the PS4 stage deployed the PROBA microsatellite in a higher orbit than the primary payload, the TES microsatellite, and the Bird microsatellite.



Courtesy ISRO.

Strap-On Motor Integration



Courtesy ISRO.

Integration of IRS-P4, KITSAT-3, and DLR-TUBSAT on the Fourth Stage.

Vehicle Design

PSLV Stages

The following data reflect the PSLV configuration as of 2003.

	Strap-on Boosters (PSOM, S9)	Stage 1 (PS1, S138)	Stage 2 (PS2, PL-40)	Stage 3 (HPS3, S7)	Stage 4 (PS4, L2)
Dimensions					
Length	11.3 m (37 ft)	20 m (66 ft)	12.5 m (41.0 ft)	3.5 m (11.5 ft)	2.9 m (9.5 ft)
Diameter	1.0 m (3.3 ft)	2.8 m (9.2 ft)	2.8 m (9.2 ft)	2.0 m (6.6 ft)	2.0 m (6.6 ft)
Mass					
Propellant Mass	9.0 t (19.8 klbm) each	138.0 t (304 klbm)	40.5 t (89.3 klbm)	7600 kg (16,750 lbm)	2500 kg (5500 lbm)
Inert Mass	2.0 t (4.4 klbm)	30.0 t (66.1 klbm)	5.3 t (11.7 klbm)	940 kg (2068 lbm)	830 kg (1826 lbm)
Gross Mass	11.0 t (24.3 klbm)	168.0 t (76.2 klbm)	45.8 t (101 klbm)	8500 kg (18,700 lbm)	3330 kg (7326 lbm)
Propellant Mass Fraction	0.81	0.82	0.88	0.88	0.73
Structure					
Type	Monocoque	Monocoque	Tanks: monocoque Interstage: skin-stringer	Filament-wound monocoque	Monocoque
Material	Steel	M250 Steel	Aluminum alloy	Polyaramid fiber/ epoxy	Tanks: titanium Structures: composite
Propulsion					
Engine Designation	PSOM (ISRO)	PS1 (ISRO)	Vikas (ISRO)	HPM (ISRO)	PS4 (ISRO)
Number of Engines	6 (3 segments each)	1 (5 segments)	1	1	2
Propellant	HTPB	HPTB	N_2O_4/UH25	HTPB	MON-3/MMH
Average Thrust (Vacuum)	677 kN (152 klbf) each	4430 kN (995 klbf)	804 kN (180.7 klbf)	260 kN (58.5 klbf)	2×7.4 kN (1.7 klbf)
Isp (Vacuum)	262 s	269 s	293 s	295 s	305 s
Chamber Pressure	?	?	58 bar (841 psi)	60.4 bar (876 psi)	8.5 bar (123 psi)
Nozzle Expansion Ratio	8:1	8:1	31:1	69:1	60:1
Propellant Feed System	—	—	Gas generator turbopump	—	Pressure fed
Mixture Ratio (O/F)	—	—	1.7:1	—	1.4:1
Throttling Capability	No	No	No	No	No
Restart Capability	No	No	No	No	1 restart
Tank Pressurization	—	—	Helium	—	Helium
Attitude Control					
Pitch, Yaw	Controlled by Stage 1	SITVC	Nozzle gimbal ±4 deg	Flexseal nozzle gimbal ±2 deg	Electromechanical nozzle gimbal ±4 deg
Roll	SITVC	Roll control thrusters	Hot gas RCS	Stage 4 RCS	50 N (11 lbf) bipropellant RCS
Staging					
Nominal Burn Time	44.2 s	105 s	147 s	115 s	500 s
Shutdown Process	Burn to depletion	Burn to depletion	Command shutdown	Burn to depletion	Command shutdown
Stage Separation	Springs	Linear shaped charge, 8 retro-rockets	Marman clamp, 4 retro-rockets	Ball release	Marman clamp and springs

Vehicle Design

GSLV Stages

The GSLV uses lower stages closely derived from the PSLV, with a new cryogenic third stage replacing the third and fourth stages of PSLV. The GSLV first stage is a combination of a large core solid motor with four liquid-fueled boosters. The core solid motor burns out first, after 105 s, but the liquid engines continue to burn for another 40 s, carrying the central solid motor with them. So, the liquid propulsion modules are not traditional strap-on boosters that are jettisoned in flight, but rather part of an integrated first-stage system comprising both solid and liquid propulsion systems.

The new technology on GSLV is the cryogenic hydrogen/oxygen third stage. This marks the first use of hydrogen/oxygen propulsion in India. For the first two flights, ISRO has purchased upper stages from Khrunichev in Russia, which ironically is now the only significant space power not to use hydrogen on its own rockets. The stage is designated the 12KRB stage in Russia, and is powered by a KVD-1 engine. ISRO is also developing cryogenic technology of its own, as part of the Cryogenic Upper Stage (CUS) program. Long-duration tests of the engine have already been performed, and the first flight of the Indian cryogenic stage is planned for the third developmental GSLV flight around 2005 or 2006.

The specifications provided reflect the configuration of the second vehicle, GSLV-D2. Several upgrades were made for the second flight. The propellant load of the first stage solid motor was increased from 129 to 139 t, matching the configuration that was implemented on PSLV beginning with PSLV-C1. An upgraded Vikas engine has been implemented that burns UH25 (a combination of 75% UDMH and 25% hydrazine hydrate) instead of pure UDMH. This increased the chamber pressure from 52.6 to 58.5 bar (850 psi) and increased GTO performance by 150 kg (330 lbm).

	Stage 1		Stage 2	Stage 3
	(L40)	**(S138)**	**(GS-2)**	**(GS3, C12)**
Dimensions				
Length	19.7 m (64.6 ft)	20.0 m (65.7 ft)	11.6 m (38.1 ft)	8.7 m (28.5 ft)
Diameter	2.1 m (6.9 ft)	2.8 m (9.2 ft)	2.8 m (9.2 ft)	2.8 m (9.2 ft)
Mass				
Propellant Mass	40 t (88 klbm) each	138 t (303.6 klbm)	39 t (85.8 klbm)	12.5 t (27.6 klbm)
Inert Mass	5.2 t (11.4 klbm)	23 t (50.6 klbm)	5.1 t (11.2 klbm)	2.5 t (5.5 klbm)
Gross Mass	45.2 t (99.4 klbm)	161 t (354.2 klbm)	44 t (96.8 klbm)	15 t (33.0 klbm)
Propellant Mass Fraction	0.88	0.85	0.88	0.83
Structure				
Type	Skin-stringer	Monocoque	Tanks: monocoque Interstage: skin-stringer	Skin-stringer
Material	Aluminum	M250 Steel	Aluminum	Aluminum
Propulsion				
Engine Designation	Vikas (ISRO)	S138 (ISRO)	Vikas (ISRO)	KVD-1 (KhimMash)
Number of Engines	4 boosters, 1 engine each	1 (5 segments)	1	1
Propellant	N_2O_4/UH25	HTPB	N_2O_4/UH25	LOX/LH_2
Average Thrust (Vacuum)	760 kN (189.4 klbf)	4801 kN (1079 klbf)	804 kN (180.7 klbf)	75 kN (17 klbf)
Isp (Vacuum)	281 s	266 s	295 s	460 s
Chamber Pressure	58.5 bar (848 psi)	?	52.6 bar (763 psi)	57 bar (826 psi)
Nozzle Expansion Ratio	1.7:1	8:1	1.7:1	200:1
Propellant Feed System	Gas generator turbopump	—	Gas generator turbopump	Gas generator turbopump
Mixture Ratio (O/F)	1.87:1	—	1.87:1	6.0:1
Throttling Capability	No	No	No	No
Restart Capability	No	No	No	1 restart
Tank Pressurization	Helium	—	Helium	Helium
Attitude Control				
Pitch, Yaw	Single plane nozzle gimbal ±8 deg	SITVC	Nozzle gimbal ±4 deg	Two gimbaled auxiliary engines, cold-gas RCS
Roll	Single plane nozzle gimbal ±8 deg	Roll control thrusters	Hot-gas RCS	Two gimbaled auxiliary engines, cold-gas RCS
Staging				
Nominal Burn Time	149 s	105 s	135 s	First burn: 500 s Second burn: 175 s
Shutdown Process	Burn to depletion	Burn to depletion	Command shutdown	Command shutdown
Stage Separation	None	Flexible linear shaped charge, hot separation	Marmon clamp, 4 retro-rockets	Marmon clamp, spring thrusters

VEHICLE DESIGN

Attitude Control System

PSLV

During first-stage flight, pitch and yaw are controlled by a SITVC system. An aqueous solution of strontium perchlorate is injected into the nozzle through 24 ports in four quadrants in a circle positioned 35% of the distance from the nozzle throat to the exit. Each quadrant can generate 200 kN (45 klbf) of side force. The valves are controlled by electromechanical actuators. The strontium perchlorate is stored in two tanks 700 mm (27.6 in.) in diameter, which are strapped to the first stage in the pitch plane. Each tank holds 1650 liters (435 gal) and is pressurized with nitrogen. Roll control is provided by a pair of thrusters mounted below the SITVC tanks that generate 6.4 kN (1400 lbf) of vacuum thrust, and burn NTO/MMH. The thrusters fire continuously during first-stage flight and are swiveled in opposite directions by electromechanical actuators to generate the required roll control moments. Additional roll control is provided by SITVC in two strap-on boosters.

The pitch and yaw control of the second stage is provided by gimbaling the Vikas engine using an electrohydraulic actuator system with ±4 deg range. The actuators are mounted at an angle of 45 deg to the pitch and yaw planes. Roll control is provided by two hot-gas on-off thrusters mounted tangentially and back to back on the interstage. Hot gas drawn from the Vikas gas generator is expanded through thrusters to produce the 600 N (135 lbf) of thrust for roll control.

An electromechanically actuated flex bearing nozzle gimbal system is used to control the third stage in the pitch and yaw axes. The nozzle flex bearing is qualified for ±2 deg deflections. Roll control is provided by the reaction control system (RCS) of the fourth stage.

During the fourth-stage thrusting phase, three-axis control is achieved by gimbaling the two engines. Each engine is supported by a universal gimbal mount enabling two-plane gimbaling with maximum deflection of 4 deg in each axis. The fourth stage also contains a bipropellant RCS for roll control during third flight, and three-axis control during coast phases. It consists of two modules with three thrusters in each module. Each thruster generates 50 N (11 lbf) of thrust using MMH/N_2O_4 propellants drawn from the main propellant tank.

GSLV

GSLV attitude control is similar to PSLV, with a few exceptions. Because the core solid motor of the first stage burns out early, first-stage attitude control is provided by the four liquid booster engines, each of which gimbals in a single plane. The third stage KVD-1 main engine is fixed, but has two gimbaled vernier thrust chambers system for attitude control. During coast phases a cold-gas RCS system is used.

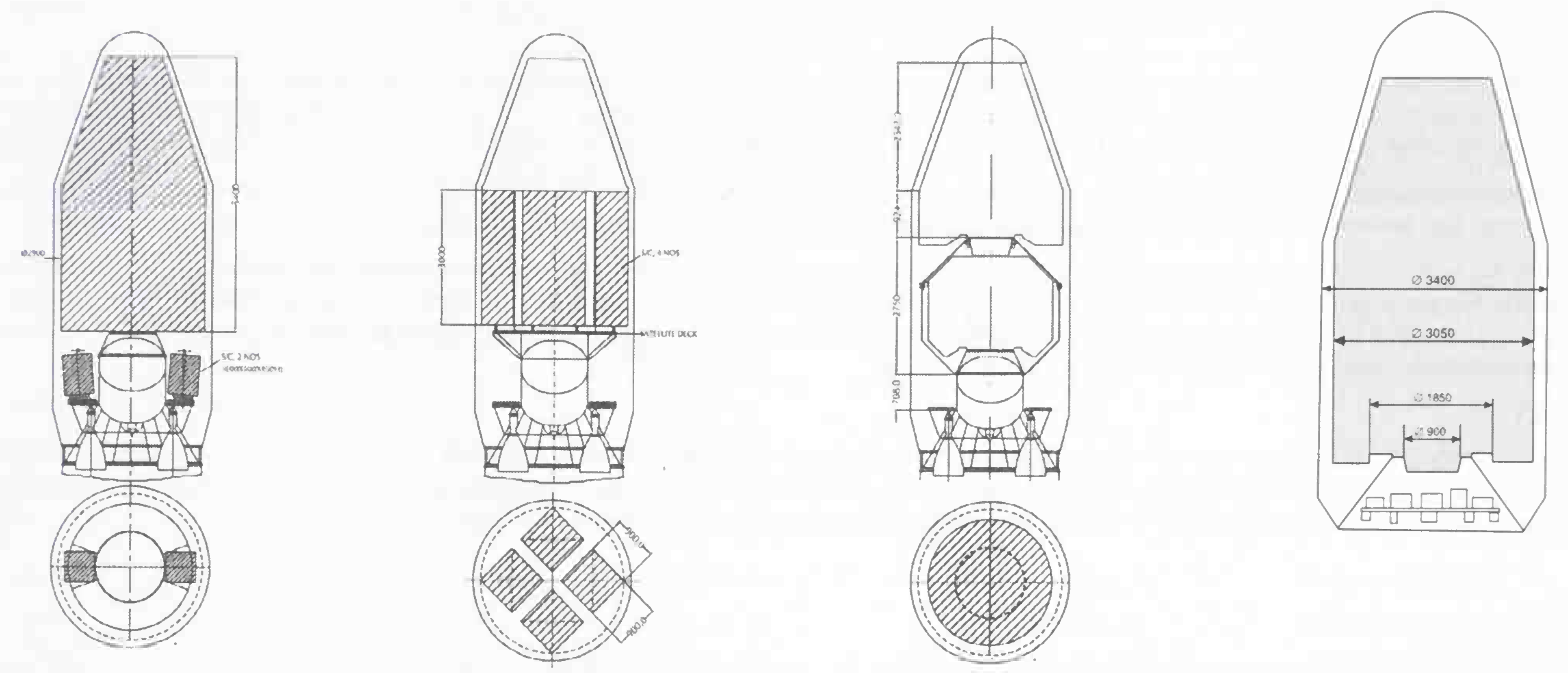
Avionics

The PSLV and GSLV are guided by an inertial guidance system. The redundant strapdown inertial navigation system (RESINS) is designed with three dry-tuned gyros (DTG) in a skewed configuration and four servoaccelerometers positioned along the orthogonal axes, with a redundant unit along the thrust axis. The navigation input is provided to the navigation processor (NGP), which provides data every 500 ms to the guidance and control processor, which issues the steering commands. Navigation software includes preprocessing of sensor signals (fixed drift, bias, scale factor compensation); failure detection and isolation (FDI) logic; transformation from body coordinates to inertial coordinates; and generation of attitude, velocity, and position information. Stage separation is controlled by the real-time decision system. The telemetry, tracking, and telecommand (TTC) system includes a PCM S-band telemetry system and C-band transponders for tracking. It also provides a ground-commanded flight termination capability through third-stage flight. Power for these systems is provided by distributed batteries, with redundancy for critical systems. Most of these components are housed in the VEB, which is a circular shelf around the fourth-stage propellant tank.

VEHICLE DESIGN

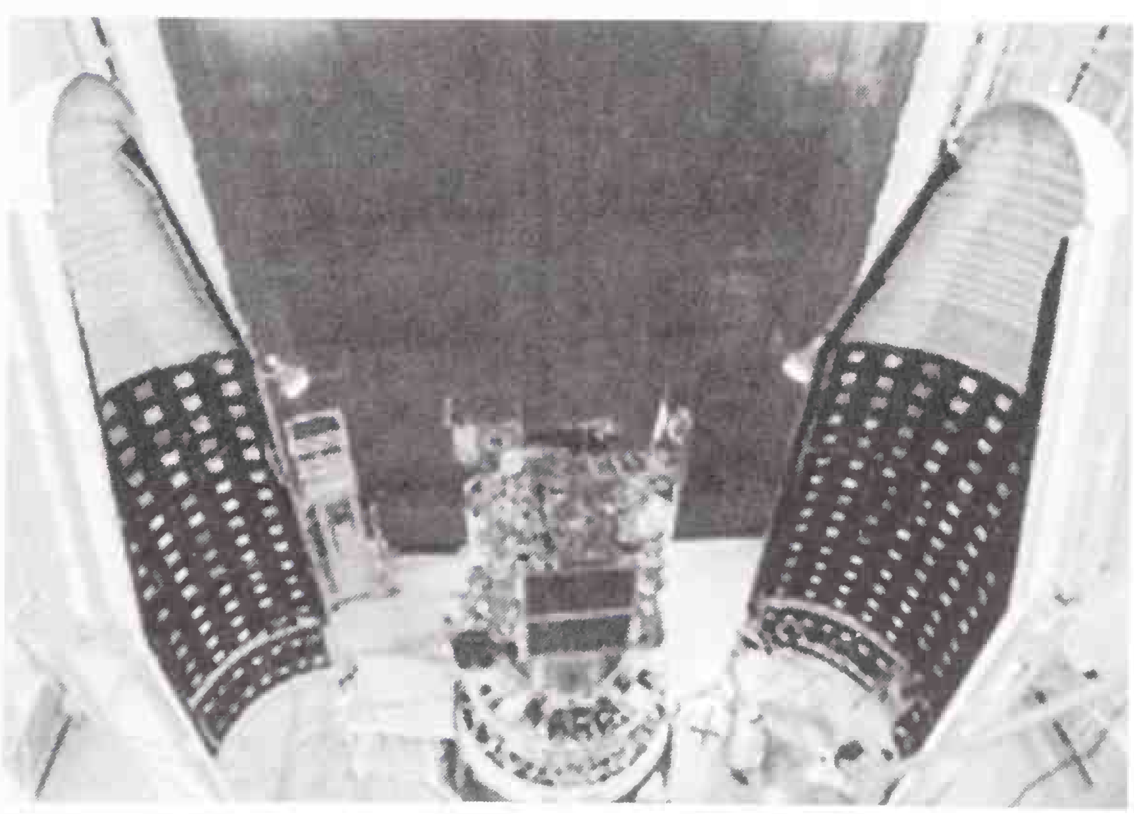
Payload Fairing

The PSLV payload fairing is made of aluminum with an isogrid cylindrical section and a 20 deg forward cone. Separation is provided by a combination of a band clamp, explosive bolts, and zip cord. The fairing is manufactured by Hindustan Aeronautics, Ltd.



Courtesy ISRO.

	PSLV	GSLV
Length	8.3 m (27 ft)	7.8 m (25.6 ft)
Primary Diameter	3.2 m (10.5 ft)	3.4 m (11.2 ft)
Mass	1100 kg (2425 lbm)	1250 kg (2760 lbm)
Sections	2	2
Structure	Skin-stringer	Skin-stringer
Material	Aluminum	Aluminum



Courtesy ISRO.

Encapsulation of the IRS-P4 (Oceansat-1) spacecraft into the PSLV payload fairing. Note: two microsatellites below the primary payload.

Payload Accommodations

	PSLV	GSLV
Payload Compartment		
Maximum Payload Diameter	2900 mm (114 in.)	3050 mm (120.1 in.)
Maximum Cylinder Length	3000 mm (118 in.)	3800 mm (149.6 in.)
Maximum Cone Length	2400 mm (94 in.)	3500 mm (137.8 in.)
Payload Adapter Interface Diameter	937 mm (37 in.)	1194 mm (47 in.)/937 mm (37 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–24 months	T–24 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	?	?
On-Pad Storage Capability	?	?
Last Access to Payload	T–60 h	?
Environment		
Maximum Axial Load	6.4 *g*	5 *g*
Maximum Lateral Load	1.1 *g*	1 *g*
Minimum Lateral/Longitudinal Payload Frequency	40 Hz/18 Hz	31 Hz/10 Hz
Maximum Acoustic Level	137 dB at 250–500 Hz	136 dB at 250–500 Hz
Overall Sound Pressure Level	143 dB	140.4 dB
Maximum Flight Shock	1500 *g*	?
Maximum Dynamic Pressure on Fairing	80 kPa (1670 lb/ft^2)	?
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	?	1 kPa/s (0.15 psi/s)
Cleanliness Level in Fairing	Class 10,000	Class 10,000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	±35 km (19 nmi), ±0.2 deg	± 650 km (351 nmi) apogee, ±5 km (2.7 nmi) perigee, 0.1 deg inclination
Attitude Accuracy (3 sigma)	?	2 deg
Nominal Payload Separation Rate	1 deg/s	1 deg/s
Deployment Rotation Rate Available	?	?
Loiter Duration in Orbit	?	?
Maneuvers (Thermal/Collision Avoidance)	yes	yes
Multiple/Auxiliary Payloads		
Multiple or Comanifest	The PSLV has typically carried single payloads in the past. However, a Dual Launch Adapter (DLA) has been developed and will be used on several flights beginning in late 2005. Copayloads are being sought for some of these flights.	
Auxiliary Payloads	PSLV can carry two 100 kg (220 lbm) auxiliary payloads on each flight. These are mounted to the VEB shelf below the primary payload, and must fit in a 600×700×850 mm (23.6×27.5×33.5 in.) volume.	

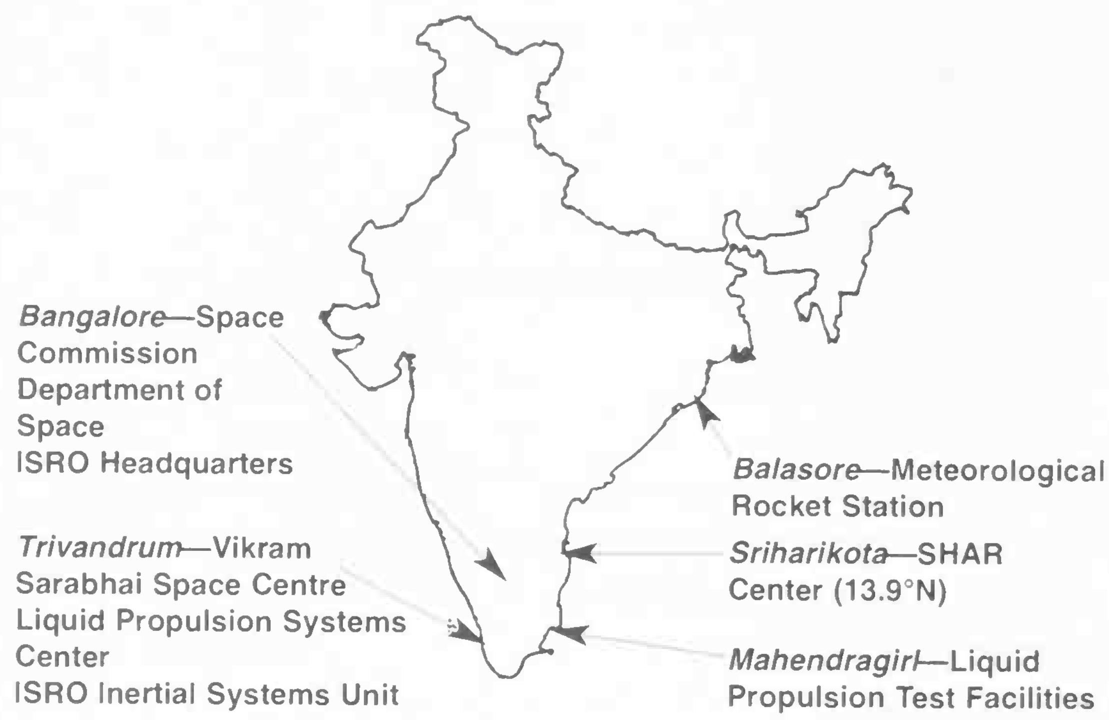
Production and Launch Operations

Production

PSLV and GSLV components are built by ISRO centers in partnership with industry. Vikram Sarabhai Space Centre, located at Trivandrum, is the lead center for launch vehicle development. The Liquid Propulsion Systems Center in Trivandrum is responsible for the Vikas second-stage engines and the fourth-stage engines, as well as spacecraft propulsion systems for ISRO's INSAT and IRS applications satellites. The solid motors are developed at the Vikram Sarabhai Space Center in Trivandrum and cast onsite at SHAR in the solid propellant space booster plant (SPROB) using domestically produced HTPB and ammonium perchlorate. Inertial guidance systems are produced by the ISRO inertial systems unit based in Trivandrum. Hindustan Aeronautics, Ltd. produces structural elements such as the interstages and payload fairings. For the GSLV, ISRO is purchasing seven upper stages from Khrunichev of Russia.

Launch Operations—Satish Dhawan Space Centre, SHAR

In September 2002 India's SHAR space launch site at Sriharikota was renamed in honor of Professor Satish Dhawan, the head of ISRO during its early years. Its official name is now Satish Dhawan Space Centre, SHAR. The center is located on Sriharikota Island off the southeast coast of India in the Bay of Bengal. The facility is connected by road and rail to Chennai (formerly Madras), which has a seaport and international airport. The center is the only orbital launch site in India, but is also responsible for sounding rocket launch facilities at the Thumba Equatorial Rocket Launching Station in southern India and the Balasore Rocket Launching Station on the northeast coast. India generally does not schedule launches from the Satish Dhawan center between October and December—the monsoon season.



The location of the Satish Dhawan center on the southeastern coast of India imposes severe launch window and range safety constraints on launches into polar orbits. A north or south polar launch from Sriharikota is impossible because heavily populated areas in India and Sri Lanka lie in those directions. Therefore, launch azimuth is limited to 140 deg, requiring polar-orbit missions to be launched in a southeasterly direction, followed by an energy-intensive 55 deg yaw maneuver. Similarly, low inclination orbits are restricted to inclinations of about 40 deg or higher, unless yaw maneuvers are used. A launch corridor at an azimuth of 102 deg is available for GTO missions, with a resulting inclination of 18 deg.

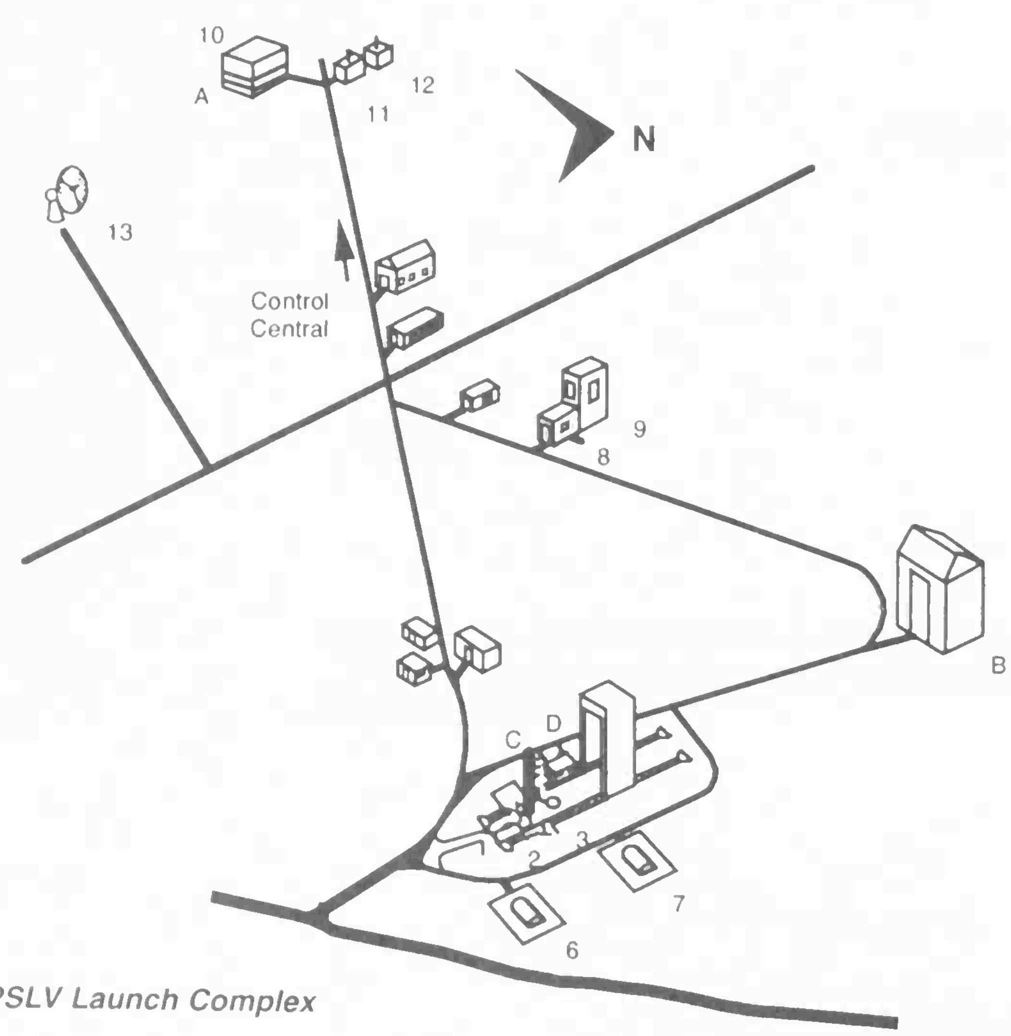
The Satish Dhawan Space Centre has two active launch complex for PSLV and GSLV launches. The existing launch pad consists of a fixed umbilical tower which has been modified to service the GSLV cryogenic upper stage, and a 75-m (246-ft) tall mobile service tower for assembly of the rocket. The mobile service tower has a clean room for payload integration onto the launch vehicle. To provide redundancy and support for the planned increase in flight rate for GSLV and PSLV, a new launch complex uses an integrate–transfer–launch approach. Launch vehicles will be assembled in a fixed vehicle assembly building and transported on a mobile launch base to the pad, which is equipped with an umbilical tower. Two earlier launch pads at Satish Dhawan, for SLV-3 and ASLV, are inactive.

The Launch Control Center and Mission Control Center are located about 6 km (4 mi) from the pad. The SP1 spacecraft facility for initial spacecraft preparation and testing is located near these buildings. The SP2 and SP3 facilities, for spacecraft fueling and integration to the launch vehicle, are located near the pad. All three facilities provide class 100,000 clean rooms. The center also has a meteorological station, tracking and command stations, a guesthouse, and onsite medical facilities.The ISRO telemetry, tracking, and telecommand network (ISTRAC) is composed of stations at SHAR, Trivandrum, Ahmadabad, Bangalore, Lucknow, and Port Blair, Mauritius. SHAR station has telemetry receivers, a dual radar tracking facility, as well as a telecommand facility, which is required for vehicle destruction in case of a malfunction of the onboard system that could result in an unacceptable deviation in the flight path of the vehicle. The stations at Trivandrum, Bangalore, Ahmedabad, Lucknow, and Mauritius are equipped with S-band telemetry and ranging facilities.

Launch Facilities

- A Launch Control Center (LCC)
- B Subassembly preparation hall
- C Subassembly checkout room
- D Checkout Terminal Room (CRT)

1. Launch pad
2. Umbilical tower
3. Service tower track
4. Service tower
5. Jet deflector
6. N_2O_4 overflow tank and feed line terminal
7. UDMH overflow tank and feed line terminal
8. Solid motors prep facility
9. Subassembly prep facility
10. Launch Control Center and Mission Control Center
11. Telemetry STN
12. Telecommand STN
13. C-band radar

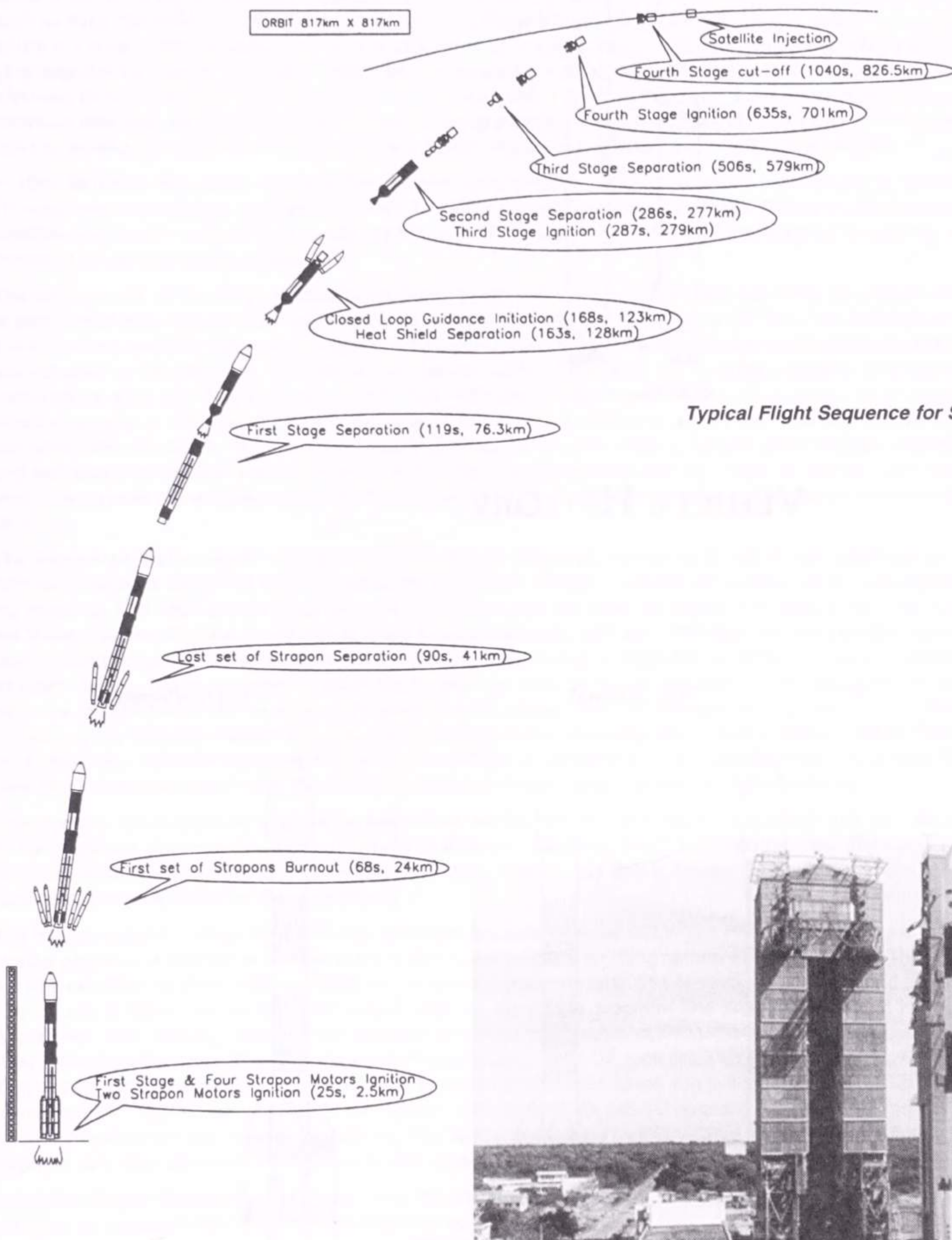


PSLV Launch Complex

PRODUCTION AND LAUNCH OPERATIONS

For PSLV the flight is monitored from SHAR ground stations from liftoff until the end of the long coast phase after third-stage separation when loss of signal occurs. The vehicle is tracked in S-band from Trivandrum ground station from about T–130 s until 100 s after ignition of fourth stage, thus providing overlapping coverage. The entire fourth stage flight is monitored by the downrange station (DRSN) on Mauritius. The tracking of the fourth stage and the spacecraft until injection and of the separated fourth stage with VEB for about 200 s enables DRSN to provide look angle predictions from the S-band telemetry stations for tracking the spacecraft from foreign ground stations and also from SHAR and other Indian stations in the sixth and seventh orbit. Preliminary orbit determination (POD) is also carried out during this phase for declaring the orbit, thus completing the mission.

Flight Sequence



Courtesy ISRO.

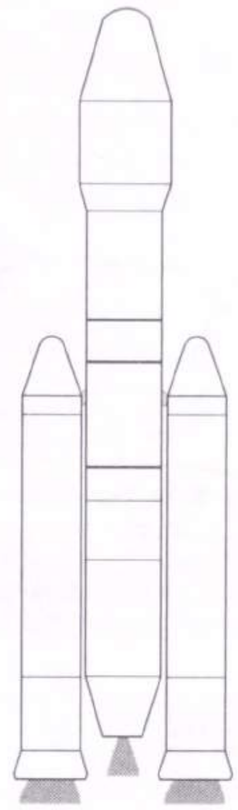
Typical Flight Sequence for Sun-Synchronous Polar Launch

Courtesy ISRO.

PSLV on the launch pad.

Vehicle Upgrade Plans

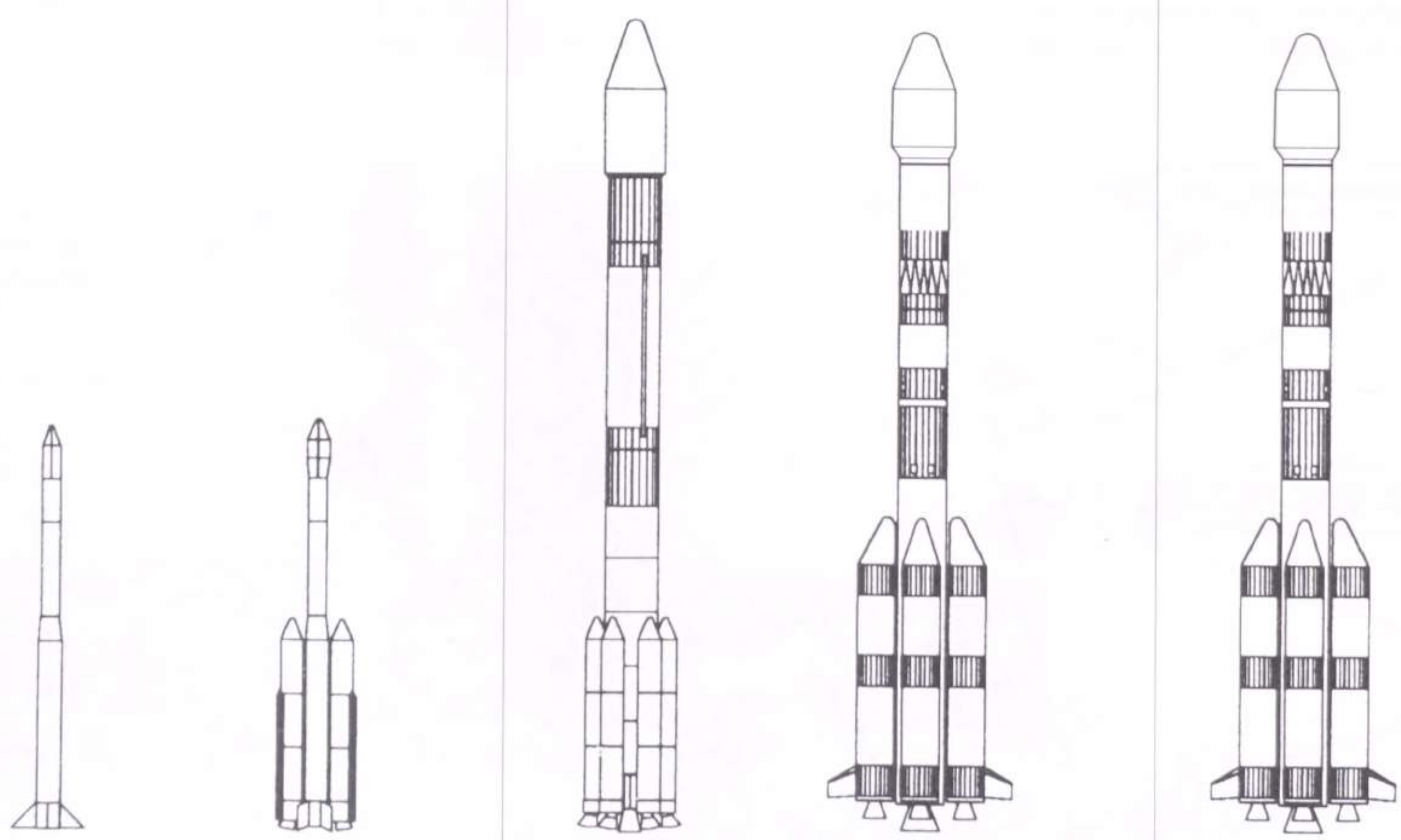
Around 2007–2008, India plans to build a heavy-lift launch vehicle designated the GSLV Mark III. The Mark III will have a more conventional configuration with a large liquid core stage flanked by a pair of solid boosters. All three stages would be new. The twin-engine L110 core stage would be 4 m (13 ft) in diameter, while the S200 strap-ons would be much larger versions of the PSLV first stage motor with 200 t (440 klbm) of propellant. These stages will be topped by a C25 cryogenic stage about twice the size of the present GSLV upper stage, and a 5-m payload fairing. Payload capacity would be around 4000 kg (8800 lbs) to GTO.



GLSV Mark III

Vehicle History

Vehicle Evolution



	Retired		Operational		In Development
Vehicle	SLV-3	ASLV	PSLV	GSLV Mark I	GLSV Mark II
Period of Service	1979–1983	1987–1994	1993–present	2001–present	2005–2006
Payload	LEO: 40 kg (90 lbm)	LEO: 150 kg (330 lbm)	LEO: 3700 kg (8150 lbm) SSO: 1350 kg (2975 lbm)	LEO: 5000 kg (11,000 lbm) GTO: 1900 kg (4200 lbm)	GTO: 2100 kg (4660 lbm)

Vehicle History

Vehicle Description

- **SLV-3** Developmental launchvehicle. Four solid propellant stages, with 0.8-m (2.6-ft) diameter payload fairing.
- **ASLV** Upgraded version of the SLV-3. Two strap-on boosters for first stage, 1-m (3.3-ft) payload fairing.
- **PSLV** New design with six solid strap-on boosters similar to the ASLV, a segmented solid-propellant first stage, storable liquid second stage, solid third stage, and small liquid fourth stage. Payload fairing diameter of 3.2 m (10.5 ft).
- **GSLV** Derived from PSLV by replacing the six solid strap-on boosters of PSLV with four liquid strap-ons similar to the second stage of PSLV. A cryogenic upper stage replaces the last two stages of PSLV.

Historical Summary

India's space launch systems trace their history to the early 1960s and the pioneering efforts of Vikram Sarabhai, considered the father of the Indian space program. At the time, a space program was considered an expensive luxury suited to rich industrialized countries, not to a poor, developing nation such as India. Sarabhai was responsible for convincing Prime Minister Jawaharlal Nehru that space research and applications were not an expensive luxury, but were in fact necessary to the development of the country. As a result, India's space efforts have been focused on applying space technology to improve the lives of its citizens, rather than on symbolic or politically motivated endeavors. Satellite technologies are used to provide India with improved communications, weather forecasting, mapping, and resource monitoring. The primary spacecraft programs in India are the INSAT geosynchronous satellites, and the IRS (Indian Remote Sensing) polar-orbiting Earth observation satellites. The purpose of India's launch vehicle program has been to develop an independent domestic space launch capability to deploy these applications satellites.

In 1962 Sarabhai was put in charge of the National Committee for Space Research, which began by launching sounding rockets from a facility at Thumba, near the southern tip of India. In 1969 the NCSR evolved into the Indian Space Research Organization (ISRO). ISRO was created to build the satellites and launch vehicles India would need in order to provide the communications, weather forecasting, and other services that were considered important for national development.

The development of the first-generation SLV-3 began in 1973 after the success of the suborbital sounding rocket program. The SLV-3 was built in order to orbit Rohini-class experimental engineering satellites, which weighed about 40 kg (90 lbm). The first flight of SLV-3 took place in 1979 and the last in 1983. The first test flight failed on 10 August 1979. It was determined that a valve in the second-stage control system malfunctioned. The second flight test occurred on 18 July 1980. The vehicle successfully launched the 35 kg (77 lbm) RS-1 satellite. The main purpose of the RS-1 was to monitor the performance of the SLV-3 during launch. On 31 May 1981 the third test flight of the SLV-3 launched the 38 kg (84 lbm) RS-D1 satellite. The satellite was intended to stay in orbit for approximately 90 days, but it was delivered to a very low orbit and reentered on 8 June 1981, after only nine days. Instrumentation aboard the spacecraft included a spinscan, solid-state imaging system with a planned resolution of about 0.9 km (0.6 mi). The fourth and last launch of the SLV-3 occurred on 17 April 1983. This launch successfully placed the RS-D2, a 41.5-kg (91-lbm) remote sensing satellite, into orbit. The purpose of the mission was to verify the operation of the solid-state imaging system, and it sent back high-quality pictures of the areas scanned by it.

The second-generation launch vehicle, the ASLV, was an upgraded version of SLV-3. It was developed to deploy stretched Rohini-class satellites (SROSS) weighing about 150 kg (330 lbm) to low Earth orbit. SROSS satellites were designed for selected scientific, technological, and remote sensing missions. The ASLV vehicle used the basic SLV-3 core plus two strap-on boosters for liftoff thrust. The strap-on boosters used the same motor as the first-stage core. The first two launches of the ASLV occurred in 1987 and 1988; both ended in failure. The first flight terminated when the core stage engine failed to ignite after separation of the boosters. This mishap is attributed to either a loose connection or a random malfunction of the igniter safe/arm device. The second vehicle failed shortly after the strap-on boosters burned out. A redesign of the vehicle was undertaken, particularly related to the transition between the strap-on boosters and first stage. The third development flight of ASLV on 20 May 1992, successfully injected SROSS C, carrying two scientific experiments. This flight validated all the corrective actions in the design of ASLV. This was followed by the launch of ASLV-D4 on 4 May 1994. ASLV-D4 successfully injected the SROSS C2 satellite into the intended orbit. The satellite functioned well, giving valuable technical data through its two experiments, the Retarding Potential Analyzer and Gamma Ray Burst Detector.

The SLV and ASLV served as engineering testbeds to develop India's space launch experience, but were too small to launch the IRS and INSAT applications satellites. These were launched primarily on Russian, European, and U.S. launch vehicles. IRS spacecraft were launched on Vostok and Molniya boosters, while INSAT satellites were launched on Delta, Ariane, and Space Shuttle flights. The purpose of the newer PSLV and GSLV is to provide domestic launch capability for these satellites.

The third-generation vehicle, the PSLV, is a significant advance from the ASLV. The PSLV is the first Indian launch vehicle large enough to deliver substantial payloads, and is the workhorse of the Indian space program for lifting remote sensing satellites. The PSLV was designed to deliver remote sensing IRS satellites weighing 1000 kg (2200 lbm) to sun-synchronous orbit. The technology of the second-stage liquid engine, Vikas, was acquired from France and is based on the Viking-IV engine used for the Ariane program. The first developmental launch of PSLV (PSLV-D1) took place on 20 September 1993. Although nearly all the systems performed to expectation, PSLV-D1 could not place the IRS-1E satellite into orbit because of a software implementation error. The second developmental launch, PSLV-D2, took place on 15 October 1994. PSLV-D2 successfully placed the 804 kg (1769 lbm) IRS-P2 remote sensing satellite into an 802 km×875 km (488 mi×533 mi) sun-synchronous orbit. After three developmental launches, the first operational launch, PSLV-C1 occured on 27 September 1997 carrying the IRS-1D spacecraft. A propulsion problem in the fourth stage caused the payload to be deployed to an orbit lower than planned. This flight was followed by PSLV-C2 in May 1999. With this flight, ISRO entered the commercial launch business, and also demonstrated multiple launch capability, by carrying two commercial auxiliary payloads.

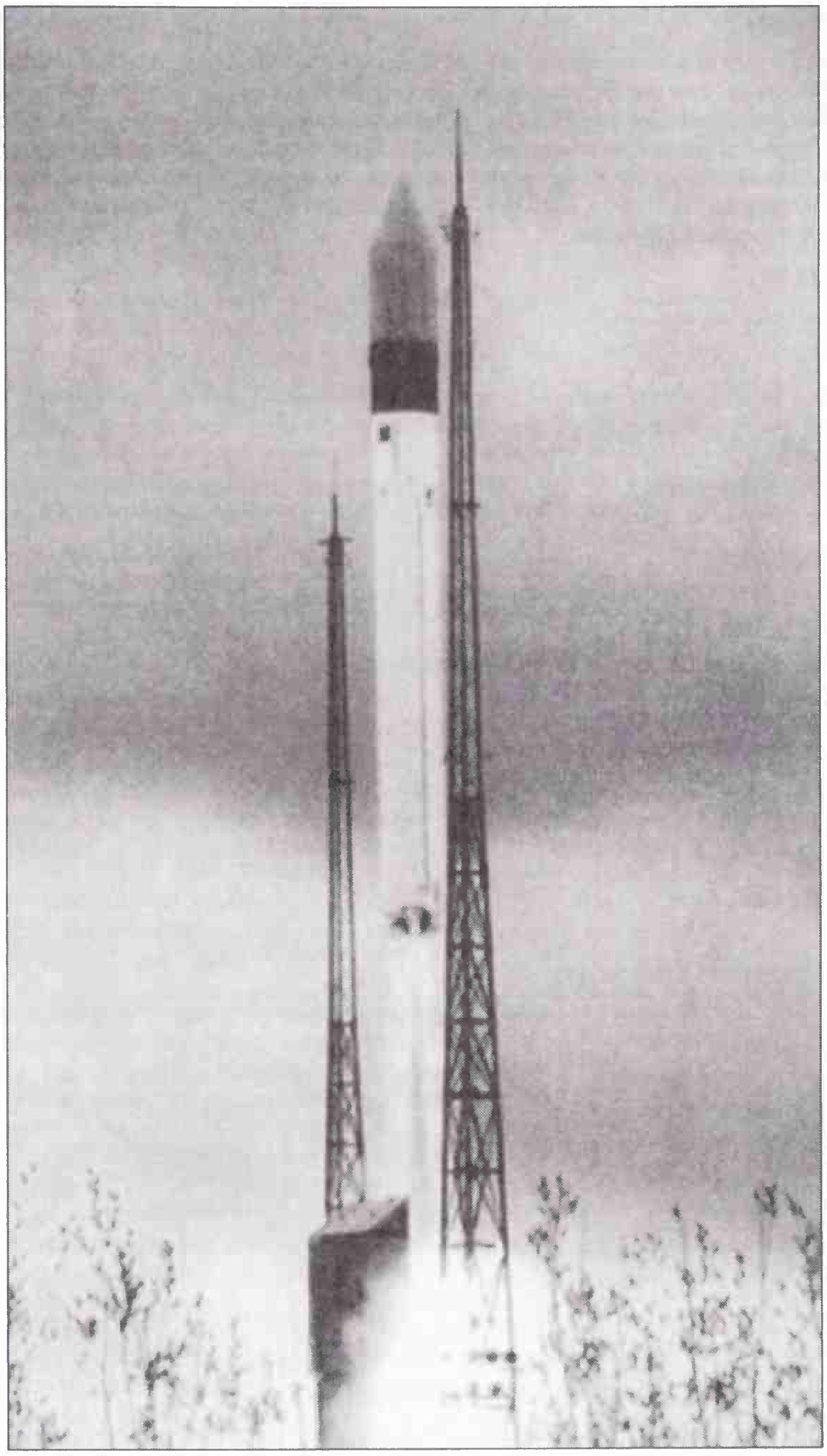
In addition to the IRS satellite series, the other important applications program is the INSAT series of geosynchronous orbiting satellites. The INSAT satellites are unusual in that they combine standard telecommunications and Earth observation/weather monitoring functions on a single spacecraft. The first INSATs were built by Ford Aerospace, but ISRO later began building the INSATs domestically. The purpose of the fourth-generation launcher, the GSLV, is to carry the INSAT satellites to GTO. The GSLV is the most ambitious Indian launch vehicle program to date. The vehicle uses the basic propulsion systems from PSLV for the first two stages. In addition, four liquid strap-on boosters derived from the PSLV second stage augments the booster. The third and final stage is a cryogenic stage burning liquid hydrogen and oxygen.

Vehicle History

At the beginning of the GSLV program, ISRO was concerned that domestic development of cryogenic technologies would take too long to meet the planned GSLV launch date. Therefore, in 1991 ISRO attempted to purchase the technology to build cryogenic engines and stages from Russia. While Russia has no operational cryogenic upper stages of its own, the technology for them was developed as part of the Soviet manned lunar program. A contract was signed with Glavkosmos, the former Soviet organization for commercialization of space technology, to deliver two completed upper stages and the technology for India to build its own. The United States objected to the deal, arguing that the transfer would violate the Missile Technology Control Regime, a nonproliferation agreement that prevents signatory countries from allowing the transfer of missile technologies to other nations. India disagreed, pointing out that cryogenic upper stages were not useful for missile applications. Nevertheless the United States imposed trade sanctions on ISRO and Glavkosmos. In response the contract was modified in 1994 to purchase seven Russian upper stages and two ground test articles, but not the basic technology itself. Instead, India will use the additional stages to operate the GSLV until domestic cryogenic technology can be developed. The first Russian upper stages were delivered in 1998.

Although the first launch of GSLV had once been planned as early as 1995, a combination of delays associated with the international dispute and normal technical difficulties during development extended the program significantly. The first GSLV Mark I was finally readied for launch in early 2001. A launch attempt on 28 March was aborted a few seconds before liftoff by a partial blockage in a feedline of one of the four booster engines. The faulty engine was replaced and the launch was performed less than three weeks later. The GSAT-1 spacecraft, which had been assembled as a demonstration payload for this test flight, was not able to reach GEO because of a problem with its propellant tanks. Nevertheless, the launch was a significant milestone for India, making it one of only a handful of nations with a domestic capability to launch geosynchronous satellites. Within a few years, ISRO expects to be ready to replace the Russian cryogenic upper stage with a domestic one. Full duration tests of the Indian cryogenic engine were performed in 2002, and first launch on the GSLV Mark II is planned for 2004.

Rockot



Courtesy Khrunichev.

Rockot is a small space launch vehicle based on Russia's RS-18 (SS-19) ICBM.

Contact Information

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Web site: www.khrunichev.com

Rockot

General Description

Rockot

Summary

Rockot consists of a decommissioned Russian RS-18 (SS-19) ICBM with a new Breeze third stage and new payload fairing. Rockot is marketed internationally for commercial launches by the German–Russian joint venture company EUROCKOT Launch Services GmbH. Test flights were conducted in the early 1990s, and commercial launches began in 2002. Rockot is suitable for payloads up to 1950 kg (4300 lbm) to medium and high inclination LEO and highly elliptical orbits. The addition of the steerable and restartable Breeze upper stage allows Rockot to reach inclinations other than those directly accessible from the limited launch azimuths of Russian launch sites. This is a distinct advantage over other small Russian launch systems.

Status

Operational. First flight in 1994.

Origin

Russia

Key Organizations

Marketing Organization	EUROCKOT Launch Services GmbH
Launch Service Provider	Khrunichev State Research and Production Space Center
Prime Contractors	Khrunichev State Research and Production Space Center and EADS Launch Vehicles

Primary Missions

Small satellites to LEO and highly elliptical orbits

Estimated Launch Price

$12–15 million (FAA, 2002)

Spaceports

Launch Site	Plesetsk LC 133 (11P865PR)
Location	62.8° N, 40.6° E
Available Inclinations	63, 75, 82–86.4 deg directly, 63–108 deg available with dogleg trajectory or in-orbit plane changes

Performance Summary

The mass of a mission specific payload adapter must be subtracted from the performance shown below to determine the available spacecraft mass.

200 km (108 nmi), 63 deg	1950 kg (4300 lbm)
200 km (108 nmi), 90 deg	Nonstandard mission profile. Contact EUROCKOT for information
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	1000 kg (2205 lbm)
Highly elliptical orbits: 200×27,800 km	Up to 750 kg (1654 lbm)
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

In addition to the orbital flights listed below, Rockot has flown two suborbital test flights.

Total Orbital Flights	6
Launch Vehicle Successes	6
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–2 per year

Nomenclature

The name Rockot (also spelled Rokot) is translated roughly as "rumble" or "roar." The Rockot configuration uses a Breeze (or "Briz") upper stage on top of the RS-18 booster stages. Initial test flights used the Breeze K version. Operational missions use the modified Breeze KM and Breeze KS. These versions of Breeze are related to but distinct from the Breeze M configuration to be used on the Proton M.

Name	Article
Breeze K	*14S12*
Breeze KM	*14S45*
Breeze KS	*?*
Breeze M	*14S43*

Cost

EADS invested $35–40 million in the Eurockot venture, primarily to upgrade launch facilities closer to Western standards. It is unclear how much Khrunichev has invested, but the availability of existing missile assets has certainly reduced the start-up costs of the program. Based on open-source information, the FAA estimated in 2002 that the typical launch cost for Rockot was $12–15 million. According to Khrunichev director Alexander Medvedev, the launch price for the GRACE mission was around $10 million. However, this was a low, introductory rate to enter the market, and future missions may be more expensive. Launches of small auxiliary payloads are offered at $10,000 per kg.

Availability

The typical time from contract signing to launch of a Rockot is 18 months. A launch campaign typically takes around 40 days. Eurockot offers the "LaunchaPiggy" program, in which occasional launches will carry clusters of microsatellites that would normally piggyback with a primary payload on larger launch vehicles.

About 360 RS-18 missiles were produced between 1977 and 1991. In January 2002 Russia had 137 deployed missiles. Forty-five missiles have been assigned to Khrunichev. The guaranteed service life is 21 years, although this can be extended. Rockot could be phased out around the end of the decade, as the missiles reach the end of their service lives, and Angara and Vega begin to take over many of Rockot's missions in Russia and Europe respectively. However, Rockot will be less expensive than either of these vehicles, so it may continue to operate for many years to come.

Performance

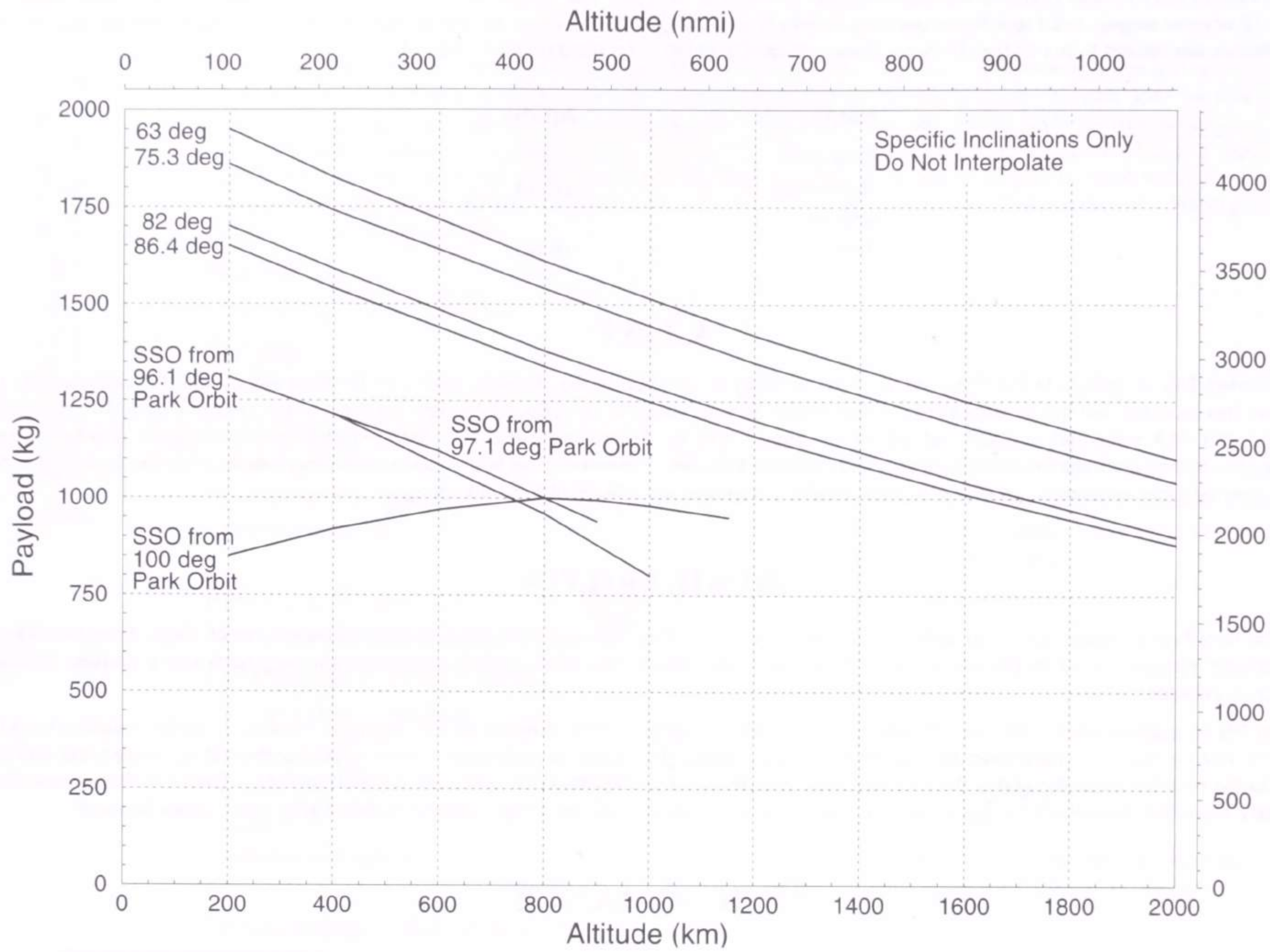
Rockot is launched from Plesetsk Cosmodrome in Russia. Because the launch site is landlocked, specific drop zones away from populated areas are reserved for the impact of separated rocket booster stages. Launch azimuths are therefore limited to those that will result in impacts in these zones. To reach orbit inclinations corresponding to other launch azimuths, Rockot trajectories can include a dogleg maneuver during the second-stage burn. If necessary, the restartable Breeze stage can also perform plane change maneuvers once in orbit to change inclination more. The multiple burn capability is also used to perform the circularization burn for orbits above approximately 400 km. Available launch azimuths for Rockot launches from Plesetsk are 90, 31.5, 15.2–4.8, and 341.5 deg. These correspond to inclinations of 63, 75.3, 82–86.4, and 99 deg, respectively. However, to avoid a second-stage impact in foreign territorial waters, launches along the 341.5-deg launch azimuth must perform a dogleg during the second-stage burn, which results in injection into either a 96.1, 97.1, or 100-deg-inclination transfer orbit rather than the expected 99 deg. Sun-synchronous or other retrograde orbits are reached from one of these three transfer orbits using a plane change maneuver. Thus, the performance curve for sun-synchronous orbits has three distinct segments depending on which initial parking orbit is used. Because dogleg or plane change maneuvers reduce performance, it is not accurate to interpolate vehicle capability for inclinations between the curves provided here. To determine the available payload mass, the mass of a mission specific payload adapter and separation system must be subtracted from the performance shown. The performance shown reflects 3 sigma flight performance reserves.

In addition to the LEO capability shown, Rockot can launch up to 750 kg (1650 lbm) to a 200×27,800 km (108×15,000 nmi) elliptical orbit using the new Breeze KS upper stage. Rockot has also been selected to launch several very small Russian GEO satellites. However, it cannot deliver them to GTO. Rather, they will be launched into LEO and use onboard electric propulsion systems to raise themselves to GEO.

Courtesy EUROCKOT Launch Services

Preparation for the launch of the Commercial Demonstration Flight

Performance

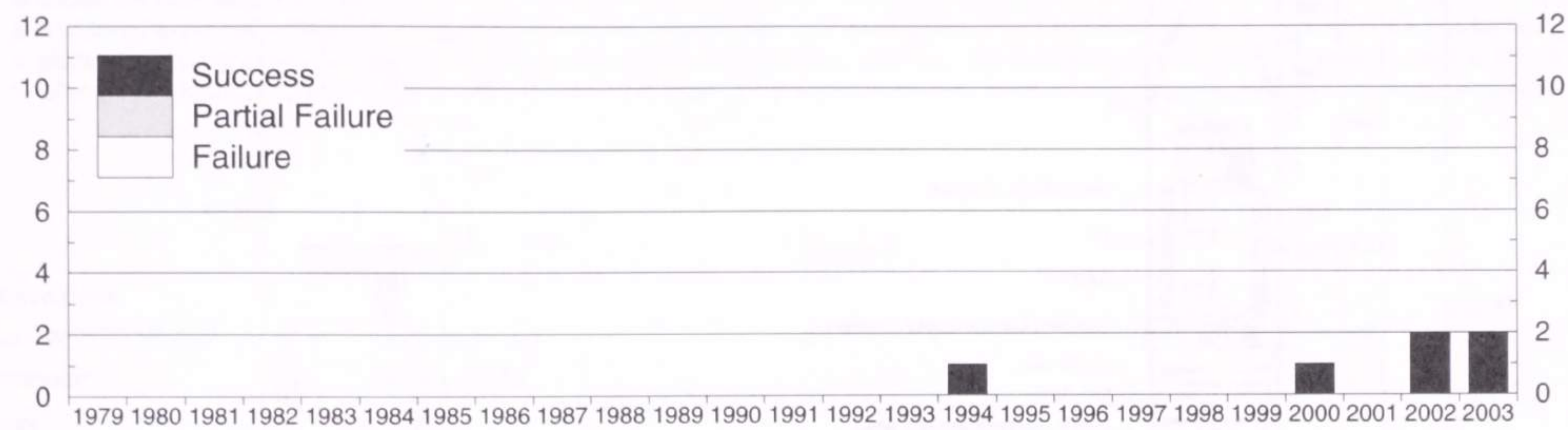


Rockot: Circular Orbit Performance in Specific LEO Inclinations

Flight History

In addition to launches of Rockot, there have been at 83 launches of the RS-18 (UR-100N) through 2001, and 68 launches of its precursor, the UR-100 (SS-11), with a total of three failures.

Total Orbital Flights



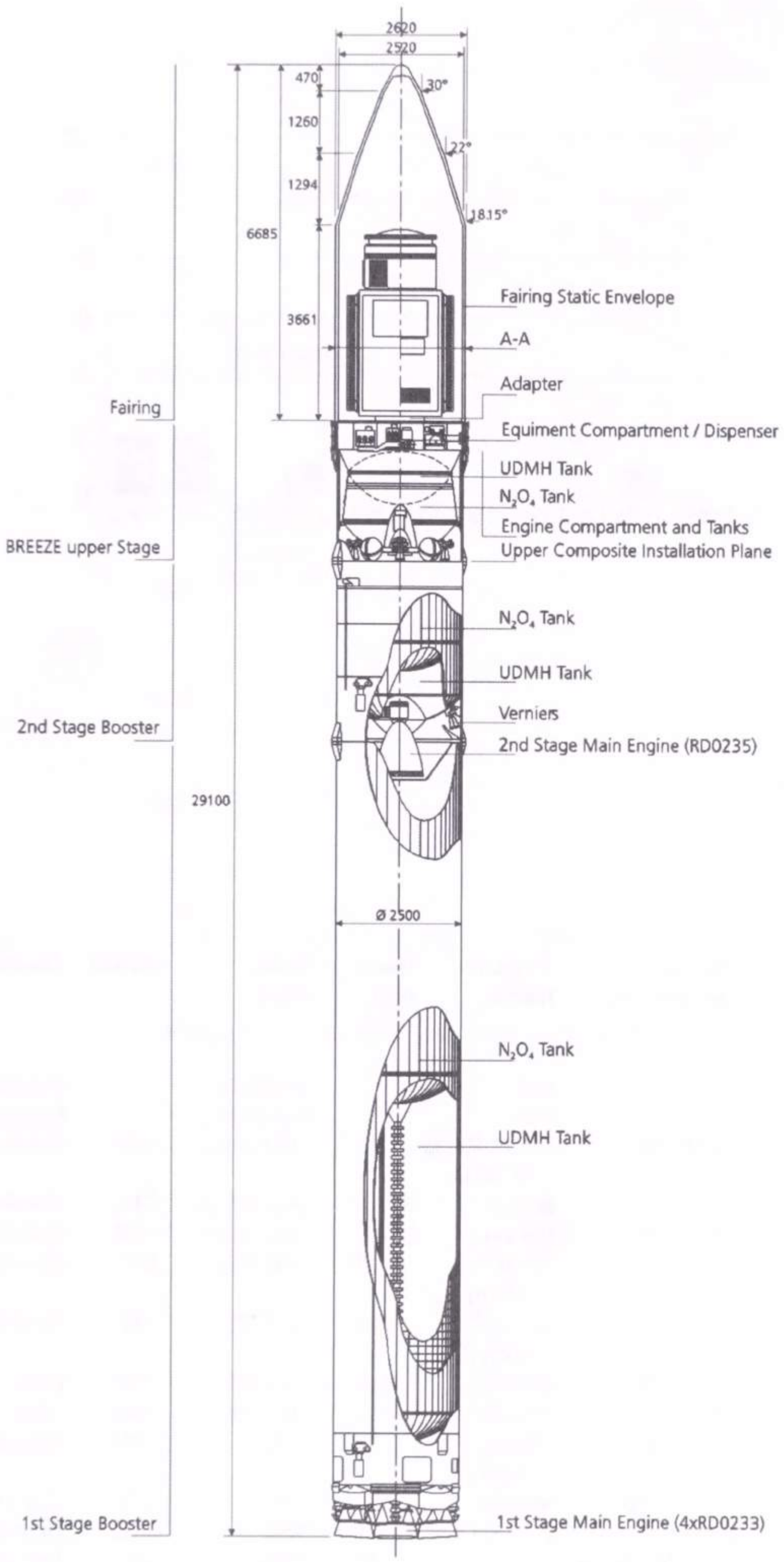
Flight Record (through 31 December 2003)	Rockot
Total Orbital Flights	6
Launch Vehicle Successes	6
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

	Date (UTC)	Launch Interval (days)	Model	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1990 Nov 20	—	Rockot/Breeze K	Baikonur	Silo 131	none		test		suborbital		Russia
T	1991 Dec 20	395	Rockot/Breeze K	Baikonur	Silo 175	none		test		suborbital		Russia
T	1994 Dec 26	1102	Rockot/Breeze K	Baikonur	Silo 175	1994 085A		RS-15 Radio-ROSTO	70	LEO (64.8)	NGO	Russia
T	2000 May 16	1968	Rockot/Breeze-KM	Plesetsk	LC 133	2000 026A		Simsat 1	657.4	LEO (86.3)	CML	Russia
						2000 026B		Simsat 2	660.3	LEO (86.4)	CML	Russia
	2002 May 17	731	Rockot/Breeze-KM	Plesetsk	LC 133	2002 012A		GRACE 1 (Tom)	432	LEO (89)	CIV	Germany
						2002 012B		GRACE 2 (Jerry)	432	LEO (89)	CIV	Germany
	Jun 20	34	Rockot/Breeze-KM	Plesetsk	LC 133	2002 031A		Iridium 97	680	LEO (86.6)	CML	USA
						2002 031B		Iridium 98	680	LEO (86.6)	CML	USA
	2003 Jun 30	375	Rockot/Breeze KM	Plesetsk	LC 133	2003 031A		Monitor E mockup	700	SSO	CML	Russia
						2003 031B	C	Mimosa	51	SSO	CIV	Czech
S						2003 031C		DTUSat 1	1	SSO	NGO	Denmark
						2003 031D	C	MOST	66	SSO	CIV	Canada
						2003 031E	A	CUTE 1	1	SSO	NGO	Japan
						2003 031F	A	Quakesat	3	SSO	CML	USA
						2003 031G	A	AAU Cubesat	1	SSO	NGO	Denmark
						2003 031H	A	Can X-1	1	SSO	NGO	Canada
						2003 031J	A	Cubesat XI-4	1	SSO	NGO	Japan
	Oct 30	122	Rockot/Breeze KM	Plesetsk	LC 133	2003 050A		SERVIS 1	840	SSO	CIV	Japan

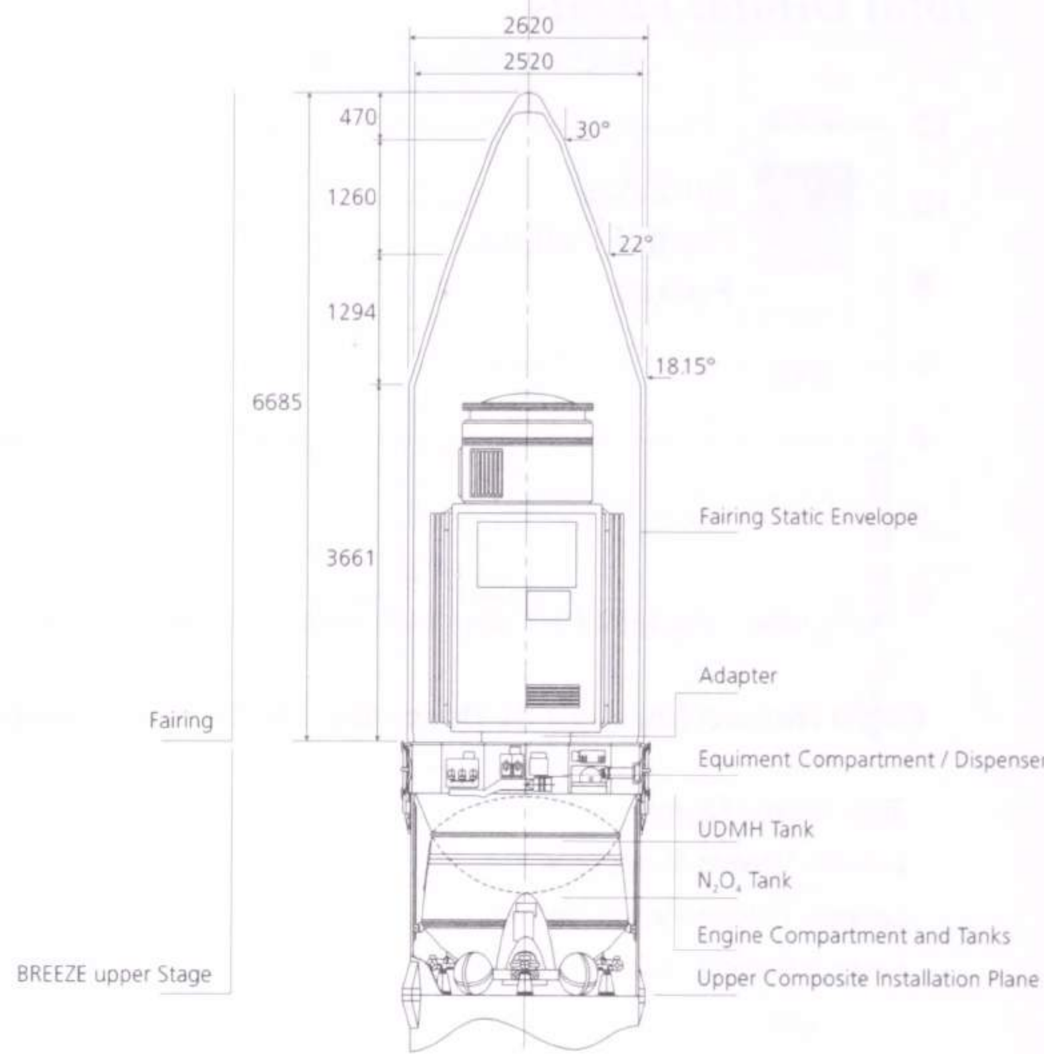
T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly
Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Vehicle Design

Overall Vehicle



Courtesy EUROCKOT Launch Services.



Courtesy EUROCKOT Launch Services.
Rockot Upper Composite with Payload

Courtesy EUROCKOT Launch Services.
Rockot Breeze KM

	Rockot
Height	29.15 m (95.5 ft)
Gross Liftoff Mass	107.5 t (237 klbm)
Thrust at Liftoff	1870 kN (420 klbf)

Vehicle Design

Stages

The first and second stages of Rockot are decommissioned RS-18 ICBM stages, and therefore relatively little information is available about them. Each stage contains tanks for N_2O_4 and UDMH, separated by a common bulkhead.

Rockot uses the Breeze upper stage. The new Breeze KM is a structurally modified version of the original Breeze K stage. They are functionally identical, although the weight of the Breeze K may differ slightly from the Breeze KM weights shown here. The stage includes a toroidal oxidizer tank that encircles the main engine and shares a common bulkhead with the fuel tank. The original conical equipment section of the Breeze K has been flattened and widened to provide more room for the payload, and the structure has been strengthened to provide a better payload environment. The Breeze stage is restartable in orbit, allowing orbital plane changes and deployment of multiple satellites. The main propulsion system is apparently derived from a family of engines used on Soviet military spacecraft. A new upper stage, the Breeze KS, is under development to support launches into highly elliptical orbits.

	Stage 1	Stage 2	Breeze KM
Dimensions			
Length	17.2 m (56.4 ft)	3.9 m (12.8 ft)	1.3 m (4.3 ft)
Diameter	2.5 m (8.2 ft)	2.5 m (8.2 ft)	2.5 m (8.2 ft)
Mass			
Propellant Mass	Roughly 80 t (175 klbm)	Roughly 14 t (30 klbm)	5055 kg (11,144 lbm)
Inert Mass	?	?	1420 kg (3130 lbm)
Gross Mass	?	?	6475 kg (14,275 lbm)
Propellant Mass Fraction	?	?	0.78
Structure			
Type	?	?	?
Material	Aluminum	Aluminum	Aluminum
Propulsion			
Engine Designation	RD-0233 (Khimavtomatiki Design Bureau)	RD-0235 engine with RD-0236 verniers (Khimavtomatiki Design Bureau)	S5.98M main engine (Khimmash Design Bureau) 11D458 verniers
Number of Engines	4	1 main engine + 1 vernier engine with 4 nozzles	1 main engine + 4 vernier engines
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Average Thrust (Total)	Sea level: 1870 kN (420.4 klbf) Vacuum: 2070 kN (465.4 klbf)	Main Engine: 240 kN (54 klbf) Verniers: 15.76 kN (3540 lbf)	Main engine: 19.62 kN (4410 lbf) Verniers: 396 N (89 lbf)
Isp	Sea level: 285 s Vacuum: 310 s	Main engine: 320 s Verniers: 293 s	Main engine: 325.5 s Verniers: 252 s
Chamber Pressure	205 bar (2975 psi)	75 bar (1090 psi)	?
Nozzle Expansion Ratio	?	?	?
Propellant Feed System	Closed-cycle turbopump	Main engine: closed-cycle turbopump Verniers: single shared turbopump	Main engine: closed-cycle turbopump Verniers: pressure-fed
Mixture Ratio (O/F)	?	?	2.0:1
Throttling Capability	Yes, unknown range	100% only	100% only
Restart Capability	None	None	8 total starts
Tank Pressurization	Hot gas generators	Hot gas generators	Nitrogen
Attitude Control			
Pitch, Yaw, Roll	Single axis nozzle gimbal on each engine	Vernier system with single axis gimbal on each nozzle	12 ACS thrusters, 16 N (3.6 lbf) each
Staging			
Nominal Burn Time	121 s	183 s	1000 s maximum
Shutdown Process	Command shutdown	Command shutdown	Command shutdown
Stage Separation	Four retro-rockets. Stage 2 vernier thrusters	Four retro-rockets	?

Vehicle Design

Attitude Control System

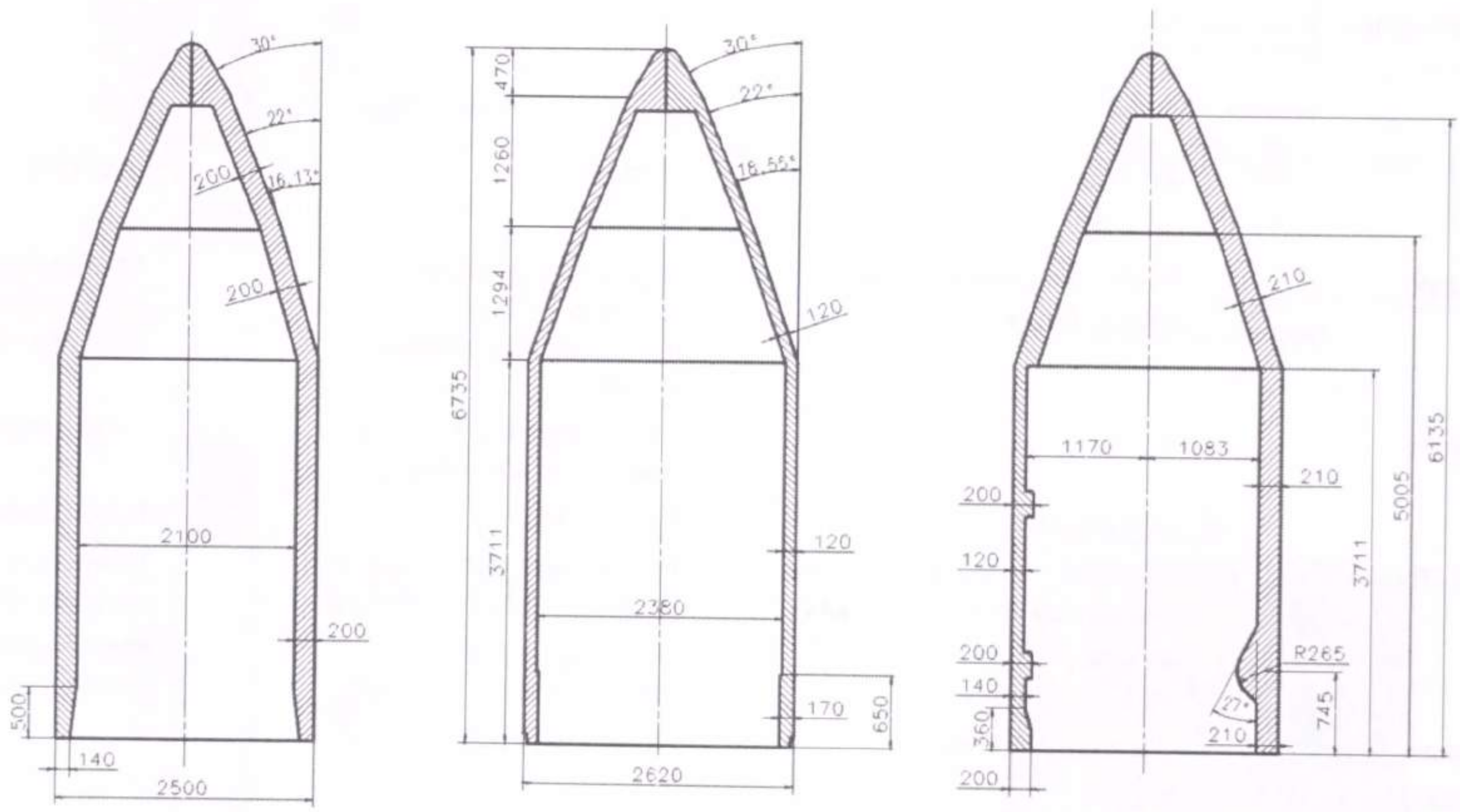
The first-stage attitude is controlled by a single-axis gimbal on each of the four engines. The second-stage attitude is controlled with a single vernier engine that has four nozzles. The Breeze stage has twelve 17D58E ACS thrusters, which control pitch, yaw, and roll.

Avionics

The Rockot avionics are located in an equipment bay on top of the Breeze stage. The guidance, navigation, and control system controls the vehicle during all stages of flight. It includes an inertial guidance system with a three-axis gyro platform, and a flight computer. Control commands are computed in three separate channels, with majority voting to correct errors. The telemetry system includes onboard tape recorders to store in-flight data until it can be transmitted to an available tracking station. Vehicle tracking is facilitated by a beacon. For longer duration missions, three silver-zinc batteries can be integrated into the Breeze stage to provide roughly 7 h of power.

Payload Fairing

The new commercial Rockot fairing replaces an older ICBM-derived fairing used during test flights and has much more room for the payload. The new composite fairing is based on the design of the Proton commercial payload fairing. It is separated by releasing mechanical locks along the vertical split line and firing pyrotechnic bolts along the horizontal split line of the fairing. The two halves are then pushed apart with springs and rotated around hinges at the top of the Breeze stage.

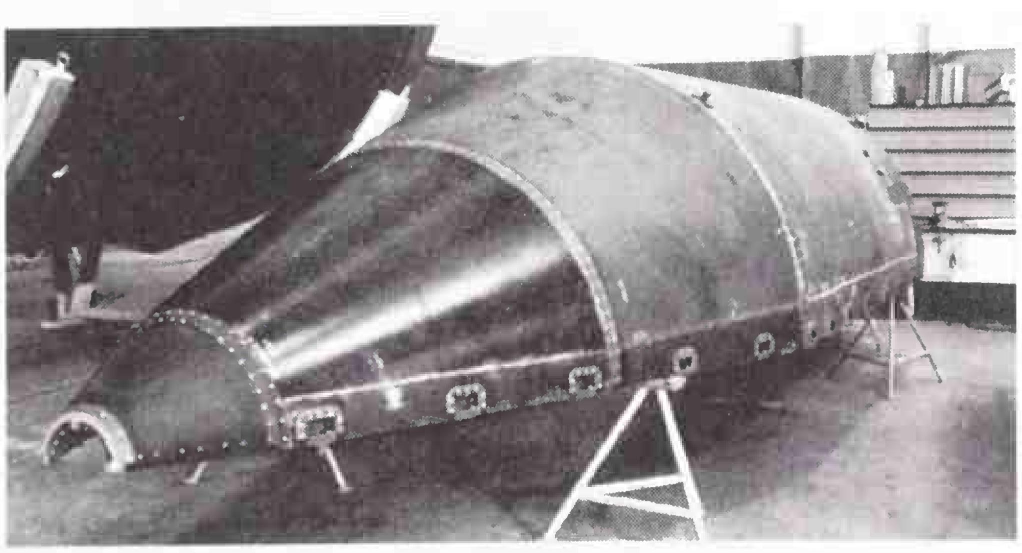


Courtesy EUROCKOT Launch Services.

	Rockot Commercial Fairing
Length	6.74 m (21.1 ft)
Primary Diameter	2.5 m (8.2 ft)
Mass	800 kg (1760 lbm)
Sections	2
Structure	Carbon fiber composite sandwich over aluminum honeycomb core
Material	Carbon fiber, aluminum

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	2380 mm (93.7 in.)
Maximum Cylinder Length	3661 mm (144.2 in.)
Maximum Cone Length	2554 mm (100.5 in.)
Payload Adapter Interface Diameter	Mission specific
Payload Integration	
Nominal Mission Schedule Begins	T–18 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	?
On-Pad Storage Capability	Unlimited
Last Access to Payload	4 days after encapsulation, or 1 h before launch with nonstandard access door
Environment	
Maximum Axial Load	8.45 *g*
Maximum Lateral Load	0.5 *g*
Minimum Lateral/Longitudinal Payload Frequency	15 Hz/33 Hz (or longitudinal can be below 16 Hz)
Maximum Acoustic Level	132 dB at 400–500 Hz
Overall Sound Pressure Level	142 dB
Maximum Flight Shock	6000 *g* at 4000 Hz
Maximum Dynamic Pressure on Fairing	66.4 kPa (1390 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	2 kPa/s (0.3 psi/s)
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	300 km, 82 deg orbit, direct injection: 1% error on altitude, ±0.03 deg inclination 700 km, 63 deg orbit, dual-burn injection: 2% error on altitude, ±0.05 deg inclination
Attitude Accuracy (3 sigma)	±1.5 deg on each axis
Nominal Payload Separation Rate	Depends on payload adapter
Deployment Rotation Rate Available	5 rpm standard for spin-stabilized, 10 rpm maximum
Loiter Duration in Orbit	5 h
Maneuvers (Thermal/Collision Avoidance)	Yes, including deorbit capability
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Rockot is designed to accommodate multiple payloads. Dispenser concepts for two to six spacecraft have been developed.
Auxiliary Payloads	Eurocket offers a "LaunchaPiggy" service for microsatellites. Maximum auxilliary payload mass is 250 kg (551 lbm). Standard dimensions are 300–600 mm (11.8–23.6 in.) in depth and width, and up to 800 mm (31.5 in.) in height.



Courtesy EUROCKOT Launch Services.

Rockot Commercial Payload Fairing Half Shell

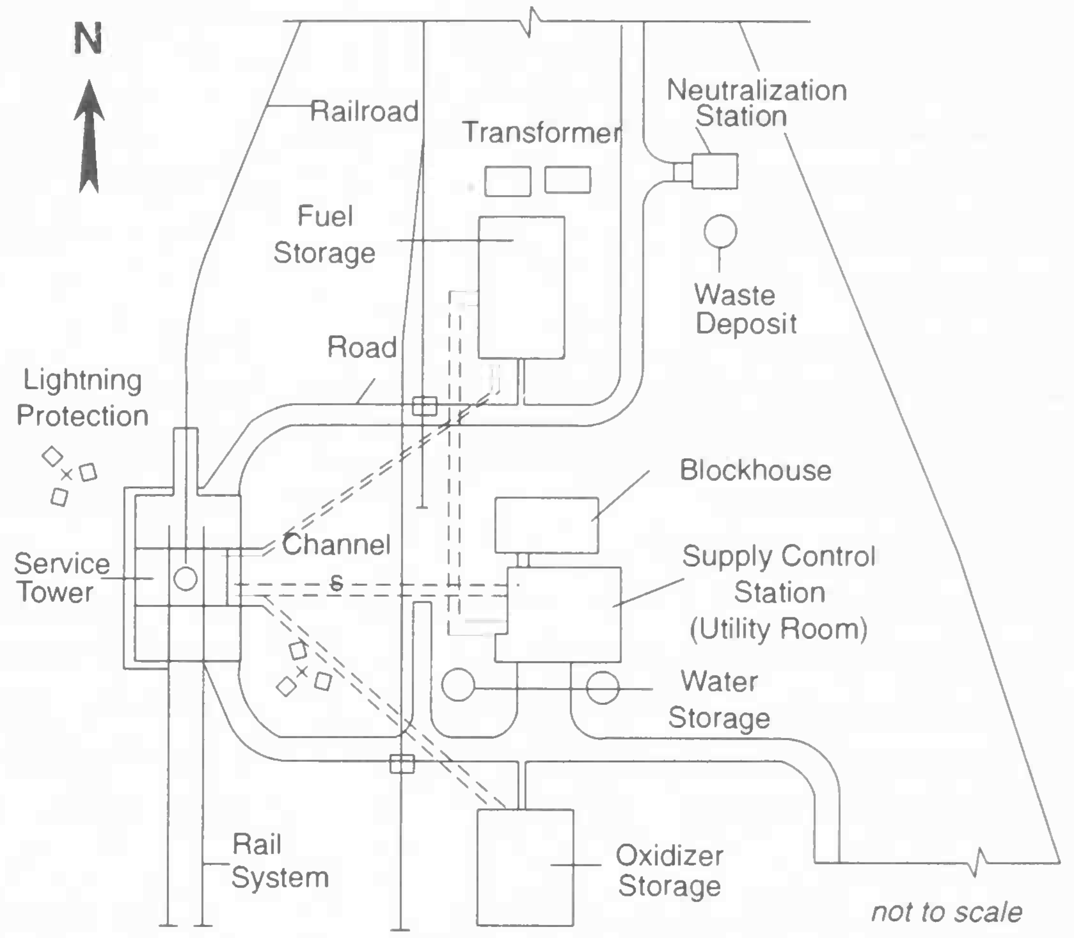
Production and Launch Operations

Production

The first and second stages of Rockot are retired stages from RS-18 (SS-19) ICBMs. The missiles are drained of propellant, removed from their silos, and maintained in climate-controlled storage inside their transport containers in either Khrunichev or Russian military facilities. Reliability is verified with periodic tests, including hot-fire engine tests. The Breeze stage for each Rockot flight is produced in Khrunichev facilities near Moscow. The first- and second-stage engines were produced by the Khimavtomatiki Design Bureau.

Launch Operations

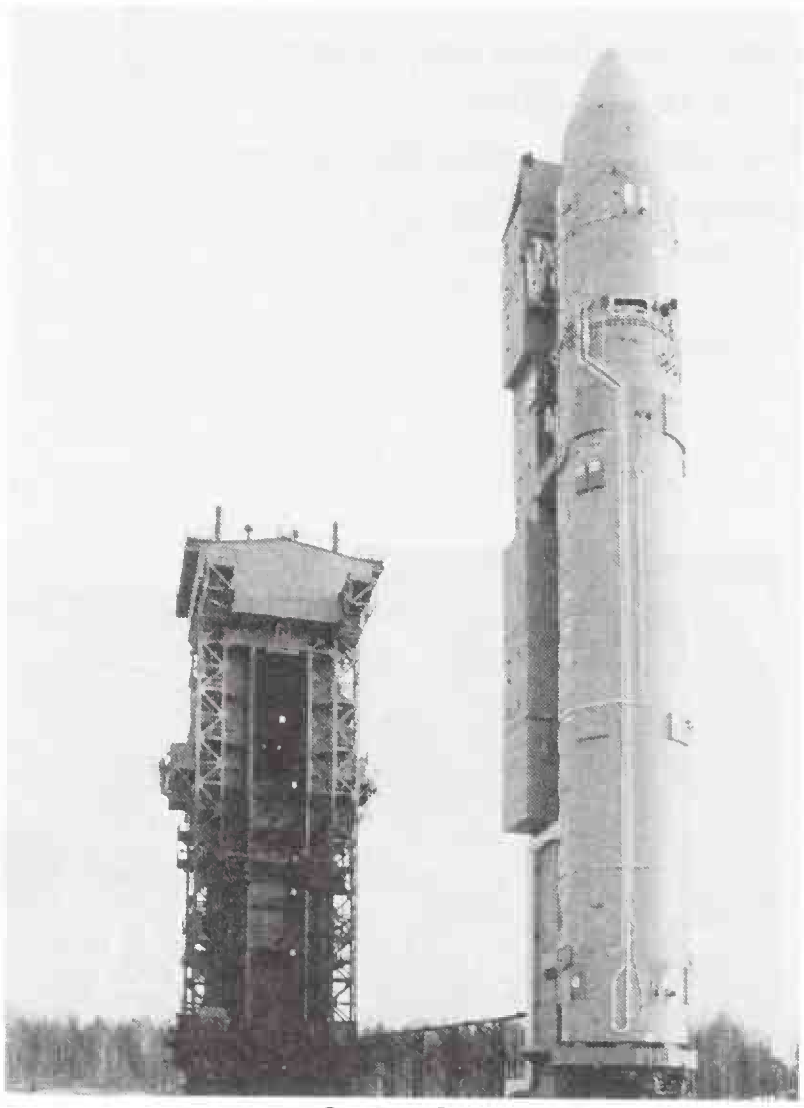
Rockot is launched from the Plesetsk Cosmodrome, approximately 200 km (125 nmi) south of Archangel. The 1750 km^2 (657 mi^2) cosmodrome is Russia's former secret military launch complex. Before the collapse of the Soviet Union, it was the most active space launch site in the world, with pads for Kosmos, Tsiklon, Soyuz, and Molniya boosters. The Plesetsk complex also includes booster and payload test and integration facilities, tracking antennae, a LOX/LN_2 production plant, an airport, railway station, and residential areas.



Courtesy EUROCKOT Launch Services.

Rockot Launch Pad Facilities

Launch operations for Rockot vehicles are conducted by the Russian Strategic Missile Forces. The booster is shipped by rail directly to the "MIK" integration facility. The payload is also processed at the MIK. The payload can be shipped by rail or aircraft. Since the 50×2000 m (160×6550 ft) runway at Plesetsk cannot accommodate large aircraft, payloads shipped by air must be transferred to smaller aircraft (such as the An-12) in Moscow or Archangel, then shipped by road or rail from the Plesetsk airport to the MIK. The MIK includes areas for testing the spacecraft. A separate hazardous processing facility 35 km away is used for fueling the Breeze stage, and can optionally support fueling of bipropellant spacecraft. Approximately two weeks each are required for spacecraft operations and booster preparation, and most tasks can be conducted in parallel. Combined launch vehicle and payload operations take approximately one week. The spacecraft is encapsulated into the fairing and attached to the Breeze stage in the MIK clean room. At T–3 days the booster is delivered to launch pad 11P865PR at Launch Complex 133, a former Kosmos launch facility. Although Rockot has been launched from silos, the acoustic environment is too harsh for most spacecraft, so an above-ground launch pad is now used. LC 133 includes a launch mount, a stationary umbilical mast, a mobile service tower with an overhead crane, and a blockhouse roughly 100 m (300 ft) from the launch mount. The booster is delivered in its transportation container and lifted into vertical position. Two days before launch, the upper composite (Breeze, payload, and fairing) is lifted into place on the booster, and an extension to the launch-transport container is put in place. On launch day, the booster is fueled through interfaces in the launch container. Launch managers monitor the status of the launch vehicle, payload, pad facilities, downrange tracking and communications networks, and the weather before approving a launch. The service tower rolls back 10 min before launch. At liftoff, the Rockot engines ignite, and the vehicle lifts off, guided out of the launch container on rails.

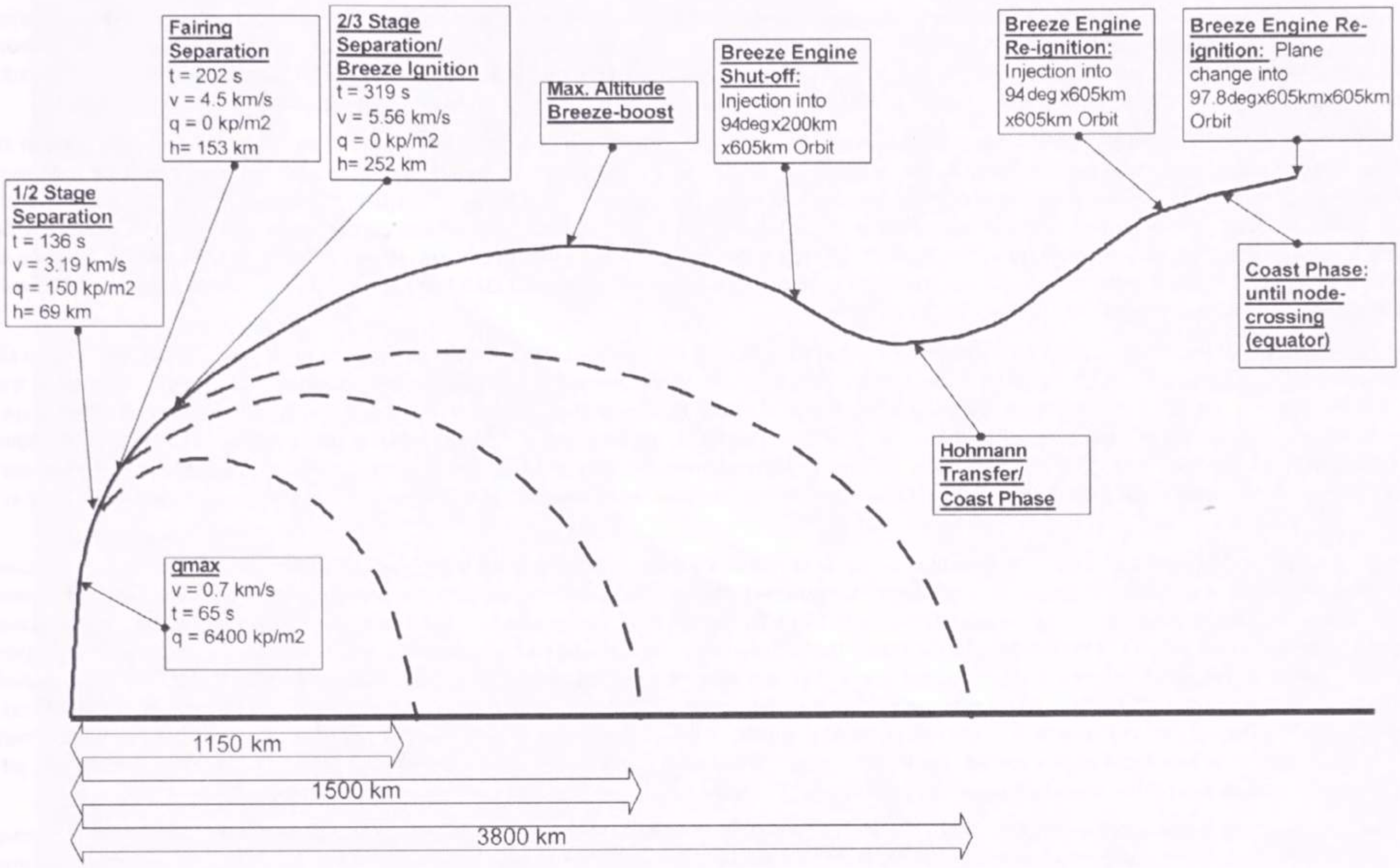


Courtesy EUROCKOT Launch Services.

The Rockot launch complex: Rockot is in its transport/launch container mounted to the fixed umbilican mast. The Mobile Service Tower is in the background.

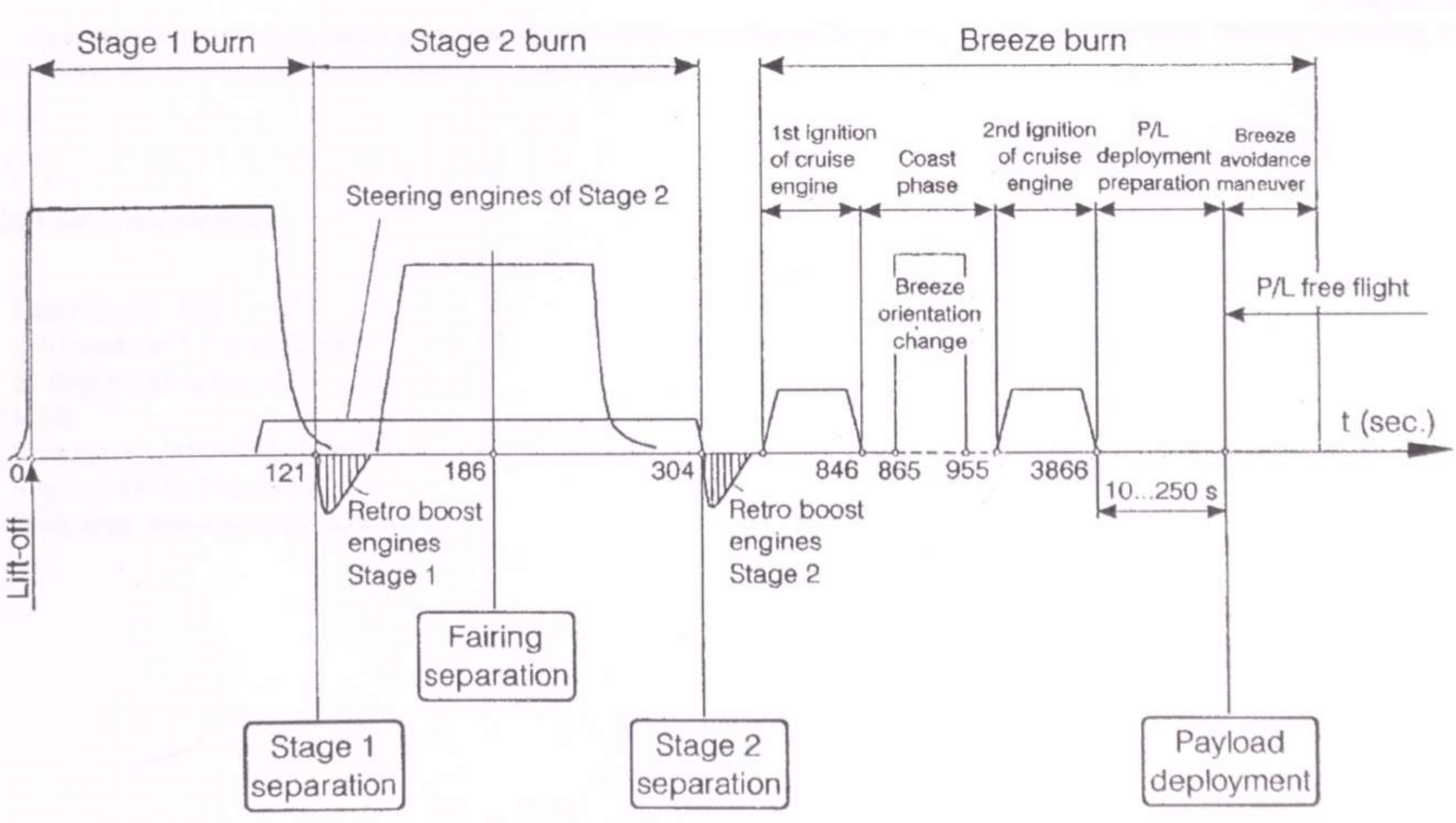
Production and Launch Operations

Flight Sequence



Courtesy EUROCKOT Launch Services.

Typical Rockot SSO Mission Profile



Courtesy EUROCKOT Launch Services.

Rockot Flight Sequence for a 700-km Reference Orbit

Vehicle History

The Rockot and Strela launch vehicles are based on the RS-18 ICBM (also known as UR-100NU in Russia and SS-19 Mod 2 in the West). The RS-18 was developed in 1975–1977 as a replacement for the earlier UR-100 (SS-11) ICBM and was commissioned into service in 1979. It was designed by NPO Mashinostroyeniya and its KB Salyut design bureau, which is now part of Khrunichev. Approximately 360 of the missiles were built through 1991.

In the mid 1980s the Soviet Ministry of Defense funded the development of a space launch vehicle derived from the RS-18, which was given the name Rockot. The Breeze K upper stage was added to the two missile stages to increase the capability of the vehicle for space launch missions. Two suborbital test flights were launched from Baikonur in 1990 and 1991, but funding disappeared after the breakup of the Soviet Union. One more test launch, this time an orbital flight carrying an amateur radio satellite, was performed in 1994 before the launch team was disbanded.

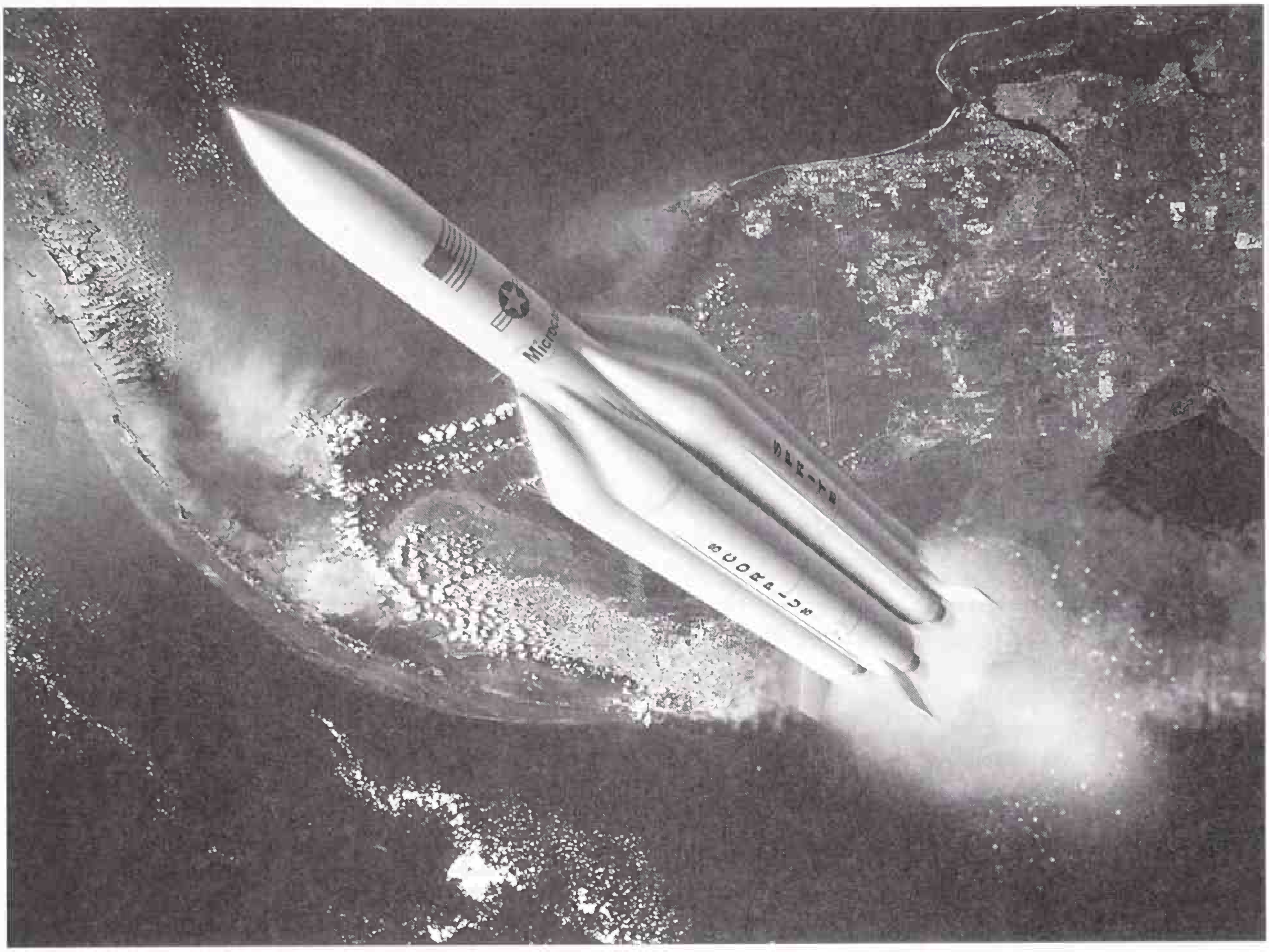
During 1993–1994, Khrunichev began an effort to commercialize Rockot as an alternative to the stalled government program. It acquired a number of missiles and began negotiations with Daimler-Benz Aerospace (DASA) of Germany to be a marketing partner. The two companies formed the joint venture Eurockot in Germany in 1995. As originally conceived, Khrunichev would provide the launch vehicle, refurbish the launch site, and conduct launch operations. DASA would contribute relatively little capital to the project, but in return for marketing the vehicle, profits from sales would be used to write off a portion of Russia's debt to Germany. However, the venture made little progress in its first two years, in part a result of bureaucratic holdups in Russia and withholding of funds from Germany. Eventually the partnership was restructured, and DASA agreed to provide a cash infusion of $35 million primarily to upgrade and refurbish the launch facilities.

Meanwhile, the design of the Breeze K had evolved. Since the future of Rockot was unclear, Khrunichev had focused on a new version, the Breeze M, as an upgrade for Proton, when it appeared that Rockot would be cancelled. When Rockot was revived, it was decided that using a variation of the Breeze M was more attractive for Rockot than restarting Breeze K production. In combination with a new payload fairing, the redesigned stage would provide more room to meet the requirements of spacecraft like Iridium. Therefore, the baseline of Rockot shifted to use the Breeze KM stage. Changes from the Breeze K include rearranging the avionics into a shorter, wider equipment section, which provides more volume for the payload and more surface area for mounting multiple satellites. The Breeze structure and fairing interface were redesigned to reduce the dynamic loads imparted on the payload.

Marketing efforts began in earnest in 1997, and Rockot soon became one of the most promising small launch vehicles. Contracts for more than a dozen firm launches and a similar number of options were signed with companies planning LEO communications constellations such as Iridium, LEO One, and E-SAT. However, the program was struck by several setbacks in 1999. In August 1999, Iridium filed for bankruptcy, presaging similar business failures by Rockot's other customers. A pair of Proton failures increased tensions between Russia and Kazakhstan regarding launches of vehicles with toxic propellants like Rockot. In response, Khrunichev announced that it was canceling the upgrades and repairs planned for Silo Launcher 175, the planned Rockot launch site at Baikonur. This effectively prevented Rockot from launching LEO One at Baikonur as planned. Finally, in December 1999, an accident occurred at the Rockot pad in Plesetsk. Khrunichev planned a launch of the original Rockot configuration, using the Breeze K and small payload fairing, to demonstrate the new facilities at Plesetsk. During ground electrical tests a few weeks before launch, the payload fairing was accidentally jettisoned inside the mobile launch tower and damaged beyond repair. The rocket and the RVSN-40 payload were mothballed.

Rockot eventually recovered from these setbacks. The demonstration flight was replanned using the new Breeze KM and larger payload fairing to serve as the first flight of the commercial configuration of Rockot. The Commercial Demonstration Mission was launched in May 2000, carrying two dummy satellites. The satellites, called Simsats, were spare spacecraft mass simulators that had been used in preparation for the Proton launches of Iridium, and the mission profile mimicked an Iridium deployment mission even though by this time Iridium appeared unlikely to emerge from bankruptcy. Business began to pick up during 2002, when Rockot finally launched its first commercial payload, a pair of GRACE satellites for Germany and NASA. Rockot has become the preferred light launch vehicle for many European government satellites because of its German connections and has also been selected to launch satellites from Asia and Canada.

SCORPIUS



Courtesy Microcosm, Inc.

Scorpius is a family of small expendable launch vehicles designed to be very simple and low cost. The Sprite vehicle, shown above, will be the first configuration to be developed.

Contact Information

Microcosm, Inc.
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El Segundo, CA 90245
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Phone: +1 (310) 726-4100
Fax: +1 (310) 726-4110
Web site: www.scorpius.com

Scorpius

Scorpius

General Description

Summary

The Sprite Mini-Lift Launch Vehicle is the first in the Scorpius family of low-cost launch vehicles. The Scorpius program goal is to reduce the cost of launch by a factor of 5–10 below existing launch systems. Scorpius launchers are also designed for responsive launch operations. Sprite is intended to be capable of a launch within 8 h of arrival of the payload at the integration site. Sprite is a three-stage, pressure-fed, liquid-propellant expendable launch vehicle designed to carry payloads up to 315 kg (700 lbm) to low Earth orbit.

Status

In development. First launch planned in 2006.

National Origin

United States

Key Organizations

Marketing Organizations	Scorpius Space Launch Company
Launch Service Provider	Scorpius Space Launch Company
Prime Contractor(s)	Microcosm, Inc.

Primary Missions

Small spacecraft to LEO

Estimated Launch Price

$2.9 million initially, decreasing to $2.5 million by the tenth flight (Microcosm, 2002)

Spaceports

The following launch sites are planned. Inclinations shown are typical for the facility but vehicle-specific limits have not been determined.

Launch Site:	Vandenberg AFB, Space Systems International pad
Location:	34.7°N, 120.6°W
Available Inclinations:	63°–145°
Launch Site:	Virginia Spaceport Authority, Wallops Flight Facility
Location:	37.9°N, 75.4°W
Available Inclinations:	38°–55°
Launch Site:	Cape Canaveral AFS
Location:	28.5°N, 81.5°W
Available Inclinations:	28.5°–57°

Performance Summary

200 km (108 nmi)	314 kg (700 lbm)
200 km (108 nmi), at 90 deg	209 kg (461 lbm)
Space Station orbit: 407 km (220 nmi), 51.6 deg	250 kg (551 lbm)
Sun-synchronous orbit: 800 km (432 nmi), 98.6 deg	125 kg (276 lbm)
GTO	No capability
Geostationary orbit	No capability

Flight Record (through 31 December 2003)

No orbital flights to date.

Flight Rate

Unknown.

Nomenclature

Sprite vehicle nomenclature has not yet been established, though it is likely to be similar to the system used to date on suborbital rockets. Scorpius suborbital vehicles are numbered with a prefix of SR for suborbital rocket followed by a dash and a letter code, designating the vehicle design. The flight number follows this series of letters: an X added before the vehicle design designator indicates a test vehicle or test flight. For example, the SR-XM-1 vehicle is a test flight of a suborbital M vehicle design, first flight.

Cost

The first two flights of the Sprite vehicle are planned as test launches and are funded as a part of the development program. Microcosm price estimates for vehicles 3 through 21 in FY02 dollars are as follows:

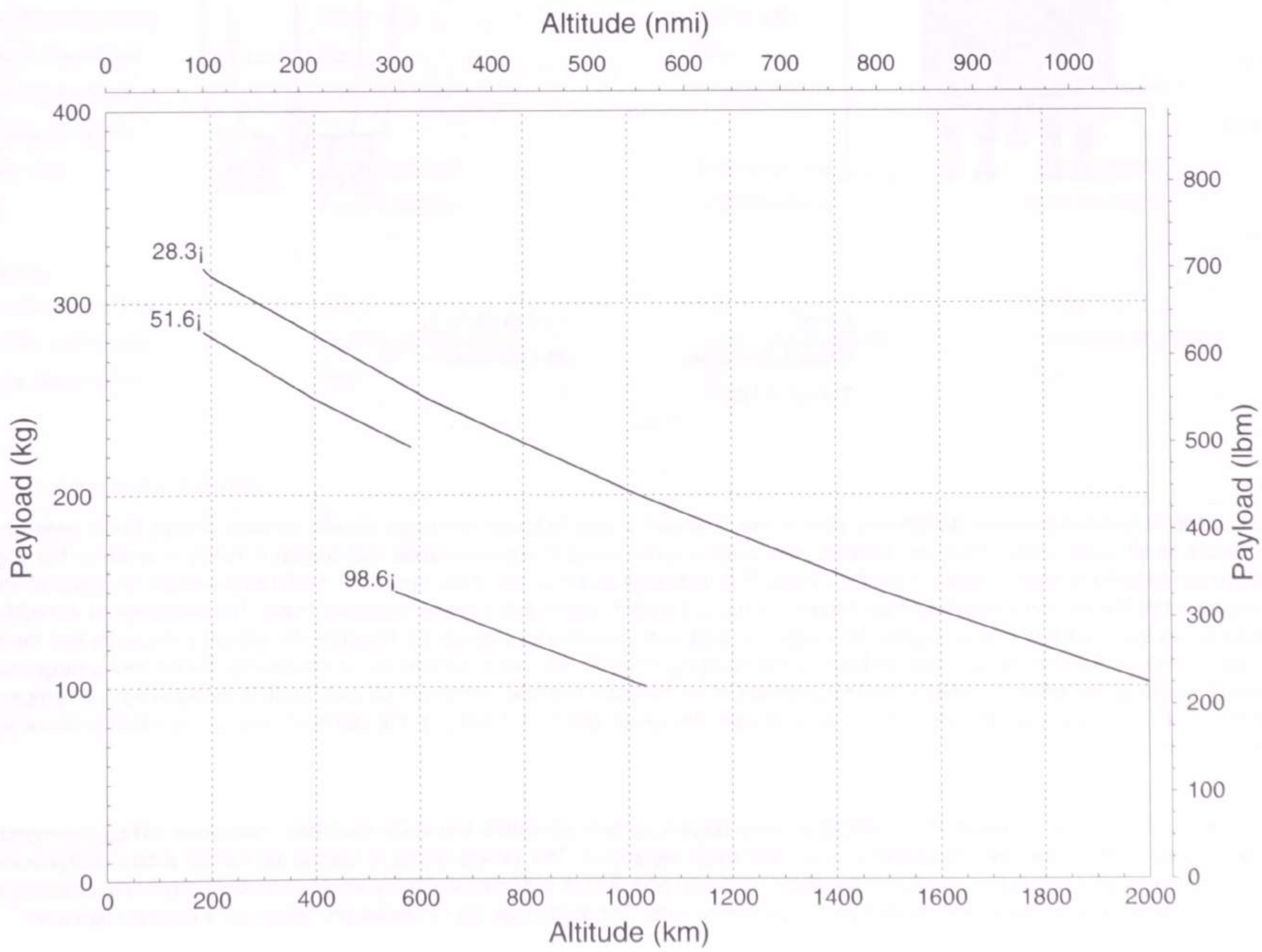
	Vehicle price (thousands)	Est. range costs (thousands)	Total price (thousands)
Vehicles 3–5	$2,492	$399	$2,892
Vehicles 6–12	$2,170	$347	$2,517
Vehicles 13–21	$2,044	$327	$2,371

Development of Scorpius has been funded primarily by the U.S. government through the U.S. Air Force, often as a congressional line item. Additional funding has come from NASA and Microcosm internal funds. About $15 million in government funding was provided by various contracts through 1997 including $7 million from the DoD in the 1997 budget. Sen. Pete Domenici (a Republican from New Mexico, where Scorpius was being tested), added $5 million for Scorpius to the fiscal year 1999 defense budget. Current estimates of the cost of development are not publicly available, but in 1997 Microcosm planned a budget of $28 million to develop Sprite, including three test launches.

Availability

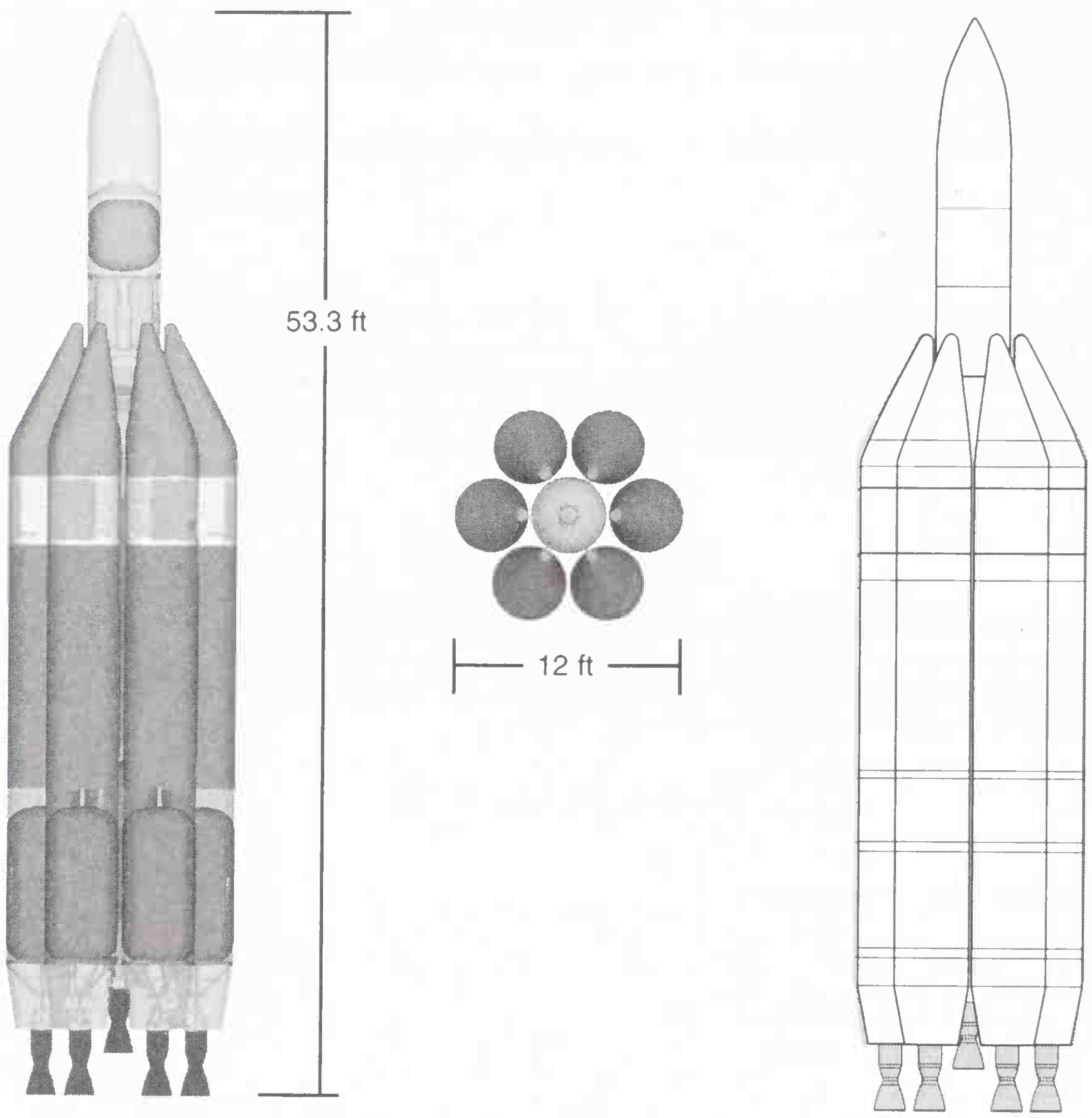
The Sprite vehicle is not currently operational and is not yet being actively marketed. Two suborbital tests of the Scorpius technology have been flown successfully from White Sands Missile Range in 1999 and 2001. Additional suborbital tests and two orbital flight tests are planned before beginning commercial operations. The first Sprite vehicle test flights are planned for 2006 depending on availability of funding. When testing is complete, Sprite launches will be marketed and conducted by Scorpius Space Launch Company (SSLC).

Performance



Vehicle Design

Overall Vehicle



Length	16.2 m (53.3 ft)
Gross Liftoff Mass	38 t (84 klbm)
Thrust at Liftoff	?

Stages

The Scorpius family of launch vehicles is designed to provide very low-cost access to space by using a simple, modular design. Sprite uses seven common "pods" and a small upper stage. Pods are identical units used in constructing Scorpius vehicles and consist of tanks, a pressure-fed engine, and electronics connected to the primary avionics in the third stage. The first stage is composed of six outer pods each using a single Scorpius 90 kN (20,000 lbf) thrust engine, while the second stage is a single center pod using a similar engine with a higher expansion ratio. The third stage is a modified smaller pod using a single 9 kN (2000 lbf) thrust engine. All stages use LOX and kerosene as propellants. Because the vehicle is pressure-fed, the tanks are robust enough to support themselves and can endure routine handling. The shorter, wider vehicle profile caused by the parallel arrangement of the stages makes it stable while vertical, enabling easier movement of an integrated vehicle to the launch pad. Sprite's empty weight is comparable to a small bulldozer, and it can be easily towed by a standard truck cab. All normal vehicle servicing on the pad is done at ground level so there is no need for a gantry or tower.

All six pods of the first stage are ignited at liftoff, and the second stage is ignited just before first-stage shutdown. Shutdown will be commanded for all six first-stage engines at the same time immediately before first stage separation. The vehicle fairing is ejected just before second-stage burnout. The Sprite third stage is a smaller and lighter version of the standard Sprite pod, and is designed to deploy single or multiple payloads, including associated attitude control, collision avoidance, and thermal protection maneuvers. The third stage also includes provisions for a deorbit maneuver.

Vehicle Design

	Stage 1	Stage 2	Stage 3
Dimensions			
Length	11.1 m (36.5 ft)	11.1 (36.5 ft)	?
Diameter	1.1 m (3.5 ft) per pod, 3.56 m (11.7 ft) total	1.1 m (3.5 ft)	1.1 m (3.5 ft)
Mass			
Propellant Mass	26,386 kg (58,050 lb)*	4408 kg (9698 lb)	1142 kg (2512 lb)
Inert Mass	4661 kg (10,254 lb)*	841 kg (1851 lb)	263 kg (578 lb)
Gross Mass	31,047 kg (68,304 lb)*	5250 kg (11,549 lb)	1723 kg (3790 lb)
Propellant Mass Fraction	0.85*	0.84	0.81
Structure			
Type	Monocoque	Monocoque	Monocoque
Material	Aluminum–composite	Aluminum–composite	Aluminum–composite
Propulsion			
Engine Designation	Microcosm 20K engine	Microcosm 20K engine	Microcosm 2.5 K engine
Number of Engines	6	1	1
Propellant	LOX/kerosene (Jet-A)	LOX/kerosene (Jet-A)	LOX/kerosene (Jet-A)
Average Thrust (vacuum)	90.7 kN (20.4 klbf) each	100.9 kN (22.7 klbf)	10.2 kN (2.3 klbf)
Isp (Vacuum)	277.5 s	309 s	323 s
Chamber Pressure	26.2 bar (385 psi)	26.2 bar (385 psi)	10.5 bar (154 psi)
Nozzle Expansion Ratio	6.56:1	30:1	80:1
Propellant Feed System	Pressure-fed	Pressure-fed	Pressure-fed
Mixture Ratio (O/F)	2:4	2:4	2:4
Throttling Capability	100% only	100% only	100% only
Restart Capability	No	Yes	TBD
Tank Pressurization	Heated helium	Heated helium	Heated helium
Attitude Control			
Pitch, Yaw	Engine gimbal	Engine gimbal	Engine gimbal
Roll	Engine gimbal	ACS thrusters	ACS thrusters
Staging			
Nominal Burn Time	122 s	122 s	Varies
Shutdown Process	Command shutdown	Command shutdown	Command shutdown
Stage Separation	TBD	TBD	TBD

* combined masses of 6 pods

Vehicle Design

Attitude Control System

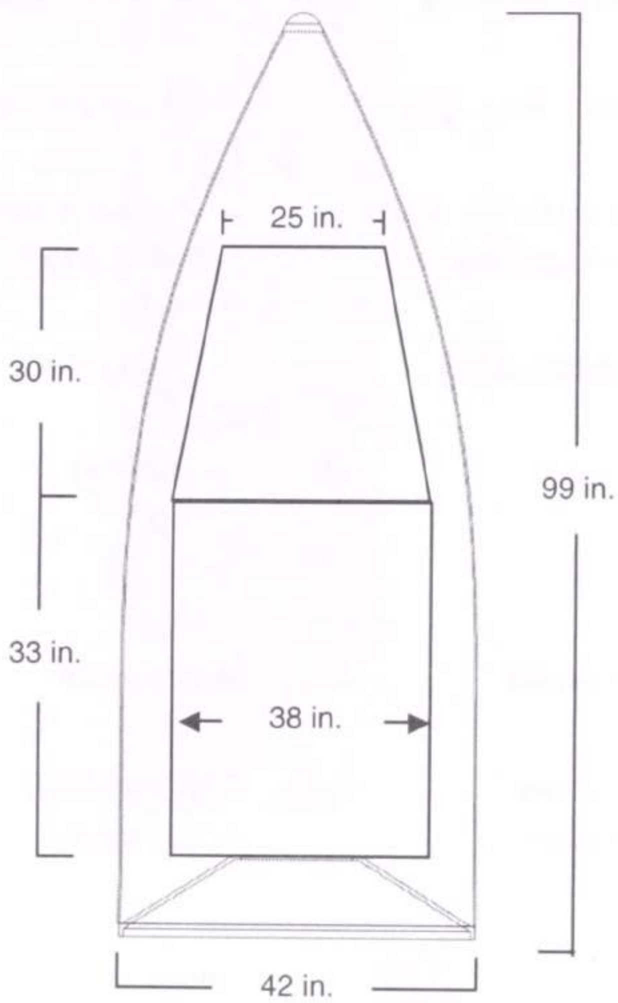
During the first-stage burn, attitude control is achieved through thrust vector control of the six hinged engines. The system provides simultaneous roll, pitch, and yaw control for the first stage. The second and third stage control pitch and yaw with a similar 2-axis TVC system. The roll axis is controlled with a thruster system connected to the pressurization system. Following orbit insertion, the Sprite Upper Stage performs a series of maneuvers to provide the desired attitude for payload separation. Any inertial attitude, preprogrammed into the mission timeline, can be achieved.

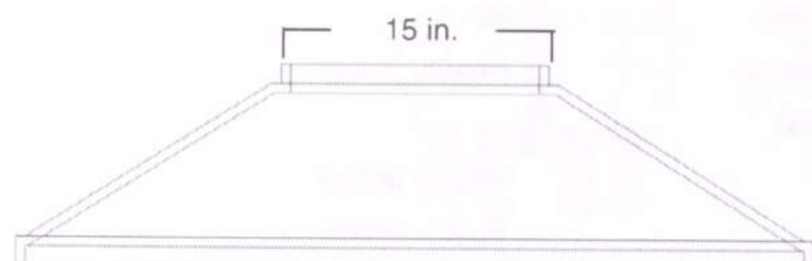
Avionics

A central flight computer, connected to remote control units on each pod via a RS-485 bus, controls the entire vehicle. An integrated INS/GPS system provides flight input data to the flight computer. Output control commands are send via the appropriate remote control units to the TVC interface. An independent telemetry system transmits relevant flight data and optional video on an S-band frequency to the ground.

Payload Fairing

Sprite is designed to have a similar payload envelope to the retired Scout G-1 launch vehicle.





Length	2.5 m (8.25 ft)
Primary Diameter	1.06 m (3.5 ft)
Mass	?
Sections	2?
Structure	?

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	970 mm (38 in.)
Maximum Cylinder Length	840 mm (33 in.)
Maximum Cone Length	760 mm (30 in.)
Payload Adapter Interface Diameter	380 mm (15 in.)
Payload Delivery	
Loiter Duration in Orbit	2 h from launch
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Marketing plans for comanifest payloads are unknown. Contact Microcosm for more information
Auxiliary Payloads	Scorpius plans to make use of extra capacity to fly auxiliary payloads at little or no cost to support space technology development

Production and Launch Operations

Production

Unlike most other launch systems, Microcosm intends to locate most Scorpius personnel and facilities near the primary launch site, which has not been selected at this time. The Scorpius program will require three main facilities, a production and administrative facility, an integration facility, and a launch site. The prime function of the production facility will be to manufacture components for Sprite and assemble the vehicle pods. The integration facility will house the integration of the vehicle pods, upper stage, and payload into a vehicle ready for launch. These two facilities will be located as close to the launch site as feasible. Sprite pods and third stages will be produced and tested as complete units in the production and administrative facility. Major pod segments, such as the tanks and engine bay, will be assembled individually in a designated area and then integrated into a pod on the production line. The production line system will use carefully aligned rails to guide segment and component handling fixtures. This facilitates easy and accurate alignment of the pod as production proceeds. Another bay in the production and administrative facility will house composites manufacture, such as filament winding of tanks and engines and composites lay-up areas for other components. This facility will also include an avionics manufacturing bay, including thermal and thermal-vacuum chambers, and a machine shop and fabrication area.

Launch Facilities

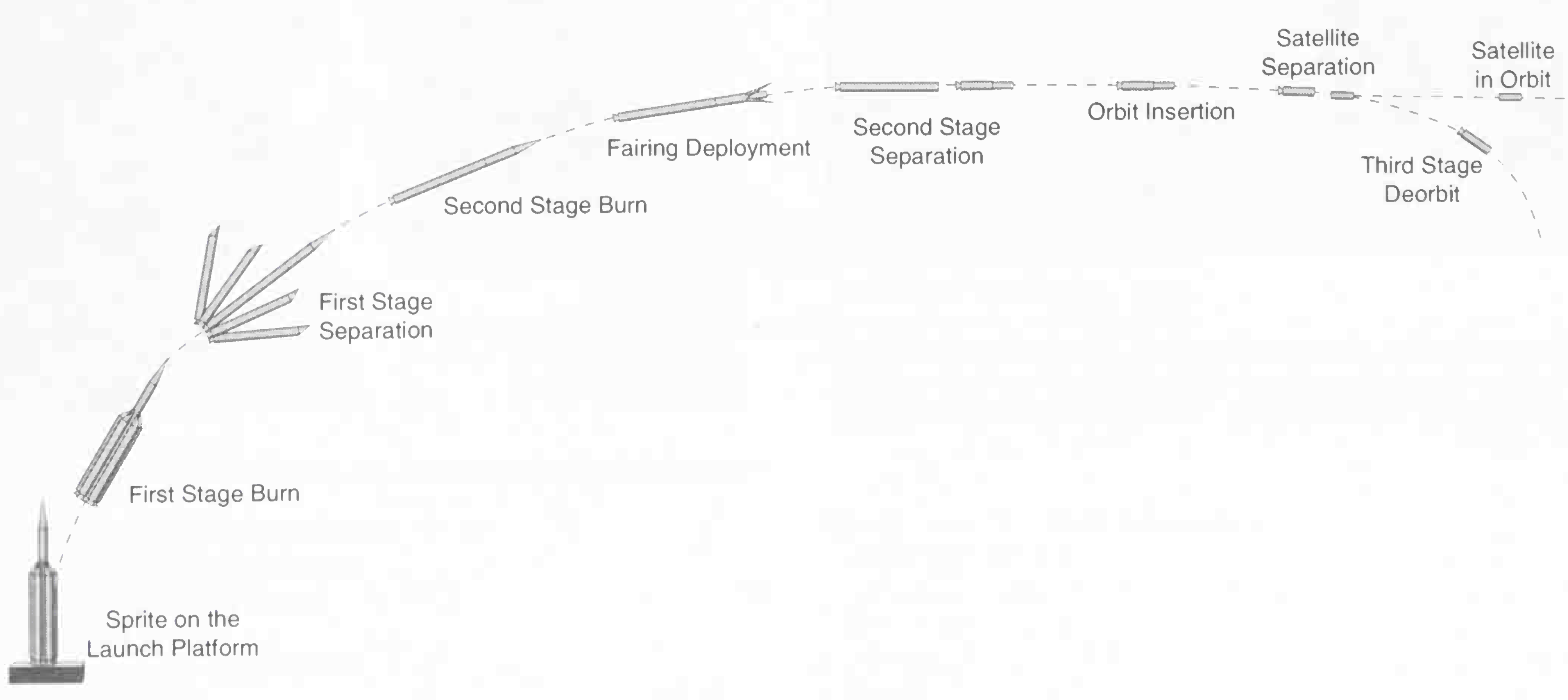
The Sprite vehicle launch operations begin with the integration of pods into a vehicle. Upon arrival at the integration facility, the pods will be installed onto a mobile launch platform to make a completed Sprite vehicle. The launch platform is hardened to also act as the launch stand during launch. The completed vehicle is then checked out and moved to a vertical storage area in the integration facility until a payload is ready for integration. To support very rapid response launch operations, completed Sprite vehicles may be placed in storage so that there is always a vehicle ready when a new payload requires a launch on short notice. The process of integrating vehicles will be designed to support launches every 8 h during surge periods.

When a payload is ready for integration with the launch vehicle, a vehicle is moved to one of the payload integration bays. The payload is weighed and balanced, integrated with the separation mechanism, and then lifted onto the Sprite. While the payload is being integrated with the separation mechanism, the launch vehicle begins a series of automated checkout tests. These and other activities will be conducted in parallel whenever feasible to decrease launch preparation time. The payload–separation system unit is bolted onto the third stage. Electrical connections are made between the launch vehicle and the payload, and a short post-integration checkout is performed. During this operation, the fairing is prepared for installation. Fairing installation begins immediately upon completion of post-integration checkout. Once fairing installation is finished, the vehicle on its launch platform begins rollout to the pad. At the pad, the platform rolls over the flame trench and is pinned in place to restrict horizontal and vertical movement. The umbilical, ignition system, and hold down systems are connected and checked, and the pad is evacuated. Further operations are conducted remotely from a sheltered command area. These operations include fueling, final health checks, and launch countdown operations.

At the pad, propellants and pressurants are loaded. Most launch activities are automated, with supervision by the launch personnel. Final prelaunch checks are completed during the fueling process, and final clearances obtained from the launch range and the FAA. The last two minutes of the prelaunch sequence are fully automated. As long as the vehicle remains on the pad, the first- and second-stage LOX and fuel lines will be connected to the vehicle, enabling removal of the propellants in the event of launch scrub.

The Sprite vehicle is designed to be robust to winds aloft, minimizing weather delays. Vehicle interfaces are designed for ground-level servicing, eliminating the need for a gantry. The vehicle design also makes launch operations from sites other than the primary launch site practical. Only a basic concrete pad with a launch stool and thrust deflector is required. The entire vehicle can be shipped in two standard trailers, and the low weight of the pods makes them easy to handle.

Flight Sequence



Vehicle Upgrade Plans

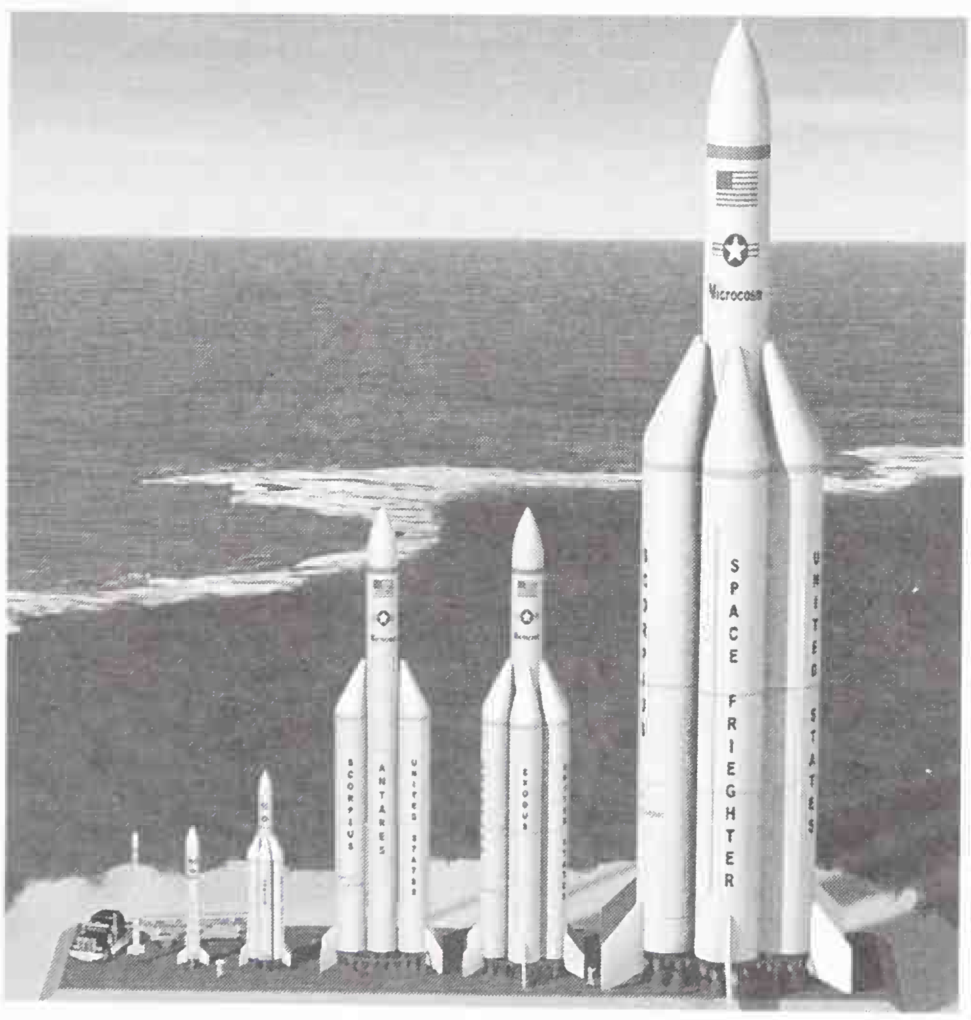
The Sprite is intended to be the first member of a broader Scorpius family. The Scorpius system is based on designs that are modular and scalable to remove as much risk as possible for larger vehicle systems by testing the technologies on the smaller vehicles. In general, the larger Scorpius vehicles will also follow an operations flow similar to the smaller Sprite vehicle.

Vehicle History

The Scorpius program began in 1993 under an Small Business Innovative Research (SBIR) contract from the Ballistic Missile Defense Organization, which was later transferred to Phillips Laboratory. Scorpius was publicly unveiled in 1995 when Microcosm reported the results of test firing of its 22 kN (5000 lbf) engine. At that time, the program was focused on building the "Liberty" vehicle that would carry 1000 kg (2200 lbm) for $1.7 million. A series of incrementally more capable sounding rockets were planned to develop and test the technology for the orbit-capable Liberty.

The first test launch of a single-engine SR-S sounding rocket failed in October 1998 when a LOX valve failed to open and the aft section of the rocket caught fire. The vehicle was refurbished and flew successfully in January 1999. In July 1999 Microcosm began testing a larger 90 kN (20 klbf) thrust engine that is planned for use on the Sprite launch vehicle. A larger twin-engine sounding rocket, the SR-XM flew in March 2001. This vehicle used a single stage of the same diameter planned for the Sprite stages.

Although demand for small launch systems has waned, military interest in Scorpius and launch systems like it has been increasing. As the U.S. Air Force completed development of the much larger Atlas V and Delta IV under its EELV program, it began to shift its focus to the benefits of a complementary, operationally responsive launch vehicle that could launch a variety of lightweight payloads on short notice. Much of the future funding for Scorpius is expected to come from military sources.



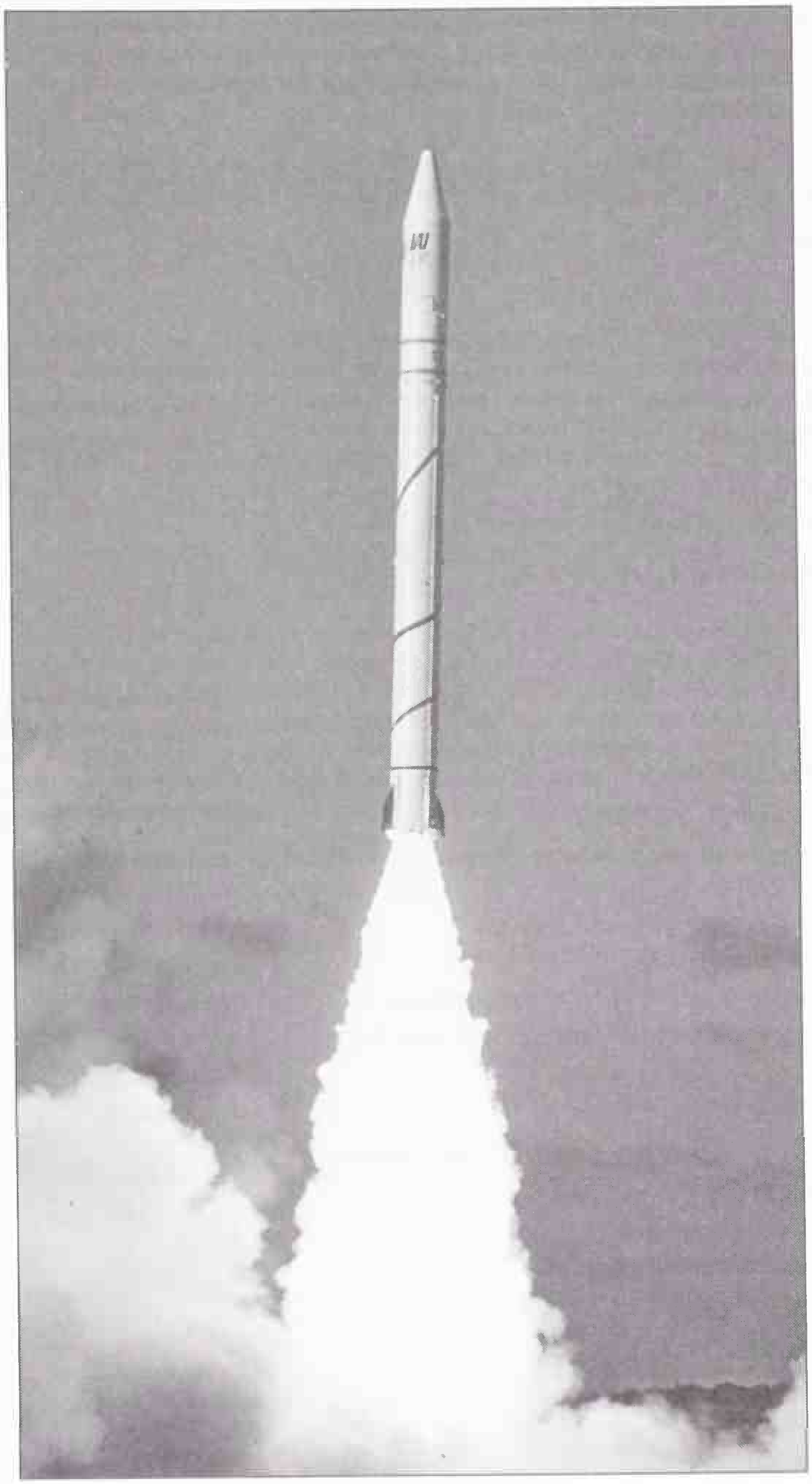
Courtesy Microcosm, Inc.

Scorpius Family Vehicles: From left the SR-S and SR-M sounding rockets, Sprite orbital launch vehicle, Antares with 3000 kg (6500 lbm) payload, Exodus 6800 kg (15,000 lbm) capability, and Space Freighter.

Courtesy Microcosm, Inc.

The SR-XM-1 flight test vehicle flew in 2001.

SHAVIT AND LEOLINK



Courtesy Israel Aircraft Industries

The Shavit is Israel's first space launch system, built to orbit the Offeq series of satellites.

Contact Information

International Business Development:
Israel Aircraft Industries
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MLM Division
P.O. Box 45, Beer Yaakov
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E-mail: mlm marketing@mlm.iai.co.il
Web site: iai.co.il

Government Program Information:
Israel Space Agency
23 Arania Street
P.O. Box 7182
Tel Aviv 61070
Israel
Phone: 972-3-260266

Shavit 1

Shavit 1 or LK-A

General Description

Summary

The Shavit series vehicles are very small launch vehicles built in Israel to launch the Offeq (Horizon) series observation satellites. Israel became the eighth nation with a space launch capability when it launched the first Shavit in September 1988. It is widely believed that the Shavit is derived from Israel's YA-3 Jericho 2 ballistic missile. The Shavit 1 is a three-stage solid-propellant vehicle with a longer first stage than the original Shavit. The LK-A, a configuration similar to the Shavit 1, is now active.

Status

Operational. First Shavit launch in 1988. First Shavit 1 launch in 1995. First LK-A launch in 2002.

Origin

Israel

Key Organizations

Marketing Organization	Israel Aircraft Industries (IAI)
Launch Service Provider	Israel Space Agency
Prime Contractor	Israel Aircraft Industries (IAI)

Primary Missions

Small government satellites in retrograde LEO

Estimated Launch Price

Unknown

Spaceport

Launch Site	Palmachim AFB, Israel
Location	31.9° N, 34.7° E
Available Inclinations	143 deg

Performance Summary

Shavit is launched only to 143-deg retrograde orbits from its present launch site.

240 × 600 km (130 × 323 nmi), 143 deg	350 kg (770 lbm)
200 km (108 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	No capability
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	5
Launch Vehicle Successes	4
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	1

Flight Rate

0–1 per year

NOMENCLATURE

Shavit means Comet in Hebrew. The first two vehicles were named simply Shavit. The second generation vehicles were designated Shavit 1. A new enhanced version of Shavit 1 is designated LK-A, as the first in a line of derivatives marketed under the name LeoLink. Despite the different designations, it is common to refer to any of these vehicles as Shavit without indicating the specific type because they are all quite similar.

COST

Development costs for the Shavit were paid by the government of Israel but the total cost is unknown. No data are available regarding the cost of each launch.

AVAILABILITY

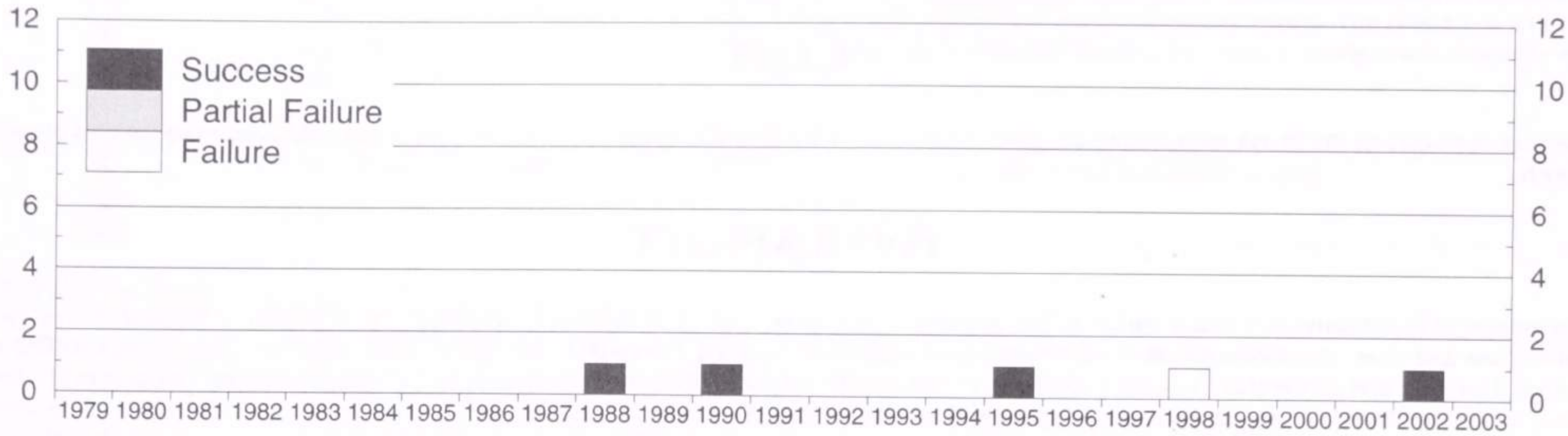
The version of Shavit currently in use is the LK-A. This appears to be a minor upgrade of the Shavit 1 vehicle in use previously. In the past IAI has marketed these and other Shavit variants, most recently in a partnership with Coleman Research in the United States and Matra Marconi (now EADS) in France. However, that partnership has ended. Officially, IAI still markets its launch services commercially in the LeoLink program, however in practice these marketing efforts are minimal and most information about the vehicles is restricted. The Shavit family is not a significant competitor in the small launch vehicle market. IAI has an agreement to operate Shavit launches from the Brazilian launch site at Alcantara in the future, which will make the vehicle more attractive to other customers.

PERFORMANCE

From the Palmachim launch facility operated by the Israeli Air Force, Shavit must launch west over the Mediterranean Sea into a retrograde orbit. This allows dropped stages to land in the Mediterranean, rather than on Israel's neighbors. In addition to traditional safety constraints, Israel is concerned that neighboring, unfriendly states could learn about missile technology from any stages that might land in their territories. The standard inclination used for Shavit launches is 143 deg. Shavit can reach only low elliptical orbits, because it does not have an upper stage that can circularize the orbit at higher altitudes. Maximum performance of the LK-A is 350 kg (770 lbm) to a 240×600 km (130×323 nmi), 143-deg orbit. The first LK-A flight carried about 300 kg (660 lbm) to a 262×774 km (141×418 nmi) orbit. The first Shavit 1 flight carried 225 kg (495 lbm) to a slightly different 366×695 km (197×375 nmi) orbit. These are likely to be representative of the range of achievable apogees and perigees.

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	**Shavit 1 and LK-A**	**Combined Shavit Family**
Total Orbital Flights	3	5
Launch Vehicle Successes	2	4
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	1	1

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1	1988 Sep 19	—	Shavit	1	PALB	1988 087A	Ofeq 1	155	LEO (142.9)	MIL	Israel
	2	1990 Apr 03	561	Shavit	2	PALB	1990 027A	Ofeq 2	160	LEO (143.2)	MIL	Israel
	3	1995 Apr 05	1828	Shavit 1	3	PALB	1995 018A	Ofeq 3	225	LEO (143.4)	MIL	Israel
F	**4**	**1998 Jan 22**	**1023**	**Shavit 1**	**4**	**PALB**	1998 F01A	Ofeq 4			MIL	Israel
	5	2002 May 28	1587	LK-A	5	PALB	2002 025A	Ofeq 5	300	LEO (143.5)	MIL	Israel

PALB is Palmachim Air Force Base, Israel

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

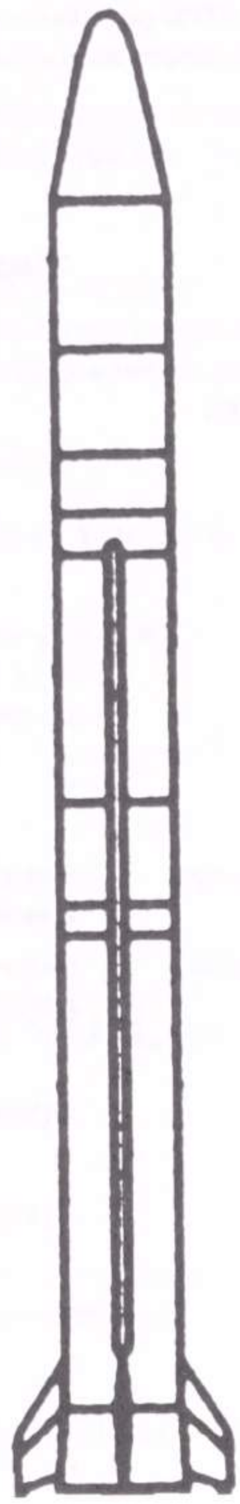
Payload Types: D = Dual Manifest; M = Multiple Manifest; A = Auxiliary Payload

Failure Description:

F	1998 Jan 22	4	1998 F01	Failure occurred during second-stage burn.

Vehicle Design

Overall Vehicle



	Shavit 1 or LK-A
Height	18 m (59 ft)
Gross Liftoff Mass	30 t (66 klbm)
Thrust at Liftoff	?

Vehicle Design

Shavit 1

Because Shavit 1 is closely derived from a military weapons system, Israel has not released many technical details on the vehicle design. All three stages use solid motors of the same diameter. In the first two Shavits, the first- and second-stage motors were essentially identical, with differences in the expansion ratio of the nozzles (the second stage is optimized for high-altitude flight as opposed to sea level operation), and the steering mechanisms. In the Shavit 1 vehicles the first stage is stretched for higher propellant load. The motor is upgraded from 9 t (20 klbm) to 13 t (29 lbm) of propellant. Both lower stages have composite cases with conventional HTPB propellants and EPDM insulation. The third stage is a spherical, spin-stabilized motor built by Rafael. The LK-A configuration has some modest performance enhancements compared to the Shavit 1, however it is unclear what they are. It does not appear that the motors are significantly larger.

	Stage 1	Stage 2	Stage 3
Dimensions			
Length	6.5 m (21.3 ft) ? (May be original Shavit first stage length)	5.3 m (17.4 ft)	2.6 m (8.5 ft)
Diameter	1.35 m (4.4 ft)	1.35 m (4.4 ft)	1.35 m (4.4 ft)
Mass			
Propellant Mass (each)	12,750 kg (28,100 lbm)	?	1900 kg (4200 lbm)
Inert Mass (each)	?	?	?
Gross Mass (each)	Roughly 13 t (29 klbm)	9.1 t (20 klbm) (may be propellant mass)	Roughly 2 t (4400 lbm)
Propellant Mass Fraction	?	?	?
Structure			
Type	Motor: Filament-wound monocoque Interstages: skin-stiffener	Motor: Filament-wound monocoque Interstages: skin-stiffener	Monocoque
Material	Motor: Graphite–epoxy composite Interstages: aluminum	Motor: Graphite–epoxy composite Interstages: aluminum	Titanium
Propulsion			
Motor Designation	? (Israel Military Industries)	? (Israel Military Industries)	AUS 51 (Rafael)
Number of Motors	1	1	1
Propellant	HTPB	HTPB	HTPB
Number of Segments	1	1	1
Average Thrust (each)	?	?	?
Isp	?	?	?
Chamber Pressure	?	?	?
Nozzle Expansion Ratio	?	?	?
Attitude Control			
Pitch, Yaw	Jet vanes, aerofins	LITVC	Spin-stabilized
Roll	Jet vanes, aerofins	?	Spin-stabilized
Staging			
Nominal Burn Time	?	?	92 s?
Shutdown Process	Burn to depletion?	Burn to depletion?	Burn to depletion?
Stage Separation	External pyrotechnic V-clamp	External pyrotechnic V-clamp, hot separation	

Vehicle Design

Attitude Control System

At liftoff the Shavit is controlled by jet vanes that are angled into the motor exhaust stream to exert pitch and yaw control moments. Once up to speed, the vehicle is controlled by three aerodynamic fins at the base of the vehicle. The jet vane system is jettisoned once the aerofins are functioning. The second stage is controlled by a liquid-injection thrust-vector control (LITVC) system. The third stage is spin stabilized.

Avionics

Shavit is controlled by a distributed, redundant avionics system. Guidance is provided by an inertial measurement system mounted at the top of the second stage. Commands are implemented through a MIL-STD-1553B data bus. The third stage is spin stabilized and unguided.

Payload Fairing

The Shavit payload fairing is a single-piece unit. To separate, the launch vehicle turns out-of-plane during a coast period, separates the fairing around its circumference, and jettisons the fairing forward as a single unit. The vehicle then turns back into the direction of flight for ignition of the next stage.

	Shavit 1 or LK-A
Length	?
Primary Diameter	1.56 m (5.11 ft)
Mass	?
Sections	1
Structure	?
Material	?

Production and Launch Operations

Production

Shavit and LK-A

Shavit is developed by state-owned Israel Aircraft Industries (IAI) at the MLM division in Beer Yaakov, Israel. Israel Military Industries (IMI) produces the first- and second-stage motors. The AUS-51 third stage is produced by Rafael.

Launch Operations—Palmachim Air Force Base, Israel

Shavit and LK-A

Shavit launches are performed from the Palmachim AFB in the Negev desert in southern Israel. In order to avoid overflight of unfriendly neighboring countries, Shavit flights are launched west over the Mediterranean Sea, exiting out over the Atlantic near the Straits of Gibraltar. This retrograde launch azimuth results in an orbit inclination of 143 deg, and imposes severe performance penalties. Launch procedures for the Shavit have not been reported. Some details of the launch operation can be inferred from available information. Shavit appears to be launched from a mobile transporter/erector/launcher (TEL). Marketing information for previously proposed derivatives of Shavit indicated that the launch vehicle would be assembled horizontally and tested separately from the payload, while the payload would be encapsulated in the fairing separately and mated to the assembled launch vehicle. It is likely that a similar approach is used for current Shavit launches. One interesting feature is apparent in photos of the 2002 launch. There is a trisector clamshell structure mounted atop the launch tower that appears to go around the payload fairing. This may provide environmental conditioning for the payload before launch.

Launches of observation satellites for Israel's military will likely remain on Israeli soil. However, for civil and commercial missions, Israel would like to operate Shavit from a foreign launch site that does not have Palmahim's azimuth restrictions. Several sites in the United States were considered in the mid-1990s. Because U.S. law was unclear on the legality of operating a foreign launch system at U.S. launch facilities, particularly one believed to have been derived from a military missile, representatives of IAI had difficulty even touring U.S. launch complexes. Israel has negotiated an agreement with Brazil that will allow Shavit launches from Alcantara in the future. No date has been set for a first launch.

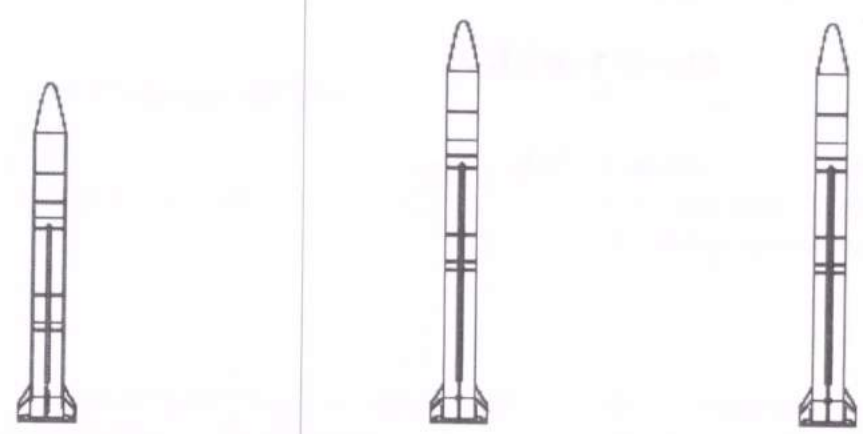
Vehicle Upgrade Plans

IAI plans company-funded development of future versions of the LeoLink line of Shavit vehicles. The LK-1 will replace the 9-t second stage with a 13-t motor like the first stage, and add a liquid-fueled orbit injection stage, which will allow delivery to higher circular orbits. This will result in a capability to deliver 350 kg (770 lbm) to a 700-km (380-nmi) polar orbit. A later configuration, the LK-2, would replace the first stage with a much larger motor. When IAI was partnered with Coleman Aerospace and considering U.S.-built motors, this stage was to have been a Castor 120. It is not clear if that is still the plan. The LK-2 would be able to carry at least 800 kg (1760 lbm) to orbit.

Another Shavit derivative in development is the Air Launched Vehicle (ALV). It would resemble the LK-1 without the first stage and would be air launched from a C-130 cargo aircraft.

Vehicle History

Vehicle Evolution



	Retired	Operational	
Vehicle	**Shavit**	**Shavit 1**	**LK-A**
Period of Service	1988–1990	1995–Present	2002–present
Payload	160 kg (340 lbm)	225 kg (495 lbm)	≤ 350 kg (770 lbm)

Vehicle Description

- **Shavit** Small solid launch vehicle widely believed to be based on Israel's Jericho 2 ballistic missile. Three solid stages, 1.5 m (5 ft) in diameter. First two stages using 9-t (20-klbm) motors.
- **Shavit 1** Similar to Shavit with first stage upgraded to a 13-t (29-klbm) motor.
- **LK-A** Enhanced version of Shavit 1.

Vehicle History

Historical Summary

Israel's desire for an independent space launch capability resulted from its experience with U.S. satellite intelligence data during the 1973 Yom Kippur War. Israel had had limited access to satellite-based information before the war. However, this was not sufficient to predict the surprise attack (though there were other serious Israeli intelligence failures as well), and the United States did not provide any satellite imagery during the conflict. From this experience, Israeli leaders concluded that the United States could not be relied upon and that an independent capability was needed. The first studies of a reconnaissance satellite and small launch vehicle were conducted in 1973 and concluded that such a program was technically feasible. However, the scope of the project was more than Israel could afford. In 1975 and again in 1981, Israel requested that the United States provide as part of its military aid either a complete satellite reconnaissance system or direct access to an existing American satellite already in orbit. The requests were refused, and Israel continued to rely on limited information provided by the United States. Formation of a satellite program was debated in the Israeli government in the early 1980s, and in November 1982 the formation of the Israel Space Agency was announced.

Israel's Shavit space launch vehicle is widely believed to be directly derived from the country's Jericho 2 ballistic missile. The Jericho 2 was developed in cooperation with South Africa in the 1970s and 1980s and was first tested in 1986. As a result of the codevelopment, both nations built very similar missile systems and both planned to use them to launch satellites. South Africa's RSA-3 satellite launch vehicle, which would have been almost identical to Shavit, was cancelled in the early 1990s without performing a launch attempt.

Israel performed its first space launch in 1988, carrying the Offeq 1 experimental satellite. Later satellite launches, performed in 1990 and 1995, have carried Offeq satellites with Earth observation capabilities, which are used by Israel's military forces. Almost from the beginning, Israel has been interested in commercializing the Shavit. Increasing the launch rate would allow overhead costs to be spread over more launches. Also, Israel's Palmachim launch site in the Negev desert is poorly located for space launches. Use of foreign launch sites, such as those in the United States, would greatly increase the capability of the Shavit. The first attempt at selling Shavit abroad occurred when IAI teamed with Delta Research Inc. of the United States in 1990 to bid for NASA's COMET (Commercial Experiment Transporter) launch services contract. It was not selected, and the contract instead went to EER Systems for its Conestoga launch vehicle, which is no longer active. In 1994, Space Vector Corporation teamed with Atlantic Research Corporation, which holds the license for export of Shavit motors to bid an Americanized Shavit for NASA's Ultra-Lite launch service procurement. IAI was not involved in this project. Again, the vehicle was not selected. Finally, in 1998, IAI teamed with Coleman Research Corporation of the United States and Matra Marconi Space of France to form the LeoLink consortium. It bid the LeoLink derivatives of Shavit for NASA's SELVS-2 (Small Expendable Launch Vehicle Services) contract. NASA selected both LeoLink and Orbital Sciences Corporation's Pegasus and Taurus vehicles. The contract did not award any immediate launches, but made LeoLink a qualified supplier for NASA launch services, so that individual spacecraft programs could select LeoLink for launches. In the end no LeoLink launches were ordered, either by NASA or for the expected boom in commercial small satellites. As a result, Coleman and Matra Marconi ended their participation in LeoLink. One additional reason for Matra's change of heart may have been that European industrial consolidation made it part of EADS, which already had more successful small launch small vehicle programs like Rockot. Plans for LeoLink have been delayed, but IAI still eventually intends to develop these vehicles eventually independently. The end of the partnership with Coleman means Shavit will not be launched from CCAFS. Instead, IAI is working with Brazil to perform commercial launches from Alcantara.

One potential domestic competitor to the Shavit family has been terminated. Rafael, another Israeli state-owned aerospace company, was given permission in early 2001 to study a microsatellite launch vehicle derived from its Black Sparrow missile. The missile would have been launched by an F-15 fighter and was intended to carry up to 100 kg (220 lbm) to orbit. In 2002 the government decided that Israel could not afford to support two competing space companies, and the project was shelved.

SHTIL AND VOLNA

Courtesy The Planetary Society.

The Shtil and the Volna are submarine-launched ballistic missiles capable of deploying small satellites into low orbits. Above, a Volna rocket is loaded into the submarine Borisoglebsk *before the launch of the suborbital Cosmos-1 solar sail experiment for the Planetary Society.*

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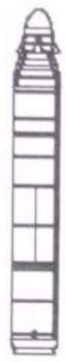
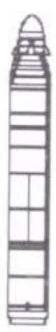
Shtil



Shtil 1

Shtil 2

Volna



Volna

General Description

Summary

The Shtil is one of Russia's most modern submarine-launched ballistic missiles (SLBMs), designated the R-29RM (RSM-54, NATO designation SS-N-23). It is a three-stage, liquid-fueled vehicle that is capable of deploying small spacecraft to LEO. The Shtil 1 uses the standard small warhead compartment to hold the spacecraft. The Shtil 2 has a larger separating payload fairing that can better accommodate spacecraft. The Volna is a smaller relative of the Shtil, using the older R-29R missile. It is used primarily for suborbital launches, but can deliver small payloads to orbit using a solid upper stage.

Status

Shtil 1: Operational. First launch in 1998.

Shtil 2: In development. First launch TBD.

Shtil 2R, 2M: In development. First launch planned in 2004.

Volna: First launch in 2004.

Origin

Russia

Key Organizations

Marketing Organization	State Rocket Center Makeyev Design Bureau.
Launch Service Provider	Russian Navy
Prime Contractor	State Rocket Center Makeyev Design Bureau

Primary Missions

Small satellites to LEO

Estimated Launch Price

Shtil 1: $1.4–2.1 million (Makeyev, 2002)

Shtil 2: $3–4.5 million (Makeyev, 2002)

Volna: $1–1.5 million (Makeyev, 2002)

Spaceports

Launch Site	Shtil 1: Delfin-class missile submarine Volna: Kalmar-class missile submarine
Location	Mobile. Typically launched from the Barents Sea, 69.3°N, 35.3° E
Available Inclinations	79 deg standard, 0–99 deg available
Launch Site	Ground Stand HC-37, Nenoksa State Central Marine Test Site
Location	64.6° N, 39.2° E
Available Inclinations	77–88 deg

Performance Summary

	Shtil 1:	Shtil 2R:	Volna:
200 km (108 nmi); 78.9 deg	140 kg (309 lbm)	220 kg (485 lbm)	?
200 km (108 nmi); 90 deg	?	220 kg (485 lbm)	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	?	No capability	140 kg (308 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi) 98.6 deg	No capability	200 kg (440 lbm)	40 kg (88 lbs)
GTO	No capability	No capability	No capability
Geostationary Orbit	No capability	No capability	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	1
Launch Vehicle Successes	1
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–1 per year

Nomenclature

Volna, meaning "wave," is the name for the space launch version of the R-29R submarine-launched ballistic missile, also designated RSM-50. It is called the SS-N-18 Stingray by NATO. The Shtil, or "calm," family of launch vehicles are based on the R-29RM, a larger variant of the R-29R. This missile is designated RSM-54, and called SS-N-23 Skiff by NATO. The Shtil 1 launch vehicle, also referred to simply as Shtil, has no significant modifications from the original missile. The Shtil 2 designation refers to vehicles with a new payload fairing added to the basic missile. Two further variants, the Shtil 2R and Shtil 2M have added upper stages to the Shtil 2 configuration.

Volna launches are performed from Kalmar-class submarines, designated Project 667BDR in Russia, and code-named Delta III class by NATO. Shtil 1 launches are conducted from Delfin-class submarines. This class is also designated Project 667BDRM and called Delta IV class by NATO.

Cost

The price of launch services on Shtil or Volna vehicles is extremely low, because existing missile assets are available for conversion to launch vehicles with little effort, and the Russian Navy must perform launches occasionally to maintain operational proficiency. In 2002 Makeyev reported that prices of Shtil and Volna launch services are based on a rate of $10,000–15,000 per kg of payload. This would put the cost of a fully loaded Volna with a 100 kg payload at $1–1.5 million, a Shtil 1 with two 70 kg spacecraft at $1.4–2.1 million, and a Shtil 2 at $3–4.5 million. Unlike many other launch providers, Makeyev offers dedicated launches of smaller payloads at the same price per kilogram, rather than at a fixed total price for the launch vehicle. This makes Shtil and Volna the least expensive launch vehicles by far for dedicated launches of small satellites. For example, in 1998 the Technical University of Berlin launched two TUBSAT nanosatellites with a combined mass of 11kg. According to Russian Navy officials, the price of the launch was only 200,000–250,000 DM ($120,000-150,000).

Availability

Shtil and Volna launch services are available commercially through the State Rocket Center Makeyev Design Bureau. Shtil 1 is operational, having performed its first orbital launch in 1998. Volna began performing suborbital launches in the 1990s and is scheduled to perform its first orbital launch in late 2004. The Shtil 2 should be available beginning in late 2003 and be able to perform up to seven commercial launches per year. The Shtil 2M version should be available beginning in 2004.

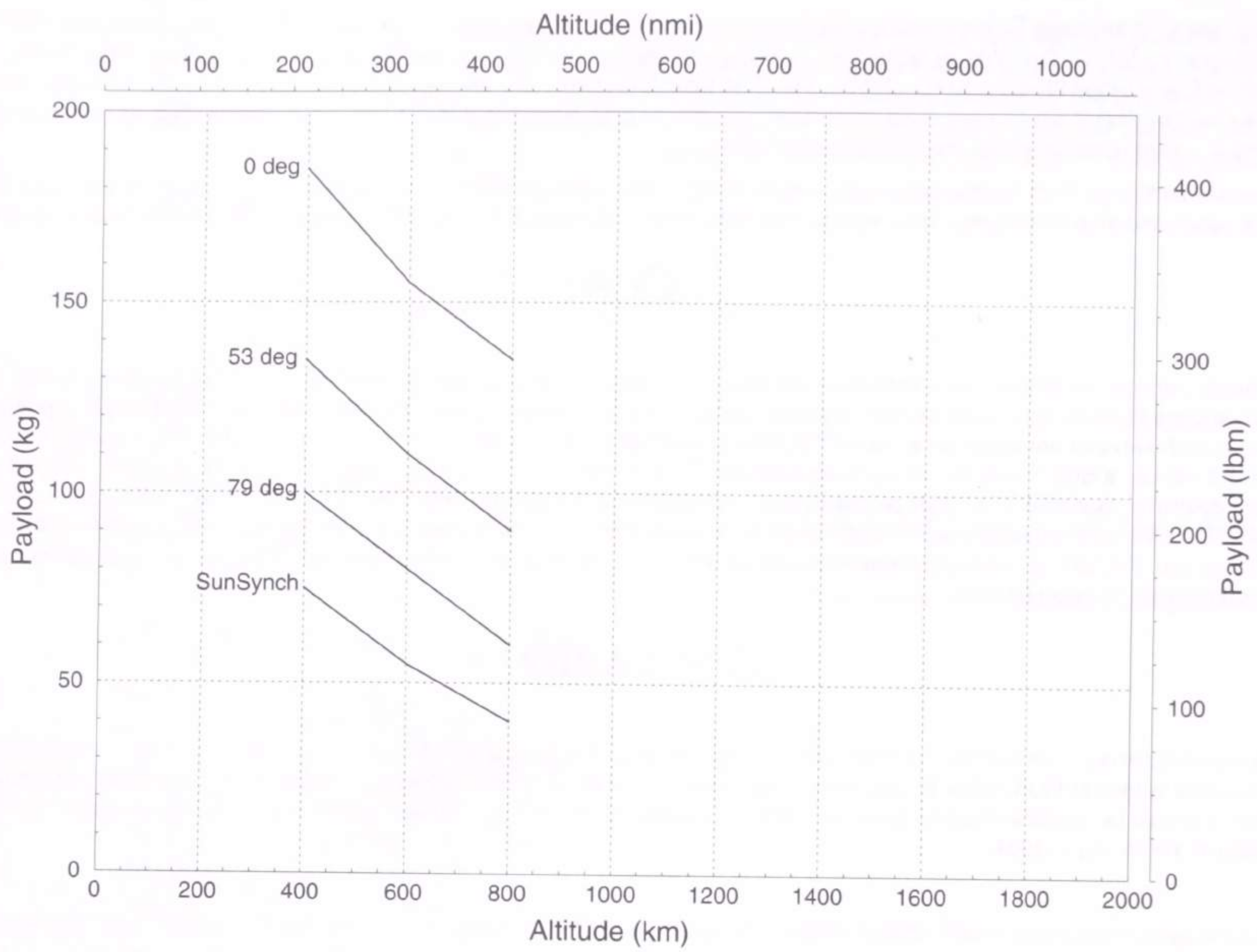
Shtil launch vehicles are converted from R-29RM missiles. According to START I treaty documents filed in January 2002, Russia had 96 actively deployed R-29RM missiles on six submarines, and 63 undeployed missiles, for a total of 159. In 1999 Russia announced that it would restart production of the R-29RM. Volna launch vehicles are converted from R-29R missiles. In January 2002 there were 112 deployed R-29R missiles, and 21 undeployed missiles.

Performance

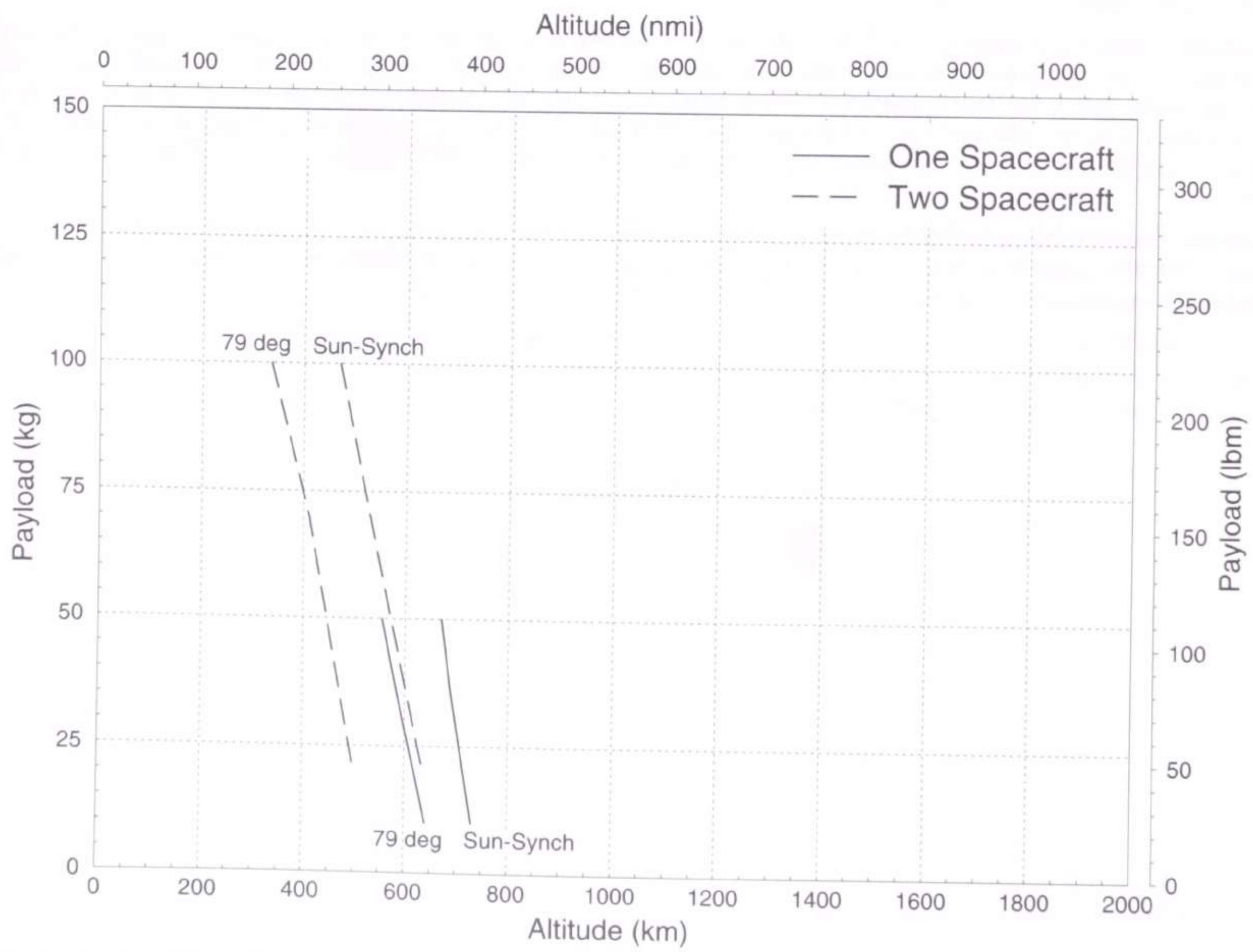
Standard inclinations for Shtil 1 and Volna launches are in the range 77–79 deg, using an established launch area and flight corridors. These trajectories begin in the Barents Sea off the northern coast of the Kola Peninsula and skirt the northern coast of the Novaya Zemlya island chain. Other inclinations can be reached by sailing the launch submarine to other locations. For example, to perform sun-synchronous missions, the launch submarine would sail approximately 1200 km (650 nmi) into the Norwegian Sea. Shtil 2 launches from the Nenoksa launch facility can safely achieve inclinations between roughly 77 and 88 deg. However, inclinations higher than 83 deg involve overflight of western Alaska, and would require U.S. government approval.

The payload capsules used to carry spacecraft on Shtil 1 launches are limited to a maximum mass of either 50 kg (110 lbm) or 70 kg (155 lbm) depending on the design. The Shtil 1 launch vehicle can carry a higher total payload mass if it is split into multiple spacecraft. Payload curves are given for both capsule types and for single and dual payloads.

Performance

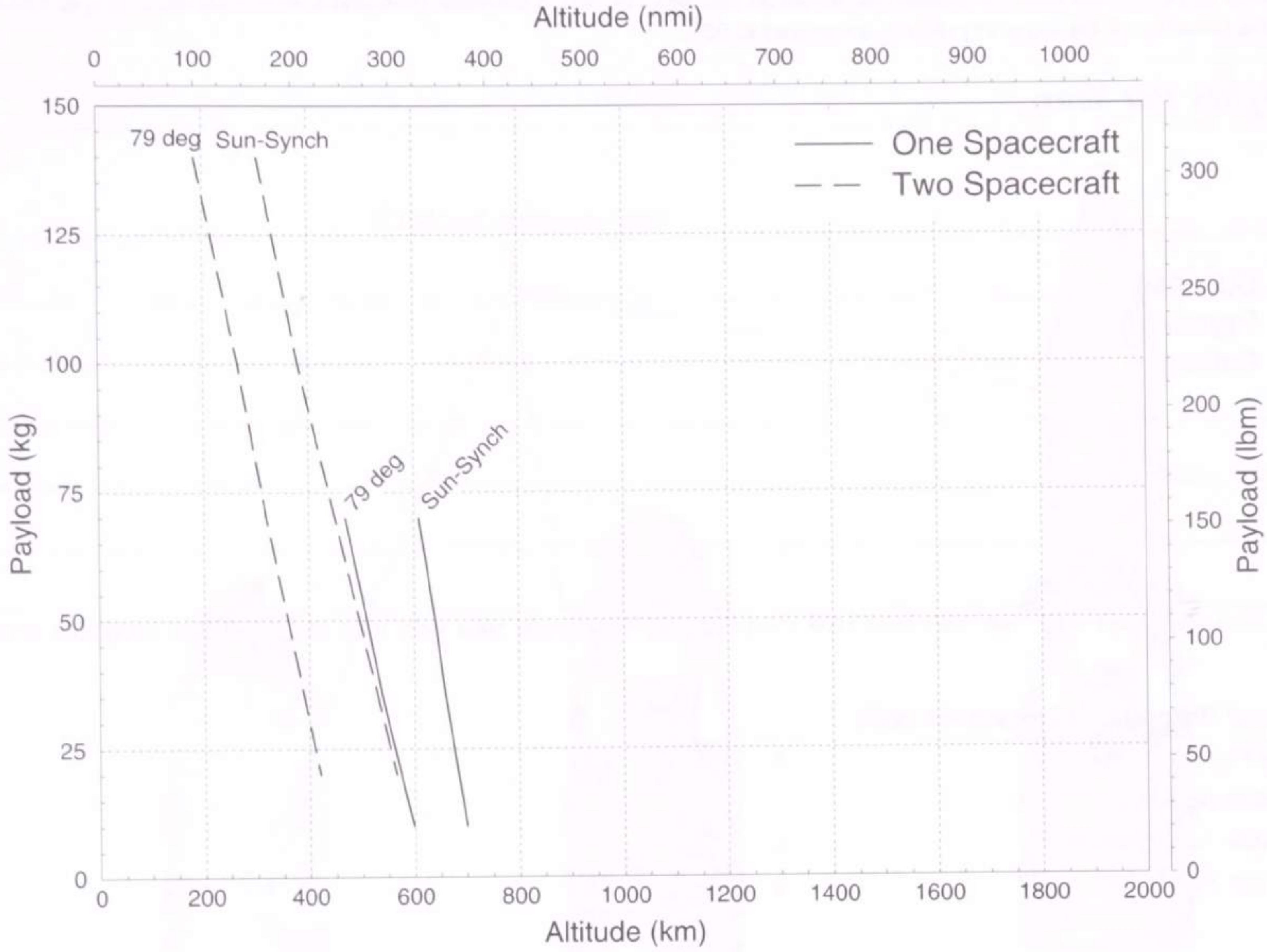


Volna: Performance to LEO

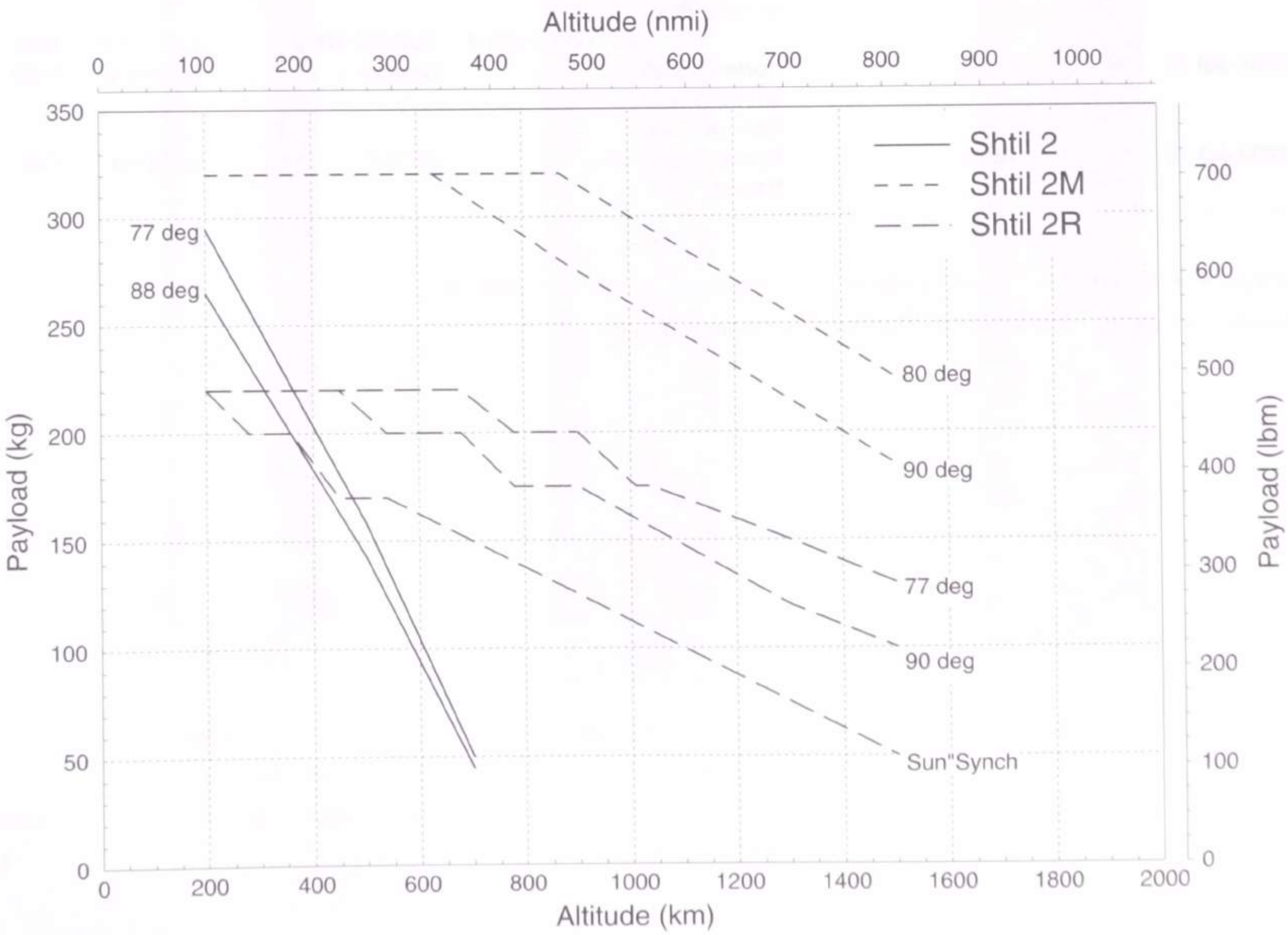


Shtil-1: Performance to LEO with Option 1 Payload Capsule(s)

Performance



Shtil-1: Performance to LEO with Option 2 Payload Capsule(s)



Shtil-2: Performance to LEO

Flight History

Certain space-related suborbital flights are included for reference in the following table, but are not counted in the tally of orbital flights and failures. In addition to the orbital launches, the RSM-54 missile performed 30 succesful launches between 1986 and March 2000 according to Makeyev. Based on this experience, the reliability of the launch system is advertised at 98%.

Orbital Flights Per Year

Flight Record (through 31 December 2003)	
Total Orbital Flights	1
Launch Vehicle Successes	1
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

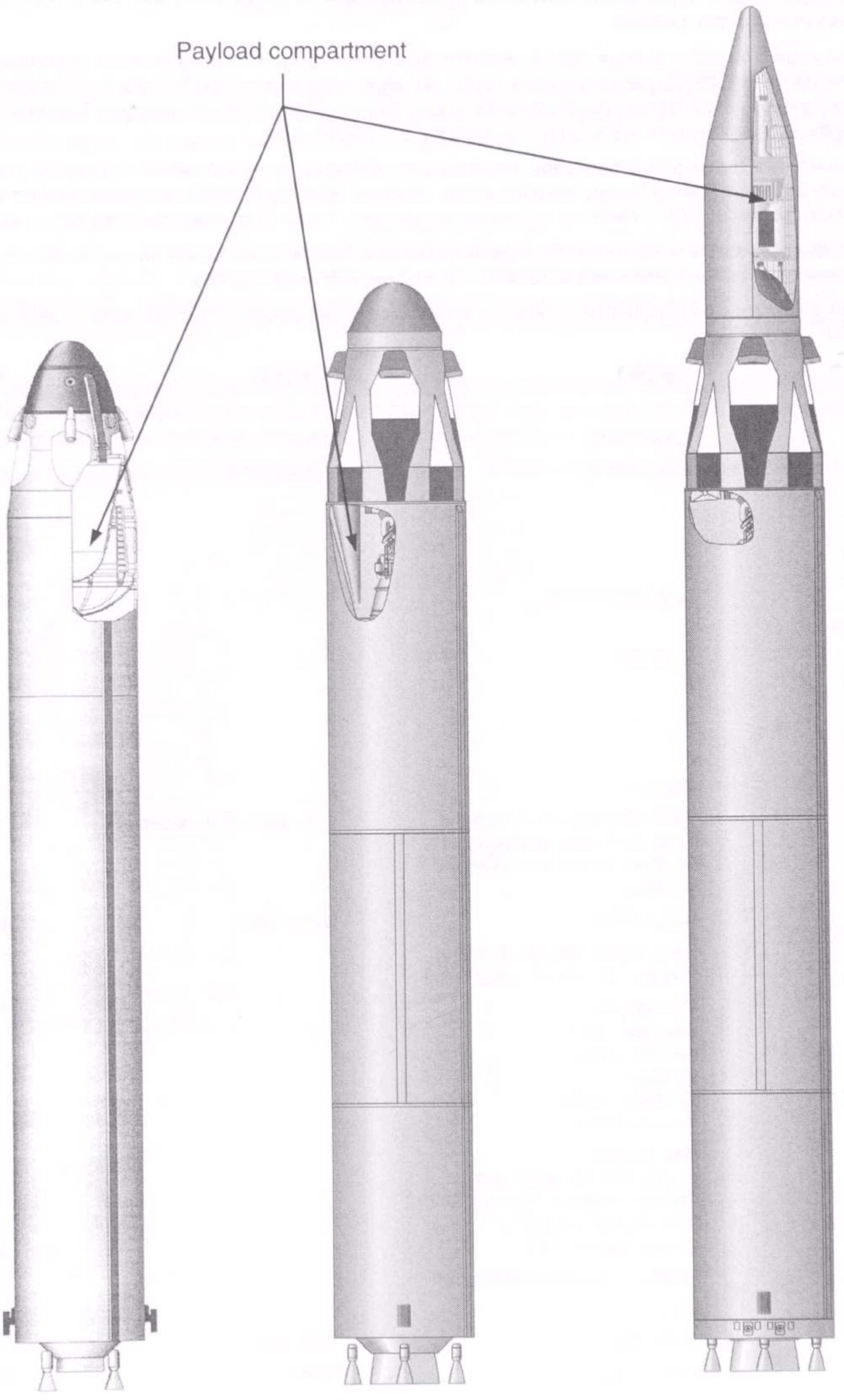
	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Launch Site	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
1	1998 Jul 07	—	Shtil 1		*Novomoskovsk* K-407 submarine, Barents Sea	1998 042A	TUBSAT N	8	LEO (78.9)	NGO	Germany
						1998 042B A	TUBSAT N1	3	LEO (78.9)	NGO	Germany
F	**2001 Jul 20**		**Volna**		**Borisoglebsk submarine Barents Sea**		Cosmos 1		suborbital	NGO	USA
F	**2002 Jul 12**		**Volna**		**Ryazan submarine Barents Sea**		IRDT 2	146	suborbital	CML	Russia

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Vehicle Design

Overall Vehicle



	Volna	**Shtil 1**	**Shtil 2, 2R**
Height	14.2 m (46.6 ft)	14.8 m (48.5 ft)	18.4 m (60.4 ft)
Gross Liftoff Mass	35.4 t (78.0 klbm)	40 t (88 klbm)	40 t (88 klbm)
Thrust at Liftoff	?	~ 810 kN (182 klbf)	~ 810 kN (182 klbf)

Vehicle Design

Stages

The extremely compact design of Volna and Shtil is unique among space launch vehicles, and is a consequence of the limited volume available in the launch tube of a submarine. The stages use common tank bulkheads not only between tanks on the same stage, but between stages as well. For example, there is no upper bulkhead on the first stage fuel tank. It is mated directly to the second stage aft bulkhead, and the second stage engine is submerged directly into the first stage fuel, completely eliminating the wasted space of an engine compartment. All stages use N_2O_4/UDMH storable liquid propellants. The first stage RD-0243 engine is described as the highest performance engine in its class, with a thrust-to-weight ratio of approximately 100:1 and an extremely high chamber pressure.

The Volna/R-29R third stage is a postboost stage with four axial thrusters. The warhead or payload is carried in a compartment attached to the bottom of the third stage. When the Shtil/R-29RM was developed, a larger main engine was added to the third stage and located in the position of the warhead compartment. Multiple reentry vehicles or payloads are clustered around the main engine in the compartment. The main engine separates from the third stage after firing, and the apogee injection burn is completed using the external thrusters.

The Volna has a diameter of 1.8 m (5.9 ft). When further developing an existing space launch vehicle it is unusual to change the tank diameter of a stage. Growth is usually accommodated by making the tanks longer. However, when the R-29RM was developed from the R-29R, it was not possible to stretch the rocket sufficiently and still fit it inside the submarine launch tubes. Therefore the diameter of the Shtil is slightly larger, at 1.9 m (6.2 ft).

For orbital missions, Volna adds a small solid motor to the basic R-29R missile. This motor was previously used to de-orbit film capsules from the Yantar-2K series of reconnaissance satellites. It produces 5.6 kN (1.25 klbf) of thrust with an Isp of 272 s.

Because Shtil and Volna are based on active Russian missiles, few technical details are available. The following data are for Shtil only.

	Stage 1	Stage 2	Stage 3
Dimensions			
Length	7.3 m (24 ft)	4.9 m (16 ft)	2.8 m (9.2 ft)
Diameter	1.9 m (6.2 ft)	1.9 m (6.2 ft)	1.9 m (6.2 ft) maximum
Mass			
Propellant Mass	?	?	?
Inert Mass	?	?	?
Gross Mass	23.3 t (49.2 klbm)	?	?
Propellant Mass Fraction	?	?	?
Structure			
Type	?	?	?
Material	?	?	?
Propulsion			
Engine Designation	RD-0243	?	?
Number of Engines	1 RD-0243 system comprising 1 RD-0244 main engine and 1 RD-0245 vernier engine with 4 thrusters	1, no verniers apparent	1 main engine + 4 thrusters
Propellant	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Average Thrust	Main engine: 682 kN (153 klbf) Verniers: 211 kN (47.4 klbf) total	?	?
Isp	Main engine: Sea level: 280 s Vacuum: 310 s Verniers: Sea level: 280 s Vacuum: 300 s	?	?
Chamber Pressure	Main engine: roughly 275 bar (4000 psi) Vernier: roughly 147 bar (2130 psi)	?	?
Nozzle Expansion Ratio	Main engine: roughly 22:1 Vernier: roughly 14:1	?	?
Propellant Feed System	Staged-combustion turbopump	?	?
Mixture Ratio (O/F)	2.6:1	?	?
Throttling Capability	100% only	100% only	?
Restart Capability	None?	None?	?
Tank Pressurization	?	?	?
Attitude Control			
Pitch, Yaw	Verniers	?	?
Roll	Verniers	?	?
Staging			
Nominal Burn Time	Main engine: 74 s Verniers: 79 s	94 s	Main engine: 87 s Thrusters: 87 s + 264 s
Shutdown Process	?	?	Command shutdown
Stage Separation	?	?	?

Vehicle Design

Attitude Control System

The first-stage main engine is fixed. Steering is performed by four thrust chambers of the vernier engine system. Second-stage attitude control is unknown. The Volna third stage has four external thrusters, while the Shtil third stage has twelve. These may be used for third-stage attitude control and as part of a velocity trim postboost propulsion system.

Avionics

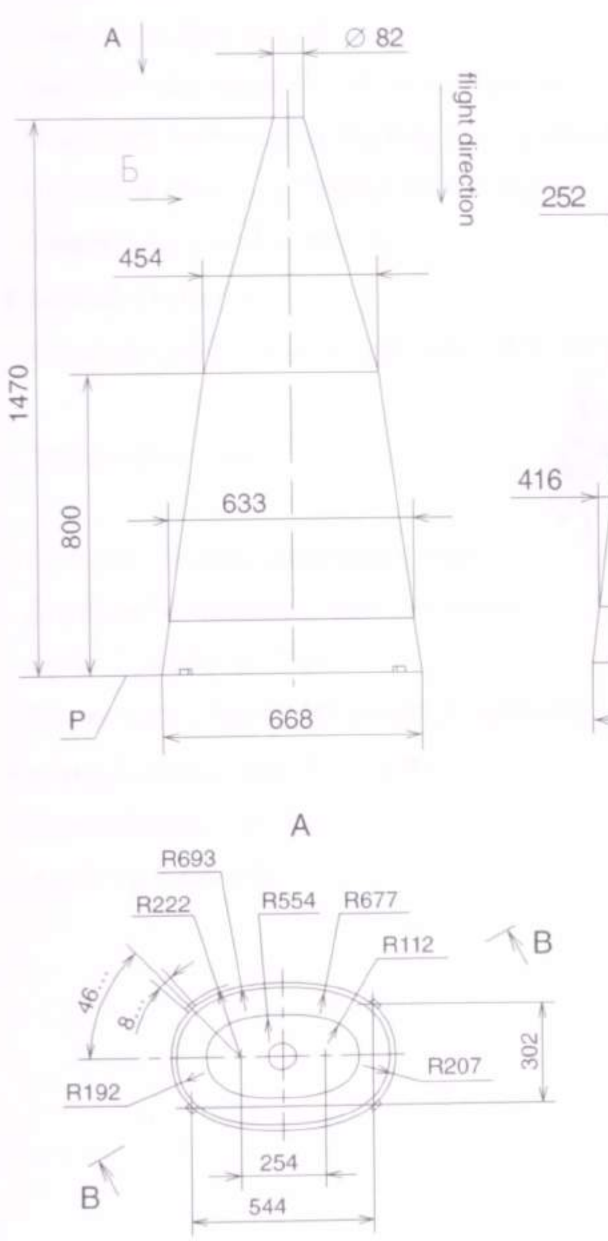
Shilt and Volna use inertial guidance. The Shtil onboard control system, designated BCSE, is capable of guiding the vehicle to orbits up to 400 km without modifications. If higher altitudes are required for a mission, hardware improvements can be made at the factory or maintenance base. Shtil uses a SCOOT telemetry system to provide vehicle and payload telemetry and tracking data in compliance with START treaty requirements.

Payload Fairing

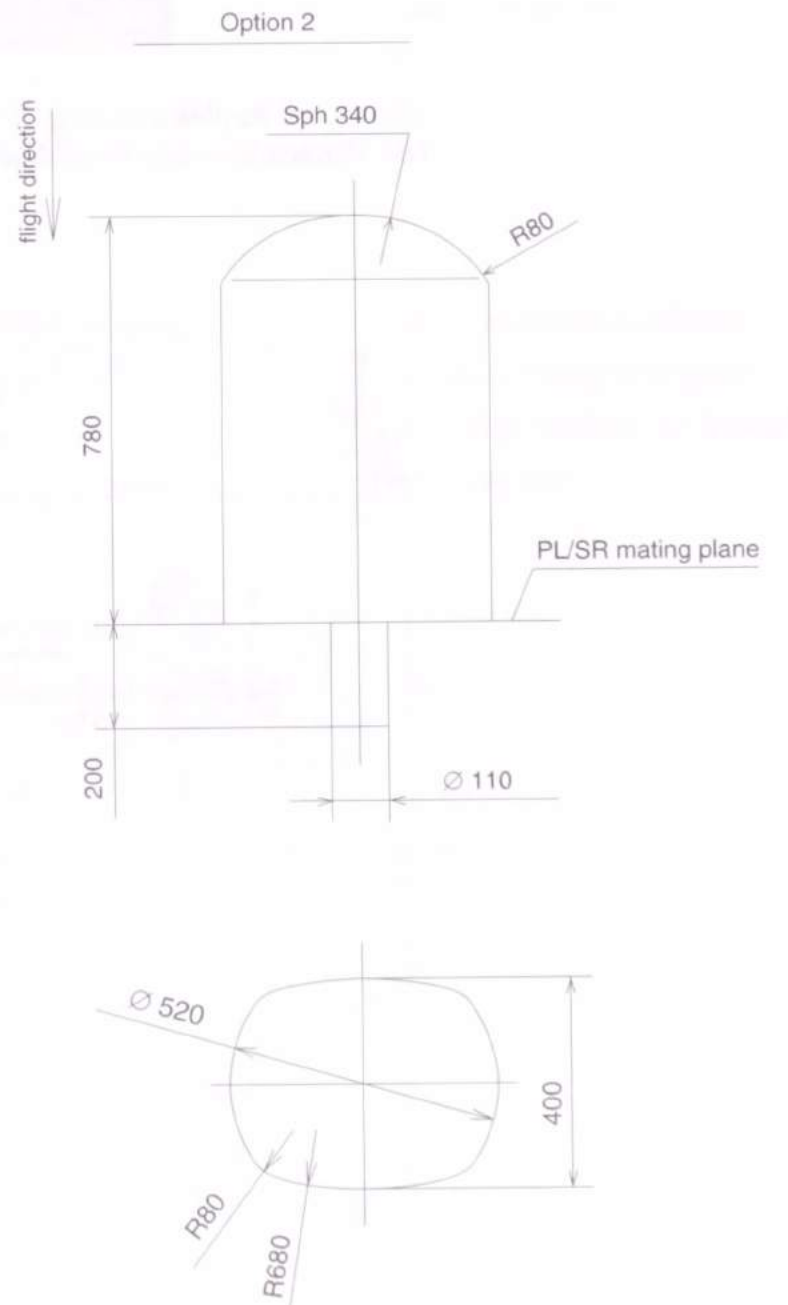
The Shtil 1 does not have a conventional payload fairing. Instead, spacecraft are mounted in protective capsules underneath the third stage, next to the third stage main engine. When orbit is achieved, the capsules are deployed from the third stage, the mounting plate of the capsule is separated using springs, and the spacecraft is released. Two payload capsule options are available. Option 1 is conical with a usable volume of 0.183 m^3 (6.5 ft^3), while Option 2 is cylindrical with a usable volume of 0.150 m^3 (5.3 ft^3). The capsules are made of fiberglass with thermal insulation and aluminum–magnesium mounting plates.

The Shtil 1 payload capsules do not accommodate larger satellites, so the Shtil 2 vehicle uses a more conventional payload fairing. The Shtil 2R configuration is shown, with a small upper stage housed in the payload fairing. The fairing is made up of two halves separated by explosive bolts. The payload is mounted to a truss structure with a five-point attachment interface. Payload access doors are optional.

Like the Shtil 1, Volna payloads are suspended from the underside of the third stage. For orbital launches, a small solid motor is carried within this volume.

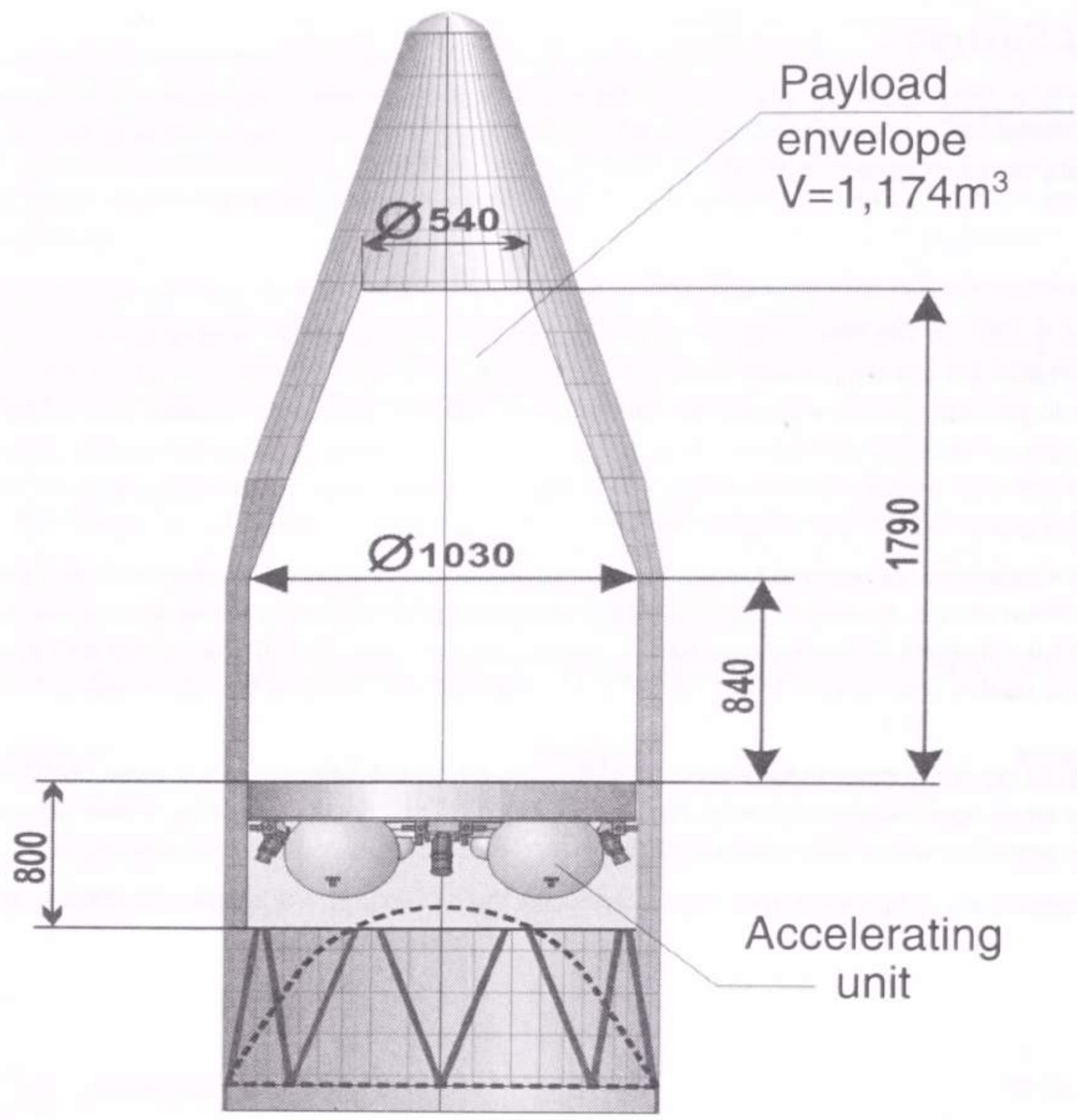


Shtil 1 Payload Capsule Option 1
(All dimensions are in millimeters.)

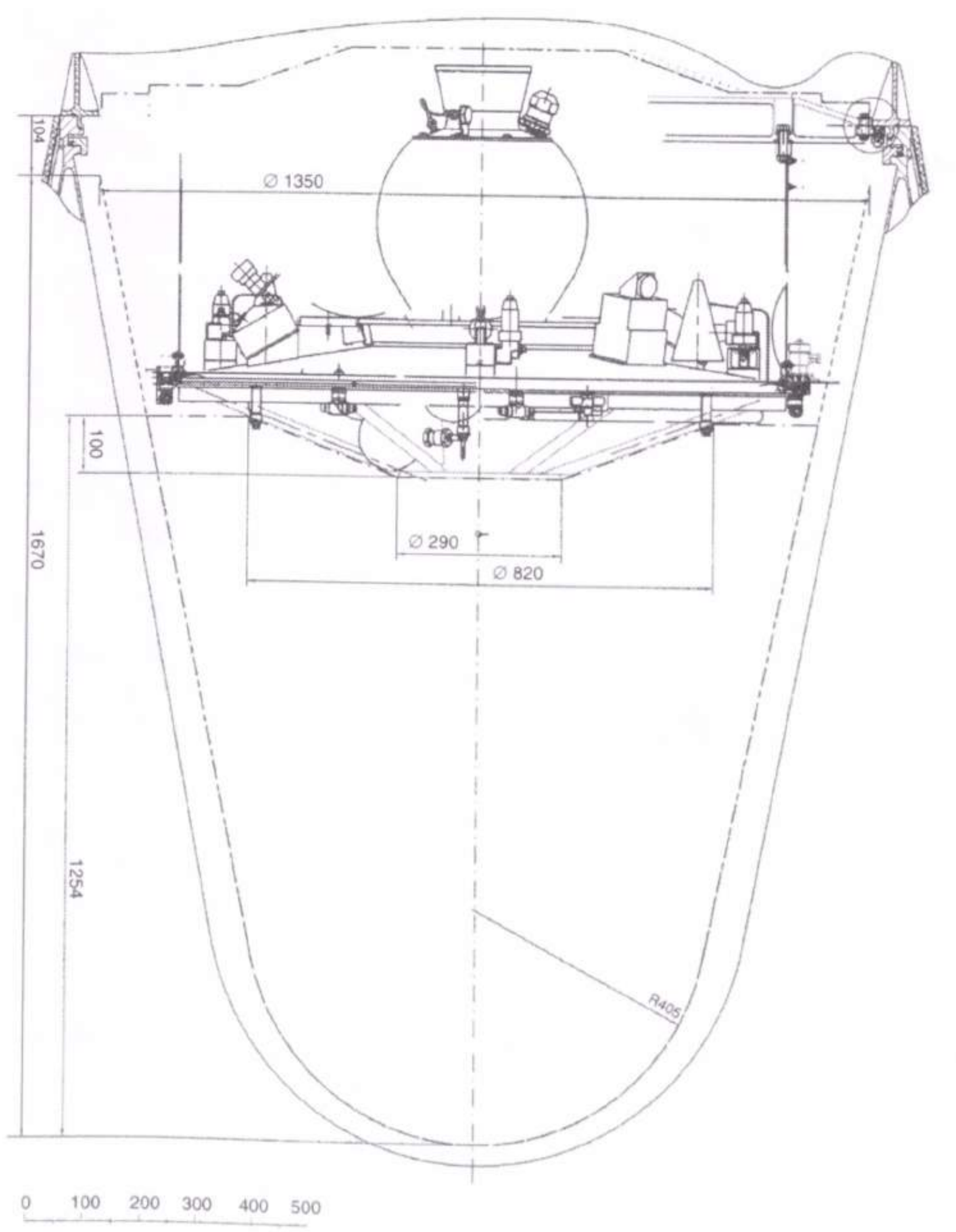


Shtil 1 Payload Capsule Option 2
(All dimensions are in millimeters.)

Vehicle Design



Shtil 2, 2R Payload Fairing
(All dimensions are in millimeters.)



Volna Payload Compartment
(All dimensions are in millimeters.)

Payload Accommodations

	Shtil-1	Shtil-2 Series
Payload Compartment		
Maximum Payload Diameter	See diagram	1030 mm (40.6 in.)
Maximum Cylinder Length	See diagram	Shtil 2: 1640 mm (64.0 in.) Shtil 2R: 840 mm (32.7 in.) Shtil 2M: 1390 mm (54.2 in.)
Maximum Cone Length	See diagram	950 mm (37.0 in.)
Payload Adapter Interface Diameter		650 mm (25.4 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–10 months	T–10 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	?	?
On-Pad Storage Capability	3–7 days	?
Last Access to Payload	?	?
Environment		
Maximum Axial Load	8 *g* tension	8 *g* compression
Maximum Lateral Load	±2 *g*	±1 *g*
Minimum Lateral/Longitudinal Payload Frequency	40 Hz	15 Hz
Maximum Acoustic Level	?	133 dB
Overall Sound Pressure Level	145 dB	140 dB
Maximum Flight Shock	6000 *g* at 1 kHz	1500 *g* at 500 Hz
Maximum Dynamic Pressure on Fairing	?	?
Maximum Aeroheating Rate at Fairing Separation	Negligible–separation occurs in orbit	?
Maximum Pressure Change in Fairing	19.6 kPa/s	?
Cleanliness Level in Fairing	Class 100,000	?
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	±4–10 km injection altitude ±0.08–0.2 deg inclination	±2–3 km injection altitude ±0.08–0.16 deg inclination
Attitude Accuracy	?	±0.8 deg pitch/roll, ±1.3 deg yaw ±0.5 deg/sec
Nominal Payload Separation Rate	?	?
Deployment Rotation Rate Available	?	?
Loiter Duration in Orbit	?	?
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes
Multiple/Auxiliary Payloads		
Dual/Multiple Manifest	Available option	Available option
Auxiliary Payloads	Unknown	Unknown

Production and Launch Operations

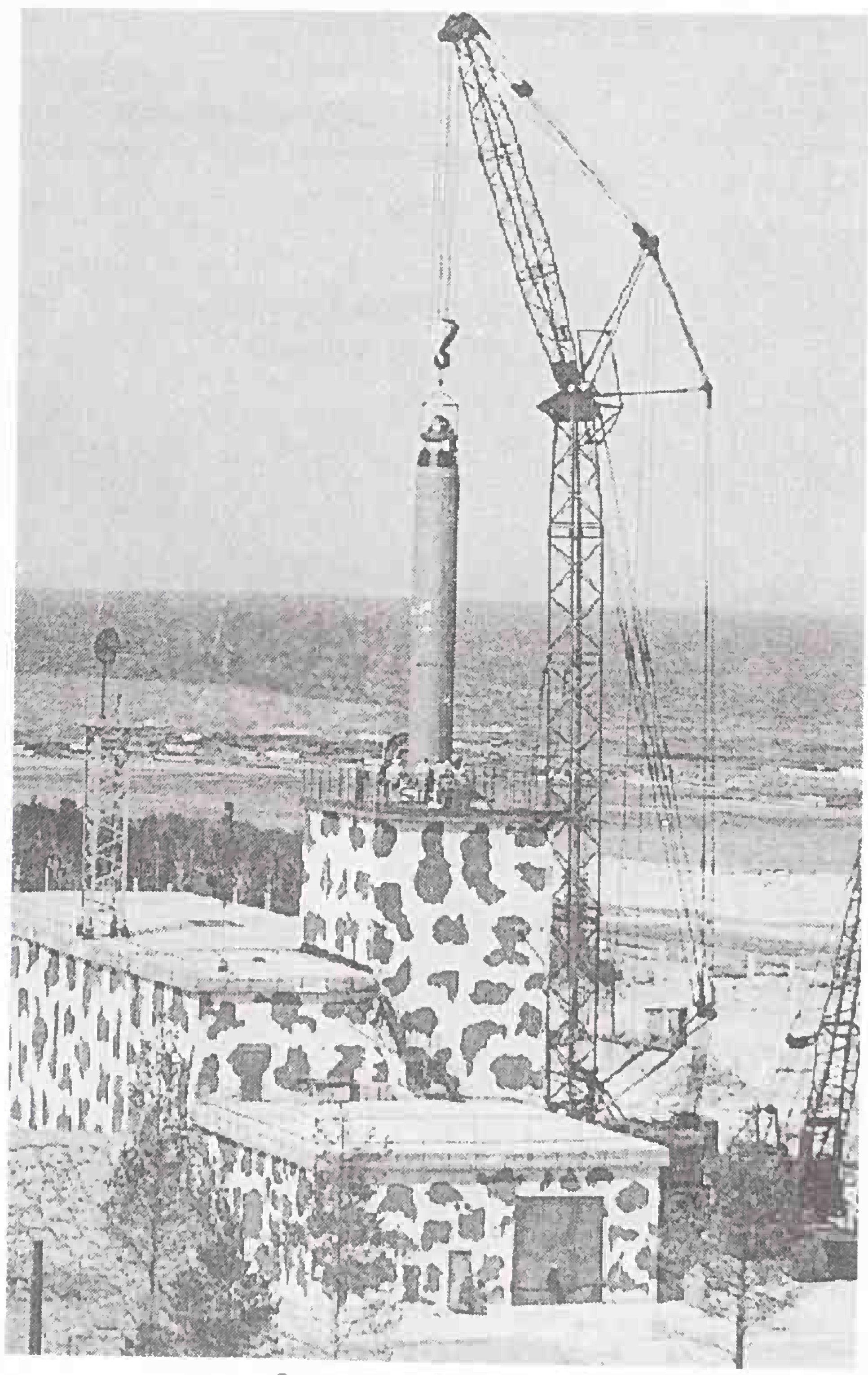
Production Operations

The Shtil and Volna are produced by modifying previously produced R-29R and R-29RM missiles. Both the missiles and their space launch versions were developed by the State Research Center Makeyev Design Bureau and its predecessor organization, the Machine Building Design Bureau (KBM). Makeyev is located in Miass, in the Chelyabinsk region. The R-29R missile has been out of production for many years, and production of the R-29RM ended in the early 1990s. However, in 1999, Russia announced that it would restart production of the R-29RM.

Launch Operations

Shtil 1 and Volna launches are performed from submarines in a manner essentially identical to the launch of a ballistic missile. Volna launches are performed from Project 667BDR Kalmar-class submarines. Shtil 1 launches are conducted from Project 667BDRM Delfin-class submarines. The first Shtil orbital launch was performed by the K-407 *Novomoskovsk*, part of the Third Flotilla of the Russian Navy's Northern Fleet. The submarines are based near Murmansk, and perform space launches from a designated test area in the Barents Sea near the coast of the Kola Peninsula. Launches can be performed with the submarine on the surface or submerged.

Shtil 2 vehicles do not fit onboard submarines because of the extended payload fairing. Instead, launches will be conducted from the land-based Nenoksa State Central Marine Test Site. This facility has been used for initial testing of Soviet and Russian SLBMs since the 1960s. It is located on the coast of the White Sea west of Severodvinsk. (See Spaceports section for map.) At Nenoksa the launch vehicle and payload are prepared in the Assembly–Test Building, which has a 1300 m^2 (14,000 ft^2) main hall. It typically takes 25–30 days to prepare the launch vehicle. The rocket is then transported to the launch facility, designated Ground Stand HC-37. This building contains a single above-ground launch silo that is similar to the launch tubes used on the Delfin submarines. The launch vehicle, without the payload, is lowered into the silo using a crane. The encapsulated payload is then mated to the top of the launch vehicle. Operations at the launch stand take only 36–48 h before the vehicle is ready to launch. When the launch occurs, the Shtil is ejected from the silo, and its main engine ignites once clear.

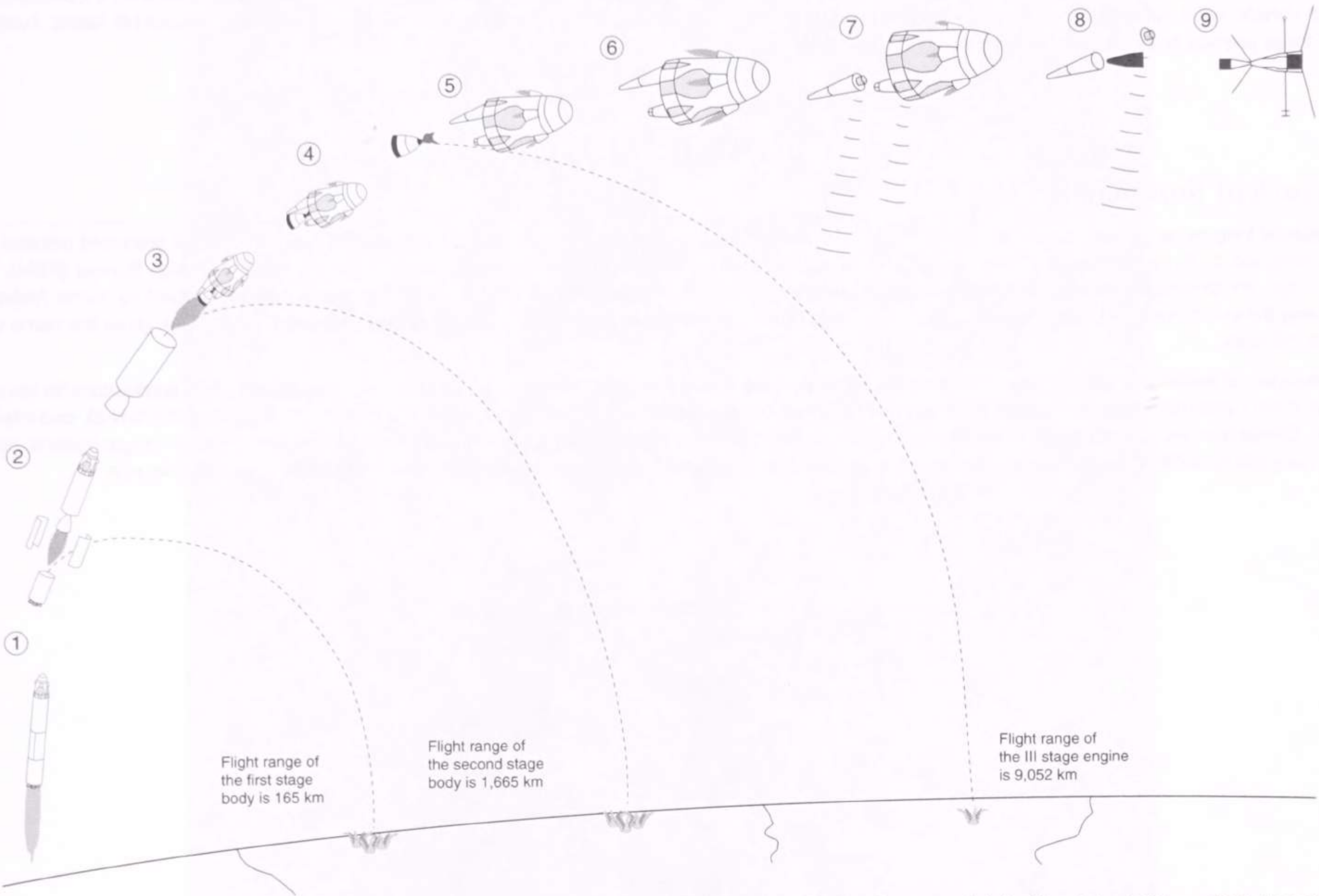


Courtesy State Rocket Center Makeyev Design Bureau.

An R-29RM ballistic missile is loaded at the HC-37 launch facility at Nenoksa.

Production and Launch Operations

Flight Sequence



Typical Mission Profile for Spacecraft Injection into 400-km, 78.9-deg orbit.

Moments featuring the Shtil LV motion *Horb = 400 km, i = 78.9 deg*

	Flight time, s	LV velocity, m/s	Flight altitude, km	Trajectory angle, degrees	Geodesic range, km
Rocket takeoff	0.0	0.0	0	90.00	0
Separation of the first stage body	75.02	1369.9	30	33.34	31
The second stage body astrodome separation, beginning mode "A"	169.4	4370.8	109	10.91	258
Start of mode "M"	256.4	7151.4	174	6.67	727
Separation of the third stage engine	319.9	7100.5	224	5.99	1163
Apogee trajectory leg:					
beginning	641.1	6938.4	387	2.26	3299
end	905.8	7579.2	400	0.0	5098
Separation of the capsule with S/C/, SO drift from the orbit	910.9	7581.1	400	0.0	5134
Capsule division, S/C release	946.9	7581.1	400	0.0	5383
S/C starts functioning in orbit	956.9	7581.1	400	0.0	5460

Vehicle Upgrade Plans

The Babakin Space Center, a branch of the Lavochkin Association, has proposed the Kaplya upper stage for use on a Shtil variant dubbed Shtil K. Kaplya is a two-ton upper stage based on the Fregat upper stage produced by Lavochkin. It would replace the existing third stage/postboost vehicle of Shtil and increase Shtil's performance capability to 700 kg (1540 lbm). Shtil K would have a larger payload fairing that matches the diameter of the Shtil vehicle with a usable payload envelope 1.5 m (59 in) in diameter.

Makeyev has been involved in a number of launch vehicle development proposals, including an air-launched version of Shtil called Shtil 3A or RIF-MA; the methane-fueled Riksha, which would have been launched from a converted refrigerated fishing trawler; and the Unity ULV-22 or Yedintsvo, a LOX/kerosene launch vehicle with a five-ton payload that was to have been built with United Launch Systems International and launched from Hummock Hill Island, Australia. All of these projects have been cancelled because of lack of funds.

Vehicle History

Historical Summary

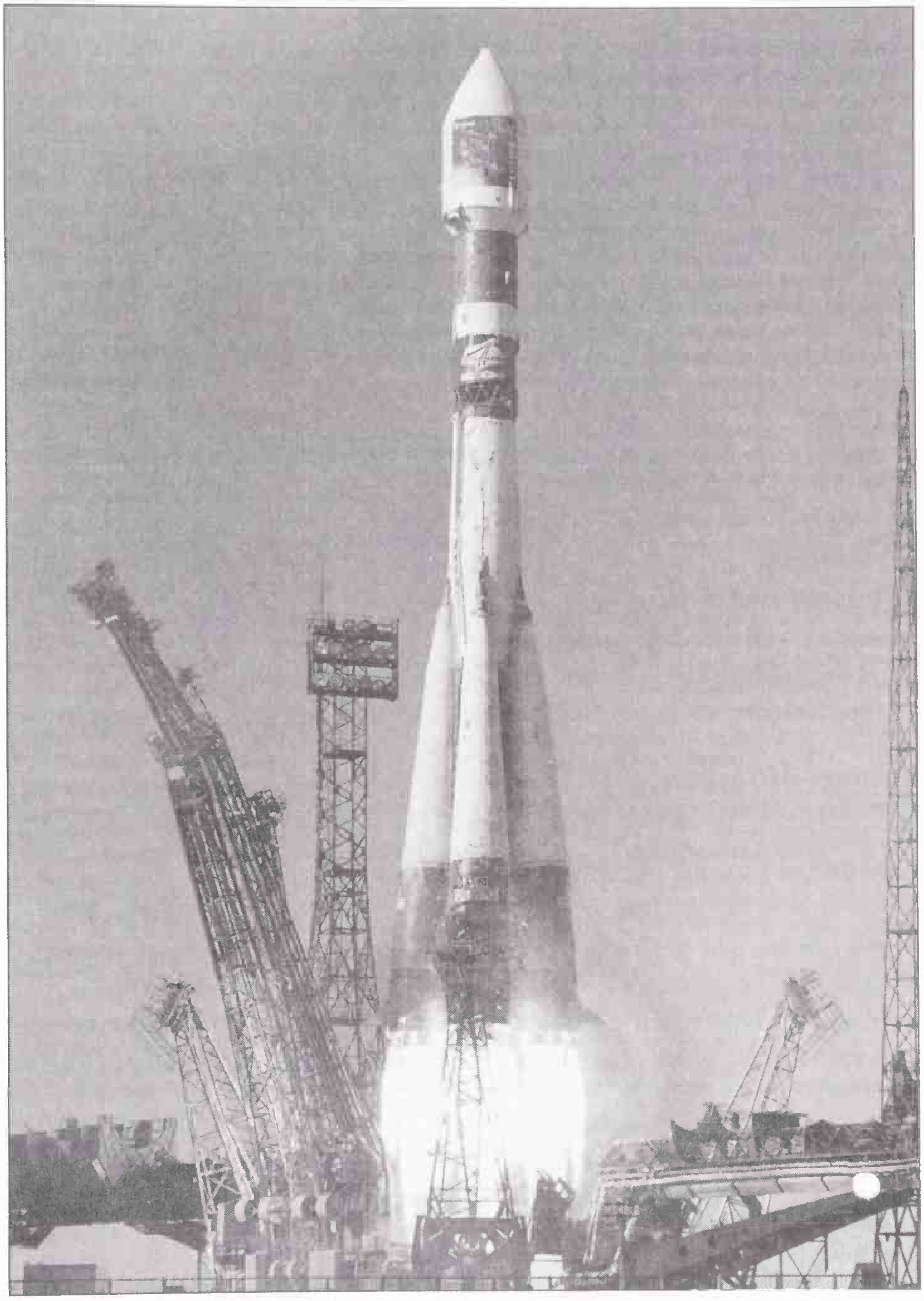
The special long range missile design bureau SKB-385 was created by a government resolution in December 1947. Before 1956 it produced missiles that were designed by Sergei Korolev's OKB-1 design bureau. During the mid-1950s the bureau, known as the Machine Building Design Bureau (KBM), was responsible for developing the Soviet Union's submarine-launched ballistic missiles. From 1955 to 1985 the bureau was managed by Victor Makeyev. Following the end of the Cold War, the organization was restructured, and the new State Rocket Center Makeyev Design Bureau was given the name of its former manager.

As production of ballistic missiles declined following the collapse of the Soviet Union, the Makeyev bureau began to search for civilian applications for its capabilities. It has performed suborbital launches of experimental capsules since at least 1991, using the small Zyb missile (R-21A/RSM-25/SS-N-6), and later the Volna. Orbital launches began in 1998. Launches have been infrequent, but Makeyev has gradually attracted the attention of Western organizations, particularly nonprofit scientific organizations and universities, because its launch vehicles are among the least-expensive options for space launch.

Courtesy State Rocket Center Makeyev Design Bureau.

An R-29RM launch from Nenoska.

Soyuz and Molniya



Courtesy Starsem.

The Soyuz/Molniya family of launch vehicles is best known for carrying the Soviet Union's manned spacecraft dating back to Yuri Gagarin, but it is also Russia's workhorse launch system for unmanned spacecraft. More than 1600 space launches have been performed by the Soyuz family, roughly 40% of all launches in the world. Soyuz is now available commercially through the Starsem joint venture.

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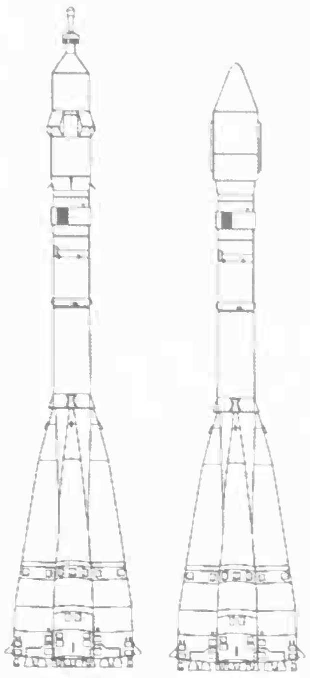
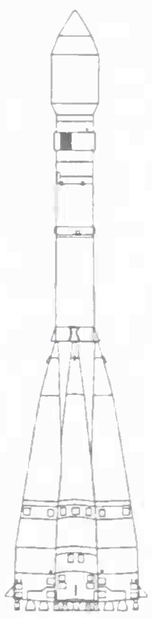
General Description



Soyuz FG Soyuz U



Soyuz 2

	Soyuz U and FG	Soyuz 2

Summary

Soyuz U and FG	Soyuz 2
The Soyuz is the most frequently used member of the Soyuz/Molniya/Vostok vehicle family, and the most active launch vehicle in the world. Over 800 launches of Soyuz have taken place, half of the more than 1600 launches by this vehicle family. Soyuz carries medium-class payloads such as military reconnaissance satellites and Soyuz capsules for the International Space Station. The Soyuz U has long been the primary variant of the family, but new configurations are beginning to replace it. The Soyuz FG is similar to the Soyuz U but has improved injectors on the first and second stage engines for added performance. Human missions with Soyuz TMA capsules are now launched on the FG configuration. The new Fregat upper stage allows Soyuz to reach higher altitude orbits. Since 1996 Soyuz has been marketed by the joint French/Russian company Starsem.	Several major enhancements to the venerable Soyuz are in development under the long running Soyuz 2 upgrade program. New digital avionics built in Russia will replace the old analog flight control system, which includes many components from Ukraine. A larger payload fairing will be available, and a new high performance engine will be used on the third stage. The Soyuz 2 will be available for both Russian government and commercial customers. The Fregat stage will be used for orbits above 500 km (270 nmi).

Status

Soyuz U and FG	Soyuz 2
Operational: First Soyuz launch in 1965, first Soyuz U in 1973, first Soyuz U/Fregat in 2000, first Soyuz FG in 2001.	In development. First launch planned in 2005.

Origin

Soyuz U and FG	Soyuz 2
USSR/Russia	Russia

Key Organizations

	Soyuz U and FG	Soyuz 2
Marketing Organization	Starsem	Starsem
Launch Service Provider	Starsem	Starsem
Prime Contractor	Central Specialized Design Bureau (TsSKB)	Central Specialized Design Bureau (TsSKB)

Primary Missions

Soyuz U and FG	Soyuz 2
Medium payloads and manned spacecraft to LEO	Medium payloads and manned spacecraft to LEO and small spacecraft to GTO

Estimated Launch Price

Soyuz U and FG	Soyuz 2
\$30–50 million (Starsem, 1999)	\$30–50 million (Starsem, 1999)

Spaceports

	Soyuz U and FG	Soyuz 2
Launch Site	Baikonur LC 1 and LC 31	Baikonur LC 1 and LC 31
Location	45.6° N, 63.3° E and 46.0° N, 63.5° E	45.6° N, 63.3° E and 46.0° N, 63.5° E
Available Inclinations	52, 65, and 70 deg and SSO directly, others with upper-stage plane change maneuver	52, 65, and 70 deg and SSO directly, others with upper-stage plane change maneuver
Launch Site	Plesetsk LC 16, LC 43 (2 pads)	Plesetsk LC 16, LC 43 (2 pads)
Location	62.8° N, 40.4°-40.7° E	62.8° N, 40.4–40.7° E
Available Inclinations	63, 67, 73, 82, and 90 deg and SSO directly, others with upper-stage plane change maneuver	63, 67, 73, 82, and 90 deg and SSO directly, others with upper-stage plane change maneuver

Performance Summary

Performance shown below is for vehicle configurations of interest for commercial satellites. Performance differs for other configurations (e.g., manned spacecraft).

	Soyuz U and FG	Soyuz 2
200 km (108 nmi), 51.8 deg	7000 kg (15,430 lbm)	7900 kg (17,400 lbm)
200 km (108 nmi), 90 deg	?	?
Space Station Orbit: 407 km (220 nmi), 51.6 deg	7200 kg (15,875 lbm)	7400 kg (16,300 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	4300 kg (9480 lbm)	4500 kg (9920 lbm)
GTO: 185 × 35,786 km (100 × 19,323 nmi) 28 deg	1660 kg (3660 lbm)	1950 kg (4300 lbm)
Geostationary Orbit:	420 kg (926 lbm)	520 kg (1146 lbm)

Flight Record (through 31 December 2003)

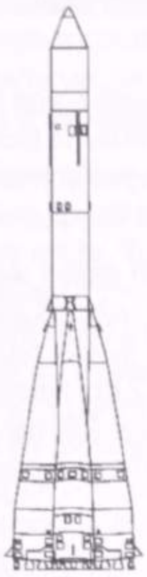
	Soyuz U and FG	Soyuz 2
Total Orbital Flights	713	0
Launch Vehicle Successes	692	0
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	21	0

Flight Rate

Soyuz U and FG	Soyuz 2
7–13 per year	To be determined

General Description

Molniya M



Molniya M

Summary

The Molniya is an older four-stage version of the basic Soyuz. The fourth stage is typically used to deliver communications and early warning spacecraft to high-inclination elliptical orbits for the Russian military space forces. It is not used for commercial launches.

Status

Operational. First launch 1960.

Origin

USSR/Russia

Key Organizations

Marketing Organization	Not marketed
Launch Service Provider	RVSN
Prime Contractor	TsSKB

Primary Missions

Communications and military payloads to elliptical orbits

Estimated Launch Price

$30–40 million (FAA)

Spaceport

Launch Site	Plesetsk LC 16, LC 43 (2 pads)
Location	62.8° N, 40.4–40.7° E
Available Inclinations	63, 67, 73, 82, and 90 deg and SSO directly, others with upper-stage plane change maneuver

Performance Summary

200 km (108 nmi), 63 deg	No capability
180 km (97 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	1500 kg (3300 lbm)
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	290
Launch Vehicle Successes	271
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	19

Flight Rate

0–2 per year

NOMENCLATURE

There is little consensus on the name of the family of launch vehicles that includes the Soyuz and Molniya. It is often referred to as the A class, after the Sheldon name, however this reflects a Western perspective. It is occasionally referred to as the R-7 or Semyorka family after the designations of the original ICBM predecessor. The family is referred to here as Soyuz, after the most commonly used and widely recognized version.

The Soyuz family includes several distinct vehicle series of which two are still in operation. The Molniya "Lightning" vehicles have a fourth stage that is used for high-energy orbits. The Soyuz "Union" configurations have historically been a three-stage system used for very low altitude missions including manned spacecraft. In the last few years the new Fregat upper stage has become available on Soyuz enabling it to perform additional types of missions. Two variations of the Soyuz are currently operational, the older Soyuz U and the newer Soyuz FG. These are due to be replaced by the Soyuz 2 series.

Two distinct configurations are planned as part of the Soyuz 2 program. The Soyuz 2-1A configuration is based on the Soyuz FG with digital avionics and modified third-stage structures. The 4-m class payload fairing and Fregat upper stage are optional. The Soyuz 2-1B configuration will add the RD-0124 engine to the third stage. The name "Rus" has also been used for Soyuz 2. This refers to fact that components from other former Soviet republics are being replaced with Russian-made components. In the past Starsem has used the designation Soyuz ST to refer to several Soyuz 2 configurations. This designation is no longer used for a specific vehicle, but the ST designation is still applied to the Starsem 4-m class payload fairing. Starsem now uses the generic "Soyuz" name to refer to all configurations, without indicating the specific vehicle type. For launches in the 2002–2003 timeframe, the Starsem "Soyuz" is a Soyuz FG with Fregat upper stage.

COST

In 1999 Starsem reported that prices for commercial Soyuz launches range from $30 to $50 million, depending on vehicle configuration and mission requirements. The European Space Agency paid $69 million for two launches of its Cluster science satellites in 2000.

Soyuz is the only launch system in operation today upon which human passage can be purchased. Extra seats in Soyuz spacecraft, which are launched on Soyuz rockets, are offered commercially at a list price of $20 million. The first "space tourists" to take advantage of this capability, Dennis Tito and Mark Shuttleworth, were launched in 2001 and 2002 respectively. Both reportedly paid less than list price for their rides. European space agencies have been paying for launches of their astronauts for much longer, albeit with less fanfare. For example, ESA agreed to pay 14.5 million ($14.3 million) for the launch of Spanish astronaut Pedro Duque in October 2003. The quoted costs of flying past European astronauts have ranged from $13.7 to $18 million.

AVAILABILITY

The Soyuz and Molniya launch vehicles have been used for Soviet/Russian government payloads since 1965 and 1960, respectively. Beginning in the 1980s they were marketed internationally to a limited degree. Two IRS Indian remote sensing satellites were launched by a Vostok (a no-longer active member of the Soyuz family) in 1991, and a Molniya-M in 1995. Since 1996, Soyuz launch services have been marketed more successfully by Starsem, a French/Russian partnership. Starsem is jointly owned by the European companies EADS (35%) and Arianespace (15%), as well as the Russian Space Agency (25%) and the Samara Space Center (25%), which designs and produces the vehicles. Starsem primarily markets present and future versions of the Soyuz.

The new Soyuz FG configuration was first launched in 2001 and after several flights it took over responsibility for launching human missions on Soyuz TMA spacecraft in late 2002. The older Soyuz U configuration is being phased out but is still in use for lower weight and lower priority spacecraft. The schedule for phase out is unknown and could take several years. The Molniya M will also eventually be replaced by the new Soyuz configurations, which use the Fregat stage to perform high-energy missions. The Soyuz 2 is anticipated to be commercially available in 2005, following Russian test flights in 2004. However, the schedule for development of the Soyuz 2 has been extended several times in the past so the current schedule may also slip.

Soyuz/Molniya launch rates have declined significantly from their peak of more than 60 per year around 1980 to 12 per year in the late 1990s. Still, this is an active schedule by the standards of any other launch vehicle. Engine production facilities can still produce enough engines for 25–30 flights per year, should the need (and sufficient funding) arise. Starsem could, in principle, perform one commercial launch every two weeks from Baikonur. Launch campaigns are roughly one month long, and two can be carried out simultaneously.

Performance

Soyuz and Molniya are launched from the Baikonur and Plesetsk cosmodromes (Molniya from Plesetsk only). Because both facilities are landlocked, specific downrange drop zones are reserved for the impact of the first and second stages and payload fairings. The vehicles are therefore constrained to launch azimuths that overfly these zones. Available inclinations are 52, 65, 70, and 95.4 deg from Baikonur, and 63, 67, 73, 82, and 90 deg and sun-synchronous orbits from Plesetsk. While the basic Soyuz U is not capable of yaw steering to reach other inclinations, the Fregat upper stage or the Soyuz 2 vehicles can reach other inclinations, including sun-synchronous orbits from Baikonur.

The basic three-stage Soyuz is capable of reaching circular orbits only at very low altitudes, because it lacks a third-stage restart capability. As a result, missions to space stations have not been delivered directly to the 407 km (220 nmi) reference orbit listed for other launch vehicles, but rather to lower altitude orbits. To achieve higher orbits, an upper stage is used: either the Block L standard in the Molniya configuration or the Fregat upper stage optional on Soyuz. Fregat can be delivered to a 200-km (108-nmi) parking orbit and perform two burns to reach the final destination, or it can be released from the third stage in a suborbital trajectory and perform an additional burn to reach parking orbit. Soyuz family performance varies depending on the detailed vehicle configuration and operational assumptions. Unless noted the following performance information assumes standard commercial practices such as a flight performance reserve providing 99.7% probability of sufficient propellant for orbit insertion, and fairing jettison times that limit aeroheating levels to no more than 1135 W/m^2 (0.1 BTU/ft^2/s). Russian government missions may use different fairing jettison times or performance reserves, resulting in higher performance. Performance graphs for the Soyuz 2 describe the capability of the Soyuz 2-1A with Fregat and a 4.1 m payload fairing. Performance of the Soyuz FG/Fregat is similar or even slightly higher because although it lacks the enhancements of the Soyuz 2 it has a smaller payload fairing with less weight and drag.

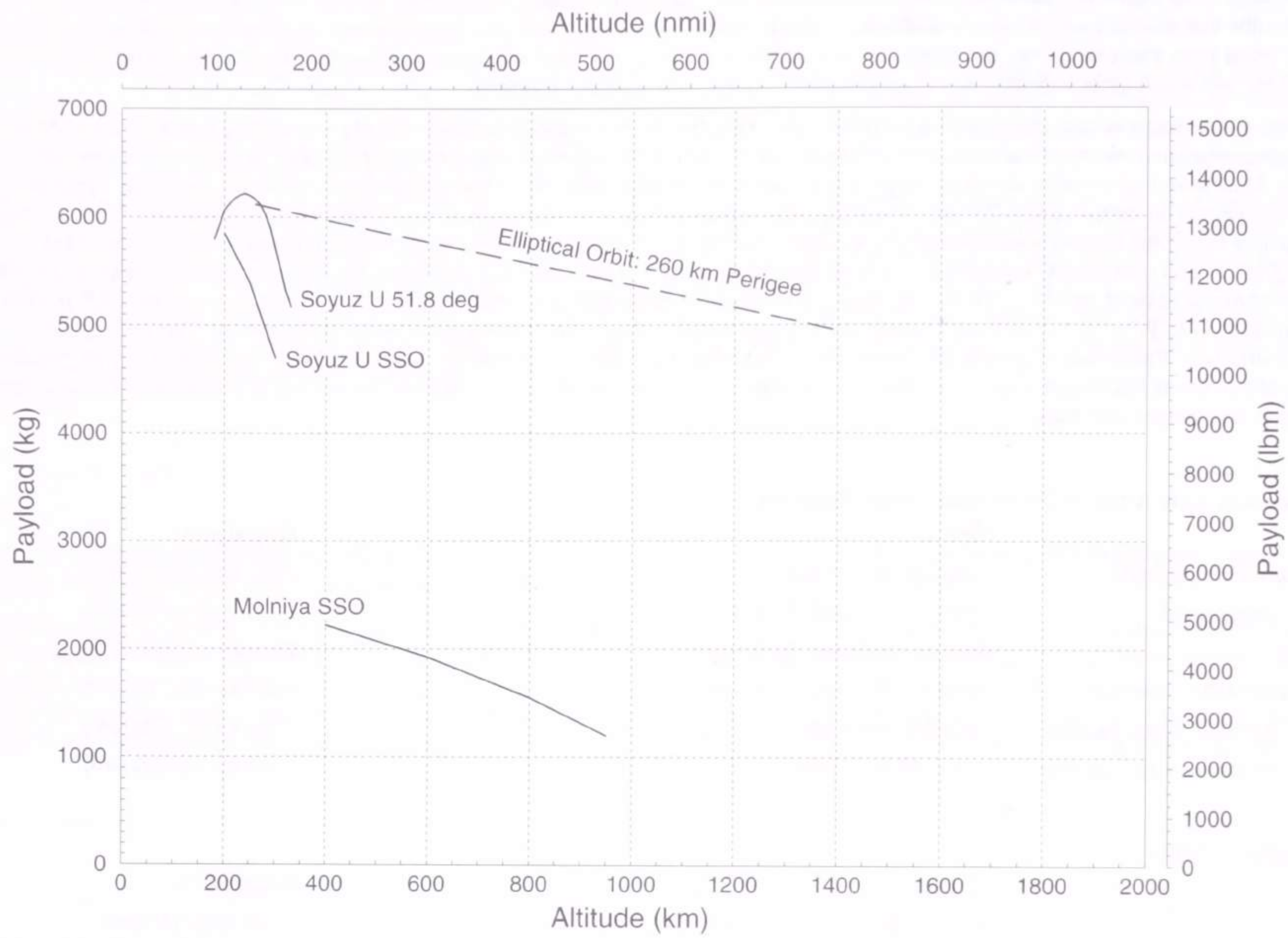
Space Station and Low Altitude Performance from Baikonur

Vehicle	Orbit	Capability
Soyuz U, maximum capability	Low altitude, 51.6 deg	7200 kg (15,875 lbm)
Soyuz FG / commercial	200 km (108 nmi), 51.8 deg	7000 kg (15,430 lbm)
Soyuz 2-1B	200 km (108 nmi), 51.8 deg	7900 kg (17,400 lbm)
Soyuz U, commercial standard	240 km (130 nmi), 51.8 deg	6220 kg (13,710 lbm)
Soyuz FG / manned Soyuz capsule	202×240 km (109×130 nmi), 51.6 deg	7150 kg (15,750 lbm)
Soyuz FG / Progress cargo capsule	193×240 km (104×130 nmi), 51.6 deg	7420 kg (16,350 lbm)

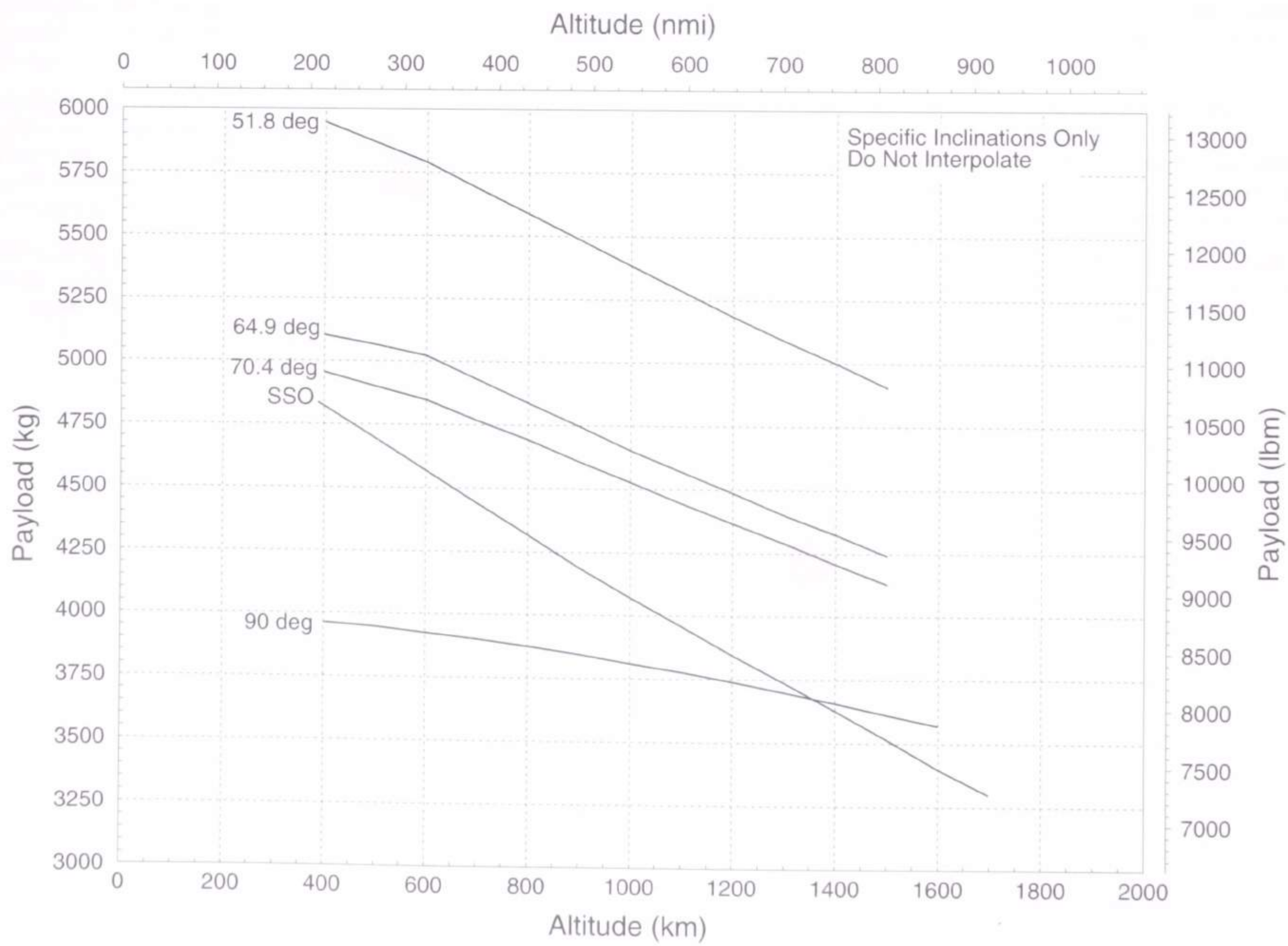
Highly Elliptical Orbits

Vehicle	Orbit	Capability
Molniya M	510×40,000 km (275×21,600 nmi), 62.8 deg	2000 kg (4400 lbm)
Molniya M	800×200,000 km (432×107,990 nmi), 62.8 deg	1250 kg (2755 lbm)

PERFORMANCE

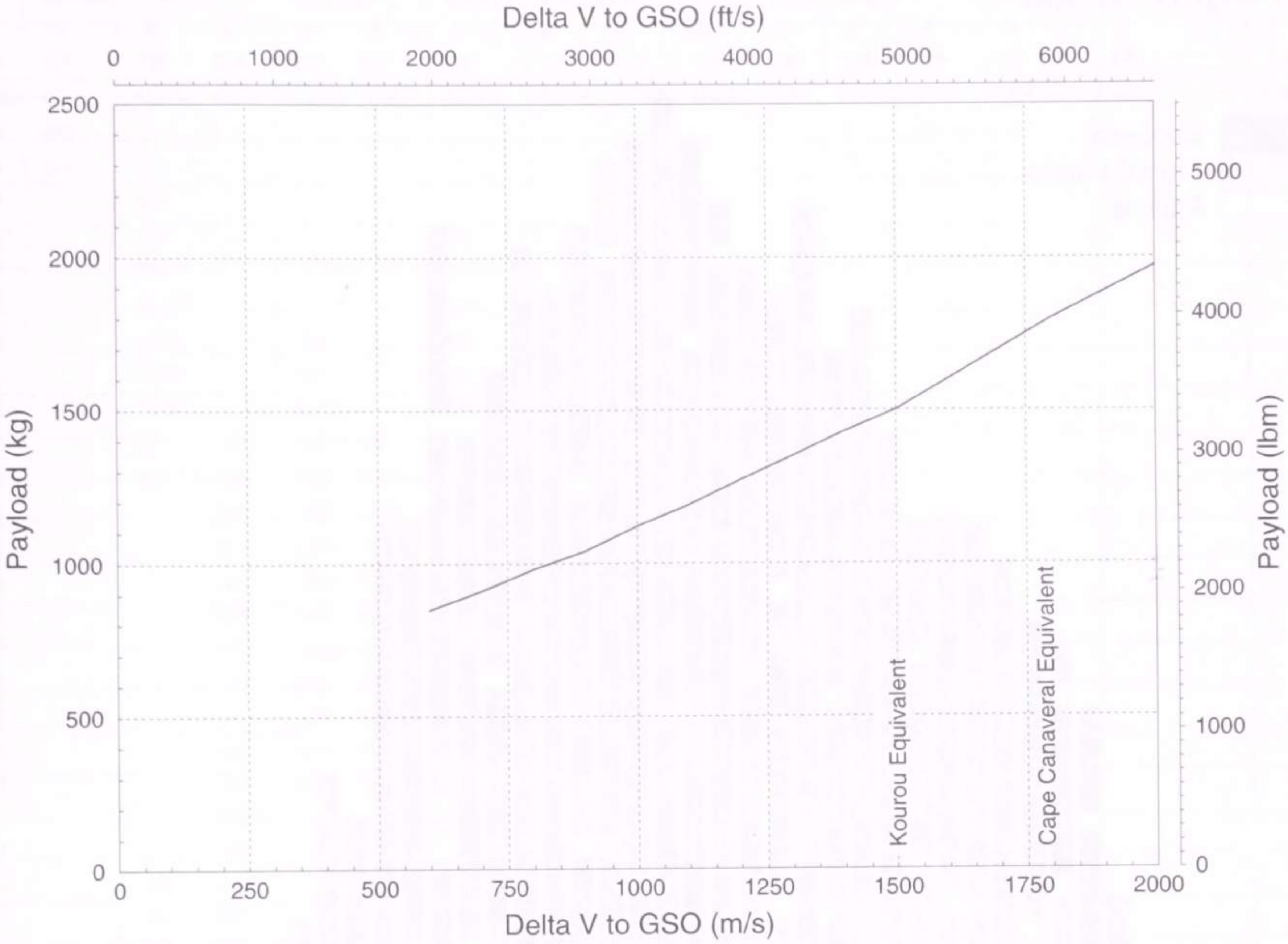


Soyuz U and Molniya LEO Performance

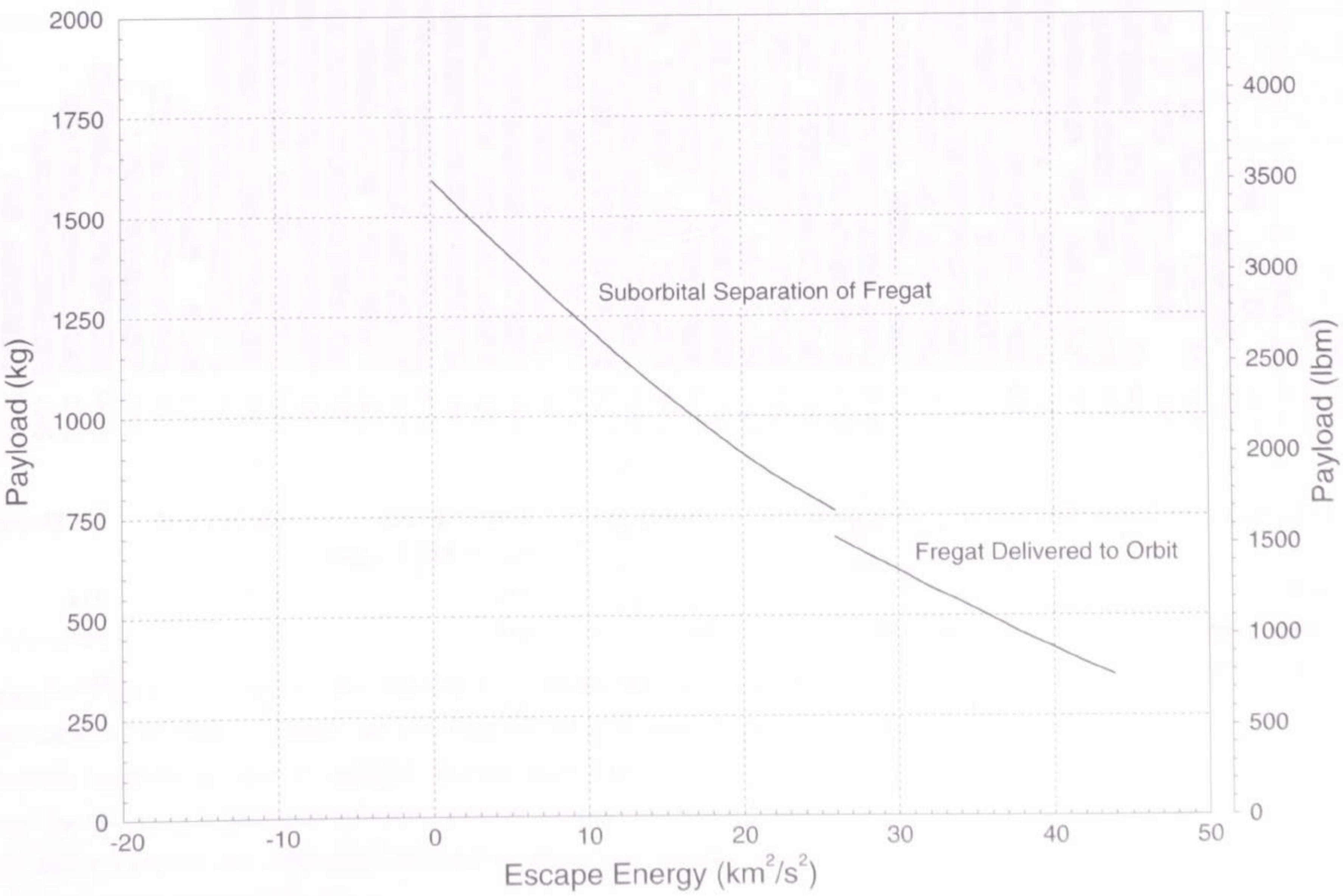


Soyuz 2/Fregat LEO Performance

Performance



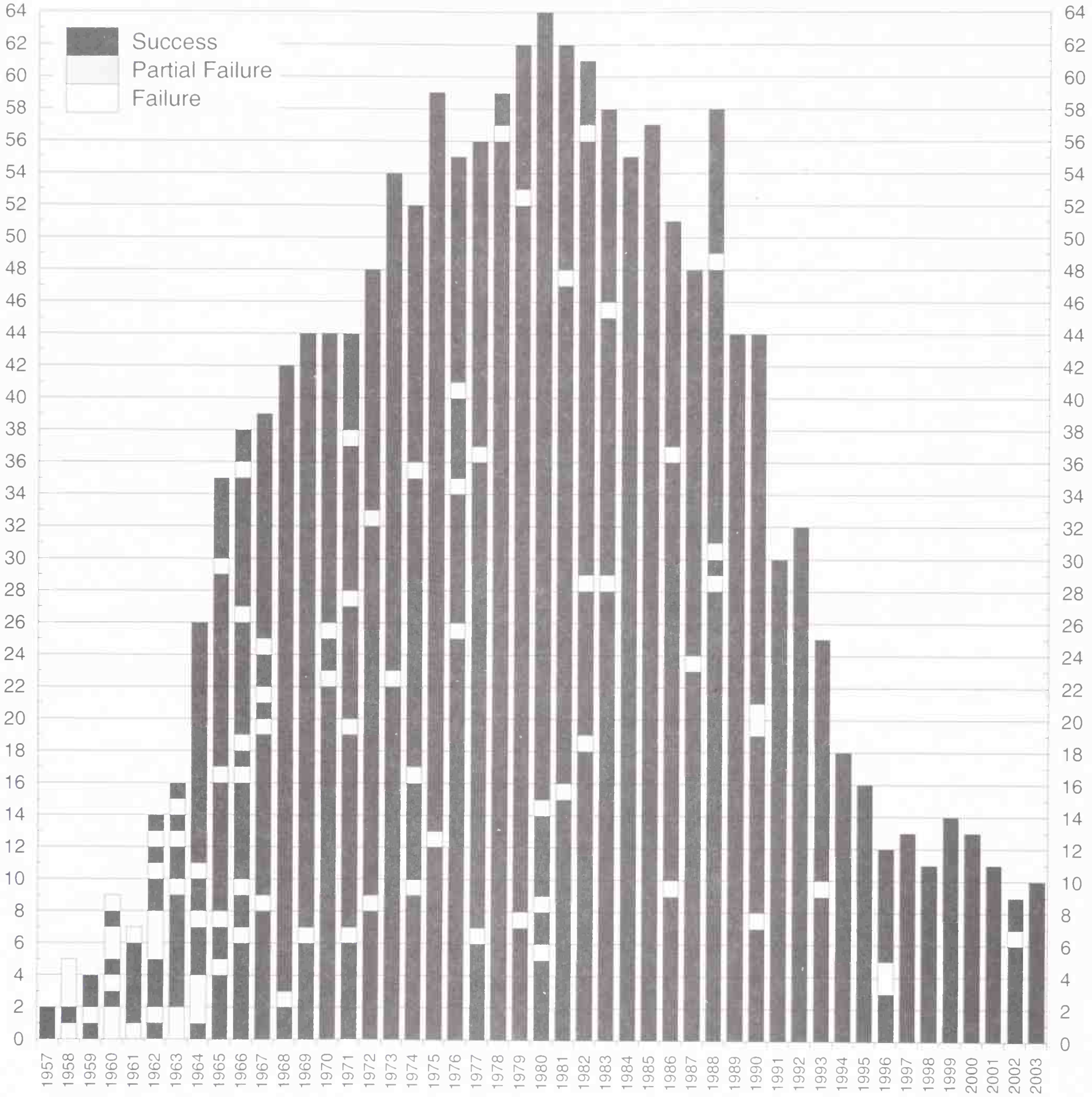
Soyuz 2/Fregat GTO Performance



Soyuz 2 Performance for Earth Escape Missions

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Soyuz U	Soyuz FG	Soyuz Series (all 11A511 types)	Molniya-M	All Molniya	Family Total
Total Orbital Flights	704	9	840	290	314	1630
Launch Vehicle Successes	683	9	816	271	281	1541
Launch Vehicle Partial Failures	0	0	0	0	0	0
Launch Vehicle Failures	21	0	24	19	33	89

Flight History

Year	Total		Sputnik	Vostok		Voskhod		Molniya	Molniya M		Soyuz, Other		Soyuz U		Soyuz U2	Soyuz U/Fregat	Soyuz FG	Human	
			B	*B*	*P*	*B*	*P*	*B*	*B*	*P*	*B*	*P*	*B*	*P*	*B*	*B*	*B*	*B*	
	T	F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T/F	T	F
Total	1630	89	4/1	80/14	86/3	138/4	168/11	24/14	67/11	223/8	35/2	8/0	275/10	425/11	84/0	4/0	9/0	96	2
1957	2	0	2/0																
1958	5	4	2/1	3/3															
1959	4	1		4/1															
1960	9	6		7/4				2/2											
1961	7	2		5/1				2/1										2	0
1962	14	6		8/1				6/5										2	0
1963	16	5		11/2		1/0		4/3										2	0
1964	26	5		13/0		5/0		3/1	5/4									1	0
1965	35	4		10/1		12/0		7/2	5/1		1/0							1	0
1966	38	6		9/1	3/0	10/0	4/2		9/2		3/1								
1967	39	4		2/0	6/0	7/2	13/2		7/0		4/0							1	0
1968	42	1			2/0	13/0	16/0		6/1		5/0							1	0
1969	44	1			3/1	13/0	19/0		4/0		5/0							5	0
1970	44	2			5/0	13/0	17/1		3/1	4/0	2/0							1	0
1971	44	4			5/0	14/1	17/3			3/0	4/0	1/0						2	0
1972	48	2			5/0	12/0	17/1		5/1	6/0	1/0	2/0							
1973	54	1			3/0	8/0	24/1		3/0	7/0	4/0	2/0		3/0				2	0
1974	52	3			6/0	10/1	17/1			7/0	3/0	2/0	4/0	3/1				3	0
1975	59	1			6/0	12/0	18/0		1/0	11/0	3/1		3/0	5/0				4	1
1976	55	3			5/0	8/0	6/0		3/0	8/2		1/0	9/0	15/1				3	0
1977	56	2			6/0				2/0	8/0			15/2	25/0				3	0
1978	59	1			5/1				1/0	8/0			18/0	27/0				5	0
1979	62	2		1/0	7/0					7/0			10/0	37/2				3	0
1980	64	3		1/0	6/1				1/0	11/2			13/0	32/0				6	0
1981	62	2		1/0	5/0				1/0	13/1			22/1	20/0				3	0
1982	61	3			5/0				2/1	9/0			24/1	21/1				3	0
1983	58	2		1/0	3/0				3/0	8/1			15/1	28/0				3	1
1984	55	0								11/0			15/0	28/0	1/0			3	0
1985	57	0		1/0					2/0	14/0			11/0	26/0	3/0			2	0
1986	51	2								14/1			15/1	15/0	7/0			1	0
1987	48	1								4/0			11/0	21/1	12/0			3	0
1988	58	3		2/0					2/0	9/0			14/2	21/1	10/0			3	0
1989	44	0							1/0	5/0			6/0	24/0	8/0			1	0
1990	44	3								12/1			2/0	20/2	10/0			3	0
1991	30	0		1/0						5/0			2/0	13/0	9/0			2	0
1992	32	0								8/0			1/0	13/0	10/0			2	0
1993	25	1								8/0			3/1	7/0	7/0			2	0
1994	18	0								3/0			6/0	4/0	5/0			3	0
1995	16	0							1/0	3/0			6/0	4/0	2/0			2	0
1996	12	2								3/0			6/1	3/1				2	0
1997	13	0								3/0			7/0	3/0				2	0
1998	11	0								3/0			7/0	1/0				2	0
1999	14	0								2/0			9/0	3/0				1	0
2000	13	0											9/0			4/0		2	0
2001	11	0								2/0			6/0	1/0			2/0	2	0
2002	9	1								2/0			2/0	2/1			3/0	2	0
2003	10	0								2/0			4/0				4/0		

B = Baikonur; P = Plesetsk; T = Total Launch Attempts; F = Failures and Partial Failures

The far-right columns for human spaceflight are not added into the totals because they are already accounted for under the correct vehicle type.

Two flights of the 11A59 configuration in 1963 and 1964 are counted with Vostok.

Two flights of the 11A510 configuration in 1965 and 1966 are counted with "Soyuz, Other."

Six flights of Soyuz U with the Ikar upper stage occurred in 1999 and are counted under Soyuz U.

Many sources combine Voskhod with Soyuz.

The Molniya ML and 2BL are both variants of the Molniya-M.

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	753	1979 Jan 11	14	Soyuz U		P		1979 001A			Kosmos 1070	5500	LEO (62.8)	MIL	USSR
	754	Jan 13	2	Soyuz U		P		1979 002A			Kosmos 1071	6000	LEO (62.8)	MIL	USSR
	755	Jan 18	5	Molniya M		P	LC 43/3	1979 004A			Molniya 3-11	1750	EEO (64.1)	CIV	USSR
	756	Jan 25	7	Vostok M		B		1979 005A			Meteor-Priroda 2 (Meteor 1-29)	1100	SSO	CIV	USSR
	757	Jan 30	5	Soyuz U		P		1979 006A			Kosmos 1073	6000	LEO (62.8)	MIL	USSR
	758	Jan 31	1	Soyuz U		B		1979 008A			Kosmos 1074	7000	LEO (51.6)	MIL	USSR
	759	Feb 13	13	Vostok M		P		1979 012A			Kosmos 1077	2500	LEO (81.2)	MIL	USSR
F	**760**	**Feb 16**	**3**	**Soyuz U**		**P**	**LC 41**	1979 F01A			Kosmos			MIL	USSR
	761	Feb 22	6	Soyuz U		P		1979 016A			Kosmos 1078	6000	LEO (72.9)	MIL	USSR
	762	Feb 25	3	Soyuz U		B		1979 018A	D	2/0	Soyuz 32	6600	STA	CIV	USSR
	763	Feb 27	2	Soyuz U		P	LC 43/3	1979 019A			Kosmos 1079	6700	LEO (67.1)	MIL	USSR
	764	Mar 01	2	Vostok M		P		1979 021A			Meteor 2-4	1300	LEO (81.2)	CIV	USSR
	765	Mar 12	11	Soyuz U	Yo15000-162	B	LC 31	1979 022A	D		Progress 5	7020	STA	CIV	USSR
	766	Mar 14	2	Soyuz U		P		1979 023A			Kosmos 1080	6000	LEO (72.8)	MIL	USSR
	767	Mar 31	17	Soyuz U		P		1979 027A			Kosmos 1090	5500	LEO (72.9)	MIL	USSR
S	768	Apr 10	10	Soyuz U		B	LC 31	1979 029A	D	2/2	Soyuz 33	6000	STA	CIV	USSR
	769	Apr 12	2	Molniya M		P		1979 031A			Molniya 1-43	1600	EEO (64.9)	CIV	USSR
	770	Apr 14	2	Vostok M		P		1979 032A			Kosmos 1093	2500	LEO (81.2)	MIL	USSR
	771	Apr 20	6	Soyuz U		P		1979 034A			Kosmos 1095	6000	LEO (72.8)	MIL	USSR
	772	Apr 27	7	Soyuz U		P	LC 43/3	1979 037A			Kosmos 1097	6700	LEO (62.7)	MIL	USSR
	773	May 13	16	Soyuz U	Zh15000-175	B	LC 31	1979 039A	D		Progress 6	7020	STA	CIV	USSR
	774	May 15	2	Soyuz U		P		1979 040A			Kosmos 1098	6000	LEO (72.9)	MIL	USSR
	775	May 17	2	Soyuz U		P	LC 43/4	1979 041A			Kosmos 1099	5500	LEO (81.3)	MIL	USSR
	776	May 25	8	Soyuz U		P	LC 41/1	1979 043A			Kosmos 1102	5500	LEO (81.3)	MIL	USSR
	777	May 31	6	Soyuz U		P		1979 045A			Kosmos 1103	6000	LEO (62.7)	MIL	USSR
	778	Jun 05	5	Molniya M		P	LC 43/4	1979 048A			Molniya 3-12	1750	EEO (64.1)	CIV	USSR
	779	Jun 06	1	Soyuz U		B		1979 049A	D	0/2	Soyuz 34	6800	STA	CIV	USSR
	780	Jun 08	2	Soyuz U		P	LC 41/1	1979 052A			Kosmos 1105	5500	LEO (81.3)	MIL	USSR
	781	Jun 12	4	Soyuz U		P	LC 43/4	1979 054A			Kosmos 1106	5500	LEO (81.4)	MIL	USSR
	782	Jun 15	3	Soyuz U		P		1979 055A			Kosmos 1107	6000	LEO (72.9)	MIL	USSR
	783	Jun 22	7	Soyuz U		P	LC 43/4	1979 056A			Kosmos 1108	5500	LEO (81.3)	MIL	USSR
	784	Jun 27	5	Molniya M		P	LC 41/1	1979 058A			Kosmos 1109	1250	EEO (62.9)	MIL	USSR
	785	Jun 28	1	Soyuz U	Zh15000-192	B	LC 31	1979 059A	D		Progress 7	1020	STA	CIV	USSR
	786	Jun 29	1	Soyuz U		P		1979 061A			Kosmos 1111	6000	LEO (62.8)	MIL	USSR
	787	Jul 10	11	Soyuz U		B		1979 064A			Kosmos 1113	6000	LEO (65)	MIL	USSR
	788	Jul 13	3	Soyuz U		P	LC43/4	1979 066A			Kosmos 1115	5500	LEO (81.3)	MIL	USSR
	789	Jul 20	7	Vostok M		P		1979 067A			Kosmos 1116	2500	LEO (81.2)	MIL	USSR
	790	Jul 25	5	Soyuz U		P		1979 068A			Kosmos 1117	6000	LEO (62.8)	MIL	USSR
	791	Jul 27	2	Soyuz U		P	LC 43/4	1979 069A			Kosmos 1118	5500	LEO (81.3)	MIL	USSR
	792	Jul 31	4	Molniya M		P		1979 070A			Molniya 1-44	1600	EEO (64.2)	CIV	USSR
	793	Aug 03	3	Soyuz U		P		1979 071A			Kosmos 1119	5500	LEO (81.3)	MIL	USSR
	794	Aug 11	8	Soyuz U		B		1979 073A			Kosmos 1120	5500	LEO (70.4)	MIL	USSR
	795	Aug 14	3	Soyuz U		P	LC 43/3	1979 074A			Kosmos 1121	6700	LEO (67.2)	MIL	USSR
	796	Aug 17	3	Soyuz U		P	LC 43/4	1979 075A			Kosmos 1122	5500	LEO (81.3)	MIL	USSR
	797	Aug 21	4	Soyuz U		P	LC 41/1	1979 076A			Kosmos 1123	5500	LEO (81.3)	MIL	USSR
S	798	Aug 28	7	Molniya M		P	LC 43/4	1979 077A			Kosmos 1124	1250	EEO (68)	MIL	USSR
	799	Aug 31	3	Soyuz U		P		1979 079A			Kosmos 1126	6000	LEO (72.8)	MIL	USSR
	800	Sep 05	5	Soyuz U		P		1979 080A			Kosmos 1127	6000	LEO (81.3)	MIL	USSR
	801	Sep 14	9	Soyuz U		P		1979 081A			Kosmos 1128	6000	LEO (62.8)	MIL	USSR
	802	Sep 25	11	Soyuz U		P	LC 41/1	1979 083A			Kosmos 1129	6000	LEO (62.8)	MIL	USSR
	803	Sep 28	3	Soyuz U		P		1979 085A			Kosmos 1138	6000	LEO (72.8)	MIL	USSR
	804	Oct 05	7	Soyuz U		P		1979 088A			Kosmos 1139	5500	LEO (72.9)	MIL	USSR
F	**805**	**Oct 12**	**7**	**Soyuz U**		**P**	**LC 43**	1979 F04A			Kosmos			MIL	USSR
	806	Oct 20	8	Molniya M		P		1979 091A			Molniya 1-45	1600	EEO (64.7)	CIV	USSR
	807	Oct 22	2	Soyuz U		P		1979 092A			Kosmos 1142	6000	LEO (72.9)	MIL	USSR
	808	Oct 26	4	Vostok M		P		1979 093A			Kosmos 1143	2500	LEO (81.2)	MIL	USSR
	809	Oct 31	5	Vostok M		P		1979 095A			Meteor 2-5	1300	LEO (81.2)	CIV	USSR
	810	Nov 02	2	Soyuz U		P	LC 43/3	1979 097A			Kosmos 1144	6700	LEO (67.1)	MIL	USSR
	811	Nov 27	25	Vostok M		P		1979 099A			Kosmos 1145	2500	LEO (81.2)	MIL	USSR
	812	Dec 12	15	Soyuz U		P		1979 102A			Kosmos 1147	6000	LEO (72.9)	MIL	USSR
	813	Dec 16	4	Soyuz U		B		1979 103A	D		Soyuz T	7000	STA	CIV	USSR
	814	Dec 28	12	Soyuz U		P		1979 106A			Kosmos 1148	6000	LEO (67.1)	MIL	USSR
	815	1980 Jan 09	12	Soyuz U		P		1980 001A			Kosmos 1149	6300	LEO (72.9)	MIL	USSR
	816	Jan 11	2	Molniya M		P		1980 002A			Molniya 1-46	1600	EEO (64.1)	CIV	USSR
	817	Jan 24	13	Soyuz U		P	LC 43/3	1980 006A			Kosmos 1152	6700	LEO (67.2)	MIL	USSR
	818	Jan 30	6	Vostok M		P		1980 008A			Kosmos 1154	2500	LEO (81.2)	MIL	USSR
	819	Feb 07	8	Soyuz U		P		1980 009A			Kosmos 1155	6300	LEO (72.8)	MIL	USSR
F	**820**	**Feb 12**	**5**	**Molniya M**		**P**	**LC 43/4**	1980 013A			Kosmos 1164	1250	EEO (62.8)	MIL	USSR
	821	Feb 21	9	Soyuz U		P		1980 017A			Kosmos 1165	6300	LEO (72.8)	MIL	USSR

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	822	Mar 04	12	Soyuz U		P		1980 020A			Kosmos 1166	6300	LEO (72.8)	MIL	USSR
F	**823**	**Mar 18**	**14**	**Vostok M**		**P**	**LC 43**				Kosmos			MIL	USSR
	824	Mar 27	9	Soyuz U	Zh15000-200	B	LC 31	1980 024A	D		Progress 8	7020	STA	CIV	USSR
	825	Apr 01	5	Soyuz U		B		1980 025A			Kosmos 1170	6300	LEO (70.3)	MIL	USSR
	826	Apr 09	8	Soyuz U		B		1980 027A	D	2/2	Soyuz 35	6570	STA	CIV	USSR
	827	Apr 12	3	Molniya M		P	LC41/1	1980 028A			Kosmos 1172	1250	EEO (66.2)	MIL	USSR
	828	Apr 17	5	Soyuz U		B		1980 029A			Kosmos 1173	6300	LEO (70.3)	MIL	USSR
F	**829**	**Apr 18**	**1**	**Molniya M**		**P**	**LC41/1**	1980 031A			Kosmos 1175	1900	EEO (62.8)	MIL	USSR
	830	Apr 27	9	Soyuz U	Zh15000-210	B	LC 1	1980 033A	D		Progress 9	7020	STA	CIV	USSR
	831	Apr 29	2	Soyuz U		P	LC 43/3	1980 035A			Kosmos 1177	6700	LEO (67.1)	MIL	USSR
	832	May 07	8	Soyuz U		P		1980 036A			Kosmos 1178	6300	LEO (72.8)	MIL	USSR
	833	May 15	8	Soyuz U		P		1980 038A			Kosmos 1180	5900	LEO (62.8)	MIL	USSR
	834	May 23	8	Soyuz U		P	LC 43/3	1980 040A			Kosmos 1182	7420	LEO (82.3)	MIL	USSR
	835	May 26	3	Soyuz U		B	LC 31	1980 041A	D	2/2	Soyuz 36	6570	STA	CIV	USSR
	836	May 28	2	Soyuz U		P		1980 042A			Kosmos 1183	6300	LEO (72)	MIL	USSR
	837	Jun 04	7	Vostok M		P		1980 044A			Kosmos 1184	2500	LEO (81.2)	MIL	USSR
	838	Jun 05	1	Soyuz U		B		1980 045A	D	2/2	Soyuz T-2	7000	STA	CIV	USSR
	839	Jun 06	1	Soyuz U		P	LC 41	1980 046A			Kosmos 1185	7420	LEO (82.4)	MIL	USSR
	840	Jun 12	6	Soyuz U		P		1980 048A			Kosmos 1187	6300	LEO (72.8)	MIL	USSR
	841	Jun 14	2	Molniya M		P	LC 43/3	1980 050A			Kosmos 1188	1250	EEO (67.5)	MIL	USSR
	842	Jun 18	4	Vostok M		B		1980 051A			Meteor 1-30	1100	LEO (97.7)	CIV	USSR
	843	Jun 21	3	Molniya M		P		1980 053A			Molniya 1-47	1600	EEO (64.8)	CIV	USSR
	844	Jun 26	5	Soyuz U		P		1980 054A			Kosmos 1189	6300	LEO (72.8)	MIL	USSR
	845	Jun 29	3	Soyuz U	P15000-232	B	LC 1	1980 055A	D		Progress 10	7020	STA	CIV	USSR
	846	Jul 02	3	Molniya M		P	LC 41/1	1980 057A			Kosmos 1191	1250	EEO (62.6)	MIL	USSR
	847	Jul 09	7	Soyuz U		P		1980 059A			Kosmos 1200	6300	LEO (72.8)	MIL	USSR
	848	Jul 15	6	Soyuz U		P	LC 43/3	1980 061A			Kosmos 1201	5700	LEO (82.3)	MIL	USSR
	849	Jul 18	3	Molniya M		P	LC 43/3	1980 063A			Molniya 3-13	1750	EEO (63.8)	CIV	USSR
	850	Jul 23	5	Soyuz U		B	LC 1	1980 064A	D	2/2	Soyuz 37	6570	STA	CIV	USSR
	851	Jul 24	1	Soyuz U		P		1980 065A			Kosmos 1202	6300	LEO (72.8)	MIL	USSR
	852	Jul 31	7	Soyuz U		P	LC 43/3	1980 066A			Kosmos 1203	7420	LEO (82.3)	MIL	USSR
	853	Aug 12	12	Soyuz U		P		1980 068A			Kosmos 1205	6300	LEO (72.8)	MIL	USSR
	854	Aug 15	3	Vostok M		P		1980 069A			Kosmos 1206	2500	LEO (81.2)	MIL	USSR
	855	Aug 22	7	Soyuz U		P	LC 41/1	1980 070A			Kosmos 1207	7200	LEO (82.3)	MIL	USSR
	856	Aug 26	4	Soyuz U		P	LC 41/1	1980 071A			Kosmos 1208	6700	LEO (67.1)	MIL	USSR
	857	Sep 03	8	Soyuz U		P	LC 41	1980 072A			Kosmos 1209	6300	LEO (82.3)	MIL	USSR
	858	Sep 09	6	Vostok M		P		1980 073A			Meteor 2-6	1300	LEO (81.2)	CIV	USSR
	859	Sep 18	9	Soyuz U		B	LC 1	1980 075A	D	2/2	Soyuz 38	6570	STA	CIV	USSR
	860	Sep 19	1	Soyuz U		P		1980 076A			Kosmos 1210	7420	LEO (82.2)	MIL	USSR
	861	Sep 23	4	Soyuz U		P		1980 077A			Kosmos 1211	7220	LEO (82.3)	MIL	USSR
	862	Sep 26	3	Soyuz U		P	LC 41/1	1980 078A			Kosmos 1212	7220	LEO (82.3)	MIL	USSR
	863	Sep 28	2	Soyuz U	P15000-219	B	LC 1	1980 079A	D		Progress 11	7020	STA	CIV	USSR
	864	Oct 03	5	Soyuz U		P		1980 080A			Kosmos 1213	6300	LEO (72.9)	MIL	USSR
	865	Oct 10	7	Soyuz U		P		1980 082A			Kosmos 1214	5700	LEO (67.1)	MIL	USSR
	866	Oct 16	6	Soyuz U		P		1980 084A			Kosmos 1216	6300	LEO (72.9)	MIL	USSR
	867	Oct 24	8	Molniya M		P	LC 41/1	1980 085A			Kosmos 1217	1250	EEO (67.2)	MIL	USSR
	868	Oct 30	6	Soyuz U		B		1980 086A			Kosmos 1218	6700	LEO (64.9)	MIL	USSR
	869	Oct 31	1	Soyuz U		P		1980 088A			Kosmos 1219	6300	LEO (72.8)	MIL	USSR
	870	Nov 12	12	Soyuz U		P		1980 090A			Kosmos 1221	6300	LEO (72.9)	MIL	USSR
	871	Nov 16	4	Molniya M		P		1980 092A			Molniya 1-48	1600	EEO (64.6)	CIV	USSR
	872	Nov 21	5	Vostok M		P		1980 093A			Kosmos 1222	2500	LEO (81.2)	MIL	USSR
	873	Nov 27	6	Soyuz U		B		1980 094A	D	3/3	Soyuz T-3	7000	STA	CIV	USSR
	874	Nov 27	0	Molniya M		P	LC 41/1	1980 095A			Kosmos 1223	1250	EEO (65.6)	MIL	USSR
	875	Dec 01	4	Soyuz U		P		1980 096A			Kosmos 1224	6300	LEO (72.9)	MIL	USSR
	876	Dec 16	15	Soyuz U		P		1980 101A			Kosmos 1227	6300	LEO (72.9)	MIL	USSR
	877	Dec 25	9	Molniya SOL		B		1980 103A			Prognoz 8	910	EEO (65.8)	CIV	USSR
	878	Dec 26	1	Soyuz U		P	LC 41/1	1980 105A			Kosmos 1236	6700	LEO (67.1)	MIL	USSR
	879	1981 Jan 06	11	Soyuz U		P		1981 001A			Kosmos 1237	6300	LEO (72.9)	MIL	USSR
	880	Jan 09	3	Molniya M		P	LC 41/1	1981 002A			Molniya 3-14	1750	EEO (64.2)	CIV	USSR
	881	Jan 16	7	Soyuz U		P		1981 004A			Kosmos 1239	5700	LEO (82.3)	MIL	USSR
	882	Jan 20	4	Soyuz U		B		1981 005A			Kosmos 1240	6700	LEO (64.9)	MIL	USSR
	883	Jan 24	4	Soyuz U	P15000-235	B	LC 1	1981 007A	D		Progress 12	7020	STA	CIV	USSR
	884	Jan 27	3	Vostok M		P		1981 008A			Kosmos 1242	2500	LEO (81.2)	MIL	USSR
	885	Jan 30	3	Molniya M		P		1981 009A			Molniya 1-49	1600	EEO (64.1)	CIV	USSR
	886	Feb 13	14	Soyuz U		P		1981 014A			Kosmos 1245	6300	LEO (72.8)	MIL	USSR
	887	Feb 18	5	Soyuz U		B		1981 015A			Kosmos 1246	6700	LEO (64.9)	MIL	USSR
	888	Feb 19	1	Molniya M		P	LC 16/2	1981 016A			Kosmos 1247	1250	EEO (67.5)	MIL	USSR
	889	Mar 05	14	Soyuz U		P	LC 41/1	1981 020A			Kosmos 1248	6700	LEO (67.1)	MIL	USSR
	890	Mar 12	7	Soyuz U		B		1981 023A	D	2/2	Soyuz T-4	7000	STA	CIV	USSR
	891	Mar 17	5	Soyuz U		B		1981 026A			Kosmos 1259	6300	LEO (70.3)	MIL	USSR

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation	Crew	Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	892	Mar 22	5	Soyuz U		B	LC 31	1981 029A	D	2/2	Soyuz 39	6800	STA	CIV	USSR
	893	Mar 24	2	Molniya M		P	LC 41/1	1981 030A			Molniya 3-15	1750	EEO (64.9)	CIV	USSR
F	**894**	**Mar 28**	**4**	**Soyuz U**		**B**		1981 F02A			Kosmos			MIL	USSR
	895	Mar 31	3	Molniya M		P	LC 41/1	1981 031A			Kosmos 1261	1250	EEO (66.9)	MIL	USSR
	896	Apr 07	7	Soyuz U		P		1981 032A			Kosmos 1262	6300	LEO (72.8)	MIL	USSR
	897	Apr 15	8	Soyuz U		B		1981 035A			Kosmos 1264	6300	LEO (70.3)	MIL	USSR
	898	Apr 16	1	Soyuz U		P		1981 036A			Kosmos 1265	6300	LEO (72.8)	MIL	USSR
	899	Apr 28	12	Soyuz U		B		1981 040A			Kosmos 1268	6300	LEO (70.3)	MIL	USSR
	900	May 14	16	Soyuz U		B	LC 1	1981 042A	D	2/2	Soyuz 40	6800	STA	CIV	USSR
	901	May 14	0	Vostok M		P		1981 043A			Meteor 2-7	1300	LEO (81.3)	CIV	USSR
	902	May 18	4	Soyuz U		B		1981 045A			Kosmos 1270	6700	LEO (64.8)	MIL	USSR
	903	May 19	1	Vostok M		P		1981 046A			Kosmos 1271	2500	LEO (81.2)	MIL	USSR
	904	May 21	2	Soyuz U		B		1981 047A			Kosmos 1272	6300	LEO (70.3)	MIL	USSR
	905	May 22	1	Soyuz U		P	LC 41/1	1981 048A			Kosmos 1273	5900	LEO (82.3)	MIL	USSR
	906	Jun 03	12	Soyuz U		P	LC 41/1	1981 052A			Kosmos 1274	6700	LEO (67.1)	MIL	USSR
	907	Jun 09	6	Molniya M		P	LC 41/1	1981 054A			Molniya 3-16	1750	EEO (64.2)	CIV	USSR
	908	Jun 16	7	Soyuz U		P	LC 43/3	1981 055A			Kosmos 1276	5900	LEO (82.3)	MIL	USSR
	909	Jun 17	1	Soyuz U		B		1981 056A			Kosmos 1277	6300	LEO (70.4)	MIL	USSR
	910	Jun 19	2	Molniya M		P		1981 058A			Kosmos 1278	1250	EEO (67.1)	MIL	USSR
	911	Jun 24	5	Molniya M		P		1981 060A			Molniya 1-50	1600	EEO (64.8)	CIV	USSR
	912	Jul 01	7	Soyuz U		B		1981 062A			Kosmos 1279	6300	LEO (70.4)	MIL	USSR
	913	Jul 02	1	Soyuz U		P	LC 43/3	1981 063A			Kosmos 1280	6300	LEO (82.3)	MIL	USSR
	914	Jul 07	5	Soyuz U		P		1981 064A			Kosmos 1281	6300	LEO (72.8)	MIL	USSR
	915	Jul 10	3	Vostok M		B		1981 065A			Meteor-Priroda	1100	LEO (97.7)	CIV	USSR
	916	Jul 15	5	Soyuz U		B		1981 066A			Kosmos 1282	6700	LEO (64.9)	MIL	USSR
	917	Jul 17	2	Soyuz U		P		1981 067A			Kosmos 1283	6300	LEO (82.3)	MIL	USSR
	918	Jul 29	12	Soyuz U		P		1981 068A			Kosmos 1284	6300	LEO (82.3)	MIL	USSR
	919	Aug 04	6	Molniya M		P	LC 16/2	1981 071A			Kosmos 1285	1250	EEO (66.9)	MIL	USSR
	920	Aug 07	3	Vostok M		P	LC 43/3	1981 075A			Bulgaria 1300 (Interkosmos 22)	1500	LEO (81.2)	CIV	USSR
	921	Aug 13	6	Soyuz U		P	LC 41/1	1981 078A			Kosmos 1296	6700	LEO (67.1)	MIL	USSR
	922	Aug 18	5	Soyuz U		P		1981 079A			Kosmos 1297	6300	LEO (72.8)	MIL	USSR
	923	Aug 21	3	Soyuz U		B		1981 080A			Kosmos 1298	6700	LEO (64.9)	MIL	USSR
	924	Aug 27	6	Soyuz U		P	LC 41	1981 083A			Kosmos 1301	6300	LEO (82.3)	MIL	USSR
	925	Sep 04	8	Soyuz U		B		1981 086A			Kosmos 1303	6300	LEO (70.4)	MIL	USSR
F	**926**	**Sep 11**	**7**	**Molniya M**		**P**	**LC 43/3**	1981 088A			Kosmos 1305	1600	EEO (62.9)	MIL	USSR
	927	Sep 15	4	Soyuz U		P		1981 090A			Kosmos 1307	6300	LEO (72.8)	MIL	USSR
	928	Sep 18	3	Soyuz U		P		1981 092A			Kosmos 1309	5700	LEO (82.3)	MIL	USSR
	929	Oct 01	13	Soyuz U		B		1981 099A			Kosmos 1313	6300	LEO (70.3)	MIL	USSR
	930	Oct 09	8	Soyuz U		P	LC 41/1	1981 101A			Kosmos 1314	6300	LEO (82.3)	MIL	USSR
	931	Oct 13	4	Vostok M		P		1981 103A			Kosmos 1315	3800	LEO (81.2)	MIL	USSR
	932	Oct 15	2	Soyuz U		B		1981 104A			Kosmos 1316	6300	LEO (70.3)	MIL	USSR
	933	Oct 17	2	Molniya M		P	LC 41/1	1981 105A			Molniya 3-17	1750	EEO (64.5)	CIV	USSR
	934	Oct 31	14	Molniya M		P		1981 108A			Kosmos 1317	1250	EEO (64.4)	MIL	USSR
	935	Nov 03	3	Soyuz U		P	LC 41/1	1981 109A			Kosmos 1318	1900	LEO (67.1)	MIL	USSR
	936	Nov 13	10	Soyuz U		B		1981 112A			Kosmos 1319	6300	LEO (70.3)	MIL	USSR
	937	Nov 17	4	Molniya M		P		1981 113A			Molniya 1-51	1600	EEO (64.1)	CIV	USSR
	938	Dec 04	17	Soyuz U		B		1981 118A			Kosmos 1329	6300	LEO (65)	MIL	USSR
	939	Dec 19	15	Soyuz U		B		1981 121A			Kosmos 1330	6700	LEO (70.4)	MIL	USSR
	940	Dec 23	4	Molniya M		B		1981 123A			Molniya 1-52	1600	EEO (63.4)	CIV	USSR
	941	1982 Jan 12	20	Soyuz U		P		1982 002A			Kosmos 1332	5700	LEO (82.3)	MIL	USSR
	942	Jan 20	8	Soyuz U		P		1982 005A			Kosmos 1334	6300	LEO (72.8)	MIL	USSR
	943	Jan 30	10	Soyuz U		B		1982 008A			Kosmos 1336	6700	LEO (70.3)	MIL	USSR
	944	Feb 16	17	Soyuz U		P		1982 011A			Kosmos 1338	6000	LEO (72.8)	MIL	USSR
	945	Feb 19	3	Vostok M		P		1982 013A			Kosmos 1340	2500	LEO (81.2)	MIL	USSR
	946	Feb 26	7	Molniya M		P		1982 015A			Molniya 1-53	1600	EEO (63.8)	CIV	USSR
	947	Mar 03	5	Molniya M		P	LC 16/2	1982 016A			Kosmos 1341	1250	EEO (66.8)	MIL	USSR
	948	Mar 05	2	Soyuz U		P		1982 018A			Kosmos 1342	6300	LEO (72.8)	MIL	USSR
	949	Mar 17	12	Soyuz U		P		1982 021A			Kosmos 1343	6300	LEO (72.8)	MIL	USSR
	950	Mar 24	7	Molniya M		P	LC 41/1	1982 023A			Molniya 3-18	1750	EEO (64.9)	CIV	USSR
	951	Mar 31	7	Vostok M		P		1982 027A			Kosmos 1346	2500	LEO (81.2)	MIL	USSR
	952	Apr 02	2	Soyuz U		B		1982 028A			Kosmos 1347	6700	LEO (70.3)	MIL	USSR
	953	Apr 07	5	Molniya M		P	LC 16/2	1982 029A			Kosmos 1348	1250	EEO (63.1)	MIL	USSR
	954	Apr 15	8	Soyuz U		P	LC 41/1	1982 032A			Kosmos 1350	6700	LEO (67.1)	MIL	USSR
	955	Apr 21	6	Soyuz U		B		1982 035A			Kosmos 1352	6300	LEO (70.3)	MIL	USSR
	956	Apr 23	2	Soyuz U		P	LC 41/1	1982 036A			Kosmos 1353	5900	LEO (82.3)	MIL	USSR
	957	May 05	12	Vostok M		P		1982 039A			Kosmos 1356	2500	LEO (81.2)	MIL	USSR
	958	May 13	8	Soyuz U		B		1982 042A	D	2/3	Soyuz T-5	6850	STA	CIV	USSR
F	**959**	**May 15**	**2**	**Soyuz U**		**P**	**LC 41**	1982 F02A			Kosmos			MIL	USSR
	960	May 20	5	Molniya M		P	LC 41/1	1982 045A			Kosmos 1367	1250	EEO (63.5)	MIL	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	961	May 21	1	Soyuz U		B		1982 046A			Kosmos 1368	6300	LEO (70.3)	MIL	USSR
	962	May 23	2	Soyuz U	Ts15000-283	B	LC 1	1982 047A	D		Progress 13	7020	STA	CIV	USSR
	963	May 25	2	Soyuz U		P	LC 43/3	1982 048A			Kosmos 1369	6300	LEO (82.3)	MIL	USSR
	964	May 28	3	Soyuz U		B		1982 049A			Kosmos 1370	6700	LEO (64.8)	MIL	USSR
	965	May 28	0	Molniya M		P		1982 050A			Molniya 1-54	1600	EEO (64.9)	CIV	USSR
	966	Jun 02	5	Soyuz U		B		1982 053A			Kosmos 1373	6300	LEO (70.4)	MIL	USSR
	967	Jun 08	6	Soyuz U		P	LC 43/3	1982 056A			Kosmos 1376	6300	LEO (82.3)	MIL	USSR
	968	Jun 08	0	Soyuz U		B		1982 057A			Kosmos 1377	6700	LEO (64.9)	MIL	USSR
F	**969**	**Jun 12**	**4**	**Soyuz U**		**B**		1982 F03A			Kosmos			MIL	USSR
	970	Jun 18	6	Soyuz U		B		1982 062A			Kosmos 1381	6300	LEO (70.3)	MIL	USSR
	971	Jun 24	6	Soyuz U		B	LC 1	1982 063A	D	3/3	Soyuz T-6	6850	STA	CIV	USSR
	972	Jun 25	1	Molniya M		P	LC 43/3	1982 064A			Kosmos 1382	1250	EEO (64.5)	MIL	USSR
	973	Jun 30	5	Soyuz U		P	LC 41/1	1982 067A			Kosmos 1384	6700	LEO (67.1)	MIL	USSR
	974	Jul 06	6	Soyuz U		P		1982 068A			Kosmos 1385	6300	LEO (82.3)	MIL	USSR
	975	Jul 10	4	Soyuz U	Shch15000-318	B	LC 1	1982 070A	D		Progress 14	7020	STA	CIV	USSR
	976	Jul 13	3	Soyuz U		P	LC 43/3	1982 071A			Kosmos 1387	5900	LEO (82.3)	MIL	USSR
	977	Jul 21	8	Molniya M		B		1982 074A			Molniya 1-55	1600	EEO (64.9)	CIV	USSR
	978	Jul 27	6	Soyuz U		P		1982 075A			Kosmos 1396	6300	LEO (72.8)	MIL	USSR
	979	Aug 03	7	Soyuz U		P		1982 077A			Kosmos 1398	5900	LEO (82.3)	MIL	USSR
	980	Aug 04	1	Soyuz U		B		1982 078A			Kosmos 1399	6700	LEO (64.9)	MIL	USSR
	981	Aug 05	1	Vostok M		P		1982 079A			Kosmos 1400	2500	LEO (81.2)	MIL	USSR
	982	Aug 19	14	Soyuz U		B		1982 080A	D	3/2	Soyuz T-7	6850	STA	CIV	USSR
	983	Aug 20	1	Soyuz U		P	LC 41	1982 081A			Kosmos 1401	6300	LEO (82.3)	MIL	USSR
	984	Aug 27	7	Molniya M		P	LC 41/1	1982 083A			Molniya 3-19	1750	EEO (63.5)	CIV	USSR
	985	Sep 01	5	Soyuz U		B		1982 085A			Kosmos 1403	6300	LEO (70.3)	MIL	USSR
	986	Sep 01	0	Soyuz U		P		1982 086A			Kosmos 1404	6300	LEO (72.8)	MIL	USSR
	987	Sep 08	7	Soyuz U		P	LC 43/1	1982 089A			Kosmos 1406	5900	LEO (82.3)	MIL	USSR
	988	Sep 15	7	Soyuz U		P	LC 41/1	1982 091A			Kosmos 1407	6700	LEO (67.1)	MIL	USSR
	989	Sep 18	3	Soyuz U	Ts15000-292	B	LC 1	1982 094A	D		Progress 15	7020	STA	CIV	USSR
	990	Sep 22	4	Molniya M		P	LC 16/2	1982 095A			Kosmos 1409	1250	EEO (64.3)	MIL	USSR
	991	Sep 30	8	Soyuz U		P		1982 098A			Kosmos 1411	6300	LEO (72.8)	MIL	USSR
	992	Oct 14	14	Soyuz U		B		1982 101A			Kosmos 1416	6300	LEO (70.3)	MIL	USSR
	993	Oct 31	17	Soyuz U	Shch15000-335	B	LC 1	1982 107A	D		Progress 16	7020	STA	CIV	USSR
	994	Nov 02	2	Soyuz U		B		1982 108A			Kosmos 1419	6300	LEO (70.3)	MIL	USSR
	995	Nov 18	16	Soyuz U		B		1982 112A			Kosmos 1421	6300	LEO (70.3)	MIL	USSR
	996	Dec 03	15	Soyuz U		P		1982 114A			Kosmos 1422	6300	LEO (82.3)	MIL	USSR
F	**997**	**Dec 08**	**5**	**Molniya M**		**B**		1982 115A			Kosmos 1423	1600	EEO (62.8)	MIL	USSR
	998	Dec 14	6	Vostok M		P		1982 116A			Meteor 2-9	1300	LEO (81.2)	CIV	USSR
	999	Dec 16	2	Soyuz U		B		1982 117A			Kosmos 1424	6700	LEO (64.9)	MIL	USSR
	1000	Dec 23	7	Soyuz U		B		1982 119A			Kosmos 1425	6300	LEO (69.9)	MIL	USSR
	1001	Dec 28	5	Soyuz U		B		1982 120A			Kosmos 1426	6700	LEO (50.5)	MIL	USSR
	1002	1983 Jan 20	23	Vostok M		P		1983 003A			Kosmos 1437	2500	LEO (81.1)	MIL	USSR
	1003	Jan 27	7	Soyuz U		B		1983 005A			Kosmos 1438	6300	LEO (70.4)	MIL	USSR
	1004	Feb 06	10	Soyuz U		B		1983 007A			Kosmos 1439	6300	LEO (70.3)	MIL	USSR
	1005	Feb 10	4	Soyuz U		P	LC 41	1983 009A			Kosmos 1440	6300	LEO (82.3)	MIL	USSR
	1006	Feb 16	6	Vostok M		P		1983 010A			Kosmos 1441	2500	LEO (81.1)	MIL	USSR
	1007	Feb 25	9	Soyuz U		P	LC 41/1	1983 012A			Kosmos 1442	2500	LEO (67.1)	MIL	USSR
	1008	Mar 02	5	Soyuz U		P		1983 014A			Kosmos 1444	6300	LEO (72.8)	MIL	USSR
	1009	Mar 11	9	Molniya ML		P	LC 41/1	1983 015A			Molniya 3-20	1750	EEO (63.3)	CIV	USSR
	1010	Mar 16	5	Soyuz U		B		1983 018A			Kosmos 1446	6300	LEO (69.9)	MIL	USSR
	1011	Mar 16	0	Molniya ML		P		1983 019A			Molniya 1-56	1600	EEO (63.6)	CIV	USSR
	1012	Mar 31	15	Soyuz U		P		1983 024A			Kosmos 1449	6300	LEO (72.8)	MIL	USSR
	1013	Apr 03	3	Molniya ML		B		1983 025A			Molniya 1-57	1600	EEO (63.6)	CIV	USSR
	1014	Apr 08	5	Soyuz U		P		1983 029A			Kosmos 1451	6300	LEO (82.3)	MIL	USSR
S	1015	Apr 20	12	Soyuz U		B		1983 035A	D	3/3	Soyuz T-8	6850	STA	CIV	USSR
	1016	Apr 22	2	Soyuz U		P	LC 41/1	1983 036A			Kosmos 1454	6700	LEO (67.1)	MIL	USSR
	1017	Apr 25	3	Molniya 2BL		P	LC 16/2	1983 038A			Kosmos 1456	1250	EEO (66.5)	MIL	USSR
	1018	Apr 26	1	Soyuz U		B		1983 039A			Kosmos 1457	6700	LEO (70.3)	MIL	USSR
	1019	Apr 28	2	Soyuz U		P	LC 43/1	1983 040A			Kosmos 1458	5900	LEO (82.3)	MIL	USSR
	1020	May 06	8	Soyuz U		B		1983 043A			Kosmos 1460	6300	LEO (70.3)	MIL	USSR
	1021	May 17	11	Soyuz U		P	LC 41	1983 045A			Kosmos 1462	6300	LEO (82.3)	MIL	USSR
	1022	May 26	9	Soyuz U		B		1983 050A			Kosmos 1466	6700	LEO (64.8)	MIL	USSR
	1023	May 31	5	Soyuz U		P		1983 052A			Kosmos 1467	6300	LEO (72.8)	MIL	USSR
	1024	Jun 07	7	Soyuz U		P	LC 41	1983 055A			Kosmos 1468	6300	LEO (82.3)	MIL	USSR
	1025	Jun 14	7	Soyuz U		P		1983 057A			Kosmos 1469	6300	LEO (72.8)	MIL	USSR
	1026	Jun 27	13	Soyuz U		B		1983 062A	D	2/2	Soyuz T-9	6850	STA	CIV	USSR
	1027	Jun 28	1	Soyuz U		P	LC 41/1	1983 064A			Kosmos 1471	6700	LEO (67.1)	MIL	USSR
	1028	Jul 01	3	Molniya SOL		B		1983 067A			Prognoz 9	1060	EEO (65.5)	CIV	USSR
	1029	Jul 05	4	Soyuz U		P		1983 068A			Kosmos 1472	6300	LEO (82.3)	MIL	USSR
F	**1030**	**Jul 08**	**3**	**Molniya 2BL**		**P**	**LC 43/4**	1983 070A			Kosmos 1481	1250	EEO (67.2)	MIL	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1031	Jul 13	5	Soyuz U		B		1983 071A			Kosmos 1482	6300	LEO (69.9)	MIL	USSR
	1032	Jul 19	6	Molniya ML		B		1983 073A			Molniya 1-58	1600	EEO (64)	CIV	USSR
	1033	Jul 20	1	Soyuz U		P	LC 43/4	1983 074A			Kosmos 1483	6300	LEO (82.3)	MIL	USSR
	1034	Jul 24	4	Vostok M		B		1983 075A			Kosmos 1484	2500	LEO (97.9)	MIL	USSR
	1035	Jul 27	3	Soyuz U		P		1983 075A			Kosmos 1485	6300	LEO (72.8)	MIL	USSR
	1036	Aug 05	9	Soyuz U		P	LC 43/4	1983 080A			Kosmos 1487	6300	LEO (82.3)	MIL	USSR
	1037	Aug 09	4	Soyuz U		P		1983 082A			Kosmos 1488	6300	LEO (72.8)	MIL	USSR
	1038	Aug 10	1	Soyuz U		B		1983 083A			Kosmos 1489	6700	LEO (64.7)	MIL	USSR
	1039	Aug 17	7	Soyuz U	Ts15000-302	B	LC 1	1983 085A	D		Progress 17	7020	STA	CIV	USSR
	1040	Aug 23	6	Soyuz U		P		1983 087A			Kosmos 1493	6300	LEO (72.8)	MIL	USSR
	1041	Aug 30	7	Molniya ML		P	LC 41/1	1983 090A			Molniya 3-21	1750	EEO (63.1)	CIV	USSR
	1042	Sep 03	4	Soyuz U		P	LC 43/4	1983 092A			Kosmos 1495	6300	LEO (82.3)	MIL	USSR
	1043	Sep 07	4	Soyuz U		P	LC 16/2	1983 093A			Kosmos 1496	6700	LEO (67.1)	MIL	USSR
	1044	Sep 09	2	Soyuz U		P		1983 095A			Kosmos 1497	6300	LEO (72.8)	MIL	USSR
	1045	Sep 14	5	Soyuz U		P	LC 41	1983 096A			Kosmos 1498	6300	LEO (82.3)	MIL	USSR
	1046	Sep 17	3	Soyuz U		P		1983 097A			Kosmos 1499	6300	LEO (72.8)	MIL	USSR
F	**1047**	**Sep 26**	**9**	**Soyuz U**		B		1983 F02A		2/-	Soyuz T	6600	STA	CIV	USSR
	1048	Oct 14	18	Soyuz U		B		1983 104A			Kosmos 1504	6700	LEO (64.8)	MIL	USSR
	1049	Oct 20	6	Soyuz U	Ts15000-287	B	LC 31	1983 106A	D		Progress 18	7020	STA	CIV	USSR
	1050	Oct 21	1	Soyuz U		P		1983 107A			Kosmos 1505	6300	LEO (72.8)	MIL	USSR
	1051	Oct 28	7	Vostok M		P		1983 109A			Meteor 2-10	1300	LEO (81.1)	CIV	USSR
	1052	Nov 17	20	Soyuz U		P		1983 112A			Kosmos 1509	6300	LEO (72.8)	MIL	USSR
	1053	Nov 23	6	Molniya ML		P		1983 114A			Molniya 1-59	1600	EEO (63.1)	CIV	USSR
	1054	Nov 30	7	Soyuz U		P	LC 41/1	1983 117A			Kosmos 1511	6700	LEO (67.1)	MIL	USSR
	1055	Dec 07	7	Soyuz U		P		1983 119A			Kosmos 1512	6300	LEO (72.8)	MIL	USSR
	1056	Dec 14	7	Soyuz U		P	LC 41/1	1983 121A			Kosmos 1514	5700	LEO (82.3)	MIL	USSR
	1057	Dec 21	7	Molniya ML		P	LC 41/1	1983 123A			Molniya 3-22	1750	EEO (64.7)	CIV	USSR
	1058	Dec 27	6	Soyuz U		B		1983 124A			Kosmos 1516	6700	LEO (64.8)	MIL	USSR
	1059	Dec 28	1	Molniya 2BL		P	LC 16/2	1983 126A			Kosmos 1518	1250	EEO (66.5)	MIL	USSR
	1060	1984 Jan 11	14	Soyuz U		P		1984 002A			Kosmos 1530	6300	LEO (72.8)	MIL	USSR
	1061	Jan 13	2	Soyuz U		P		1984 004A			Kosmos 1532	6700	LEO (67.1)	MIL	USSR
	1062	Jan 26	13	Soyuz U		B		1984 006A			Kosmos 1533	6300	LEO (70.3)	MIL	USSR
	1063	Feb 08	13	Soyuz U		B		1984 014A	D	3/3	Soyuz T-10	6850	STA	CIV	USSR
	1064	Feb 16	8	Soyuz U		P	LC 41	1984 017A			Kosmos 1537	6300	LEO (82.3)	MIL	USSR
	1065	Feb 21	5	Soyuz U		B	LC 31	1984 018A	D		Progress 19	7020	STA	CIV	USSR
	1066	Feb 28	7	Soyuz U		P		1984 020A			Kosmos 1539	6700	LEO (67.1)	MIL	USSR
	1067	Mar 06	7	Molniya 2BL		P	LC 16/2	1984 024A			Kosmos 1541	1250	EEO (63.3)	MIL	USSR
	1068	Mar 07	1	Soyuz U		B		1984 025A			Kosmos 1542	6300	LEO (70.3)	MIL	USSR
	1069	Mar 10	3	Soyuz U		P		1984 026A			Kosmos 1543	6700	LEO (62.8)	MIL	USSR
	1070	Mar 16	6	Molniya ML		P		1984 029A			Molniya 1-60	1600	EEO (64.8)	CIV	USSR
	1071	Mar 21	5	Soyuz U		P		1984 030A			Kosmos 1545	6300	LEO (72.8)	MIL	USSR
	1072	Apr 03	13	Soyuz U		B	LC 31	1984 032A	D	3/3	Soyuz T-11	6850	STA	CIV	USSR
	1073	Apr 04	1	Molniya 2BL		P	LC 16/2	1984 033A			Kosmos 1547	1250	EEO (66.5)	MIL	USSR
	1074	Apr 10	6	Soyuz U		P		1984 036A			Kosmos 1548	6700	LEO (67.1)	MIL	USSR
	1075	Apr 15	5	Soyuz U		B	LC 31	1984 038A	D		Progress 20	7020	STA	CIV	USSR
	1076	Apr 19	4	Soyuz U		P		1984 040A			Kosmos 1549	6300	LEO (72.9)	MIL	USSR
	1077	May 07	18	Soyuz U		B	LC 31	1984 042A	D		Progress 21	7020	STA	CIV	USSR
	1078	May 11	4	Soyuz U		P		1984 044A			Kosmos 1551	6300	LEO (72.8)	MIL	USSR
	1079	May 14	3	Soyuz U2		B		1984 045A			Kosmos 1552	6700	LEO (64.9)	MIL	USSR
	1080	May 22	8	Soyuz U		P	LC 43/4	1984 048A			Kosmos 1557	6300	LEO (82.3)	MIL	USSR
	1081	May 25	3	Soyuz U		P		1984 050A			Kosmos 1558	6700	LEO (67.1)	MIL	USSR
	1082	May 28	3	Soyuz U		B	LC 31	1984 051A	D		Progress 22	7020	STA	CIV	USSR
	1083	Jun 01	4	Soyuz U		P		1984 054A			Kosmos 1568	6300	LEO (72.8)	MIL	USSR
	1084	Jun 06	5	Molniya 2BL		P	LC 16/2	1984 055A			Kosmos 1569	1250	EEO (65.7)	MIL	USSR
	1085	Jun 11	5	Soyuz U		B		1984 058A			Kosmos 1571	6300	LEO (70)	MIL	USSR
	1086	Jun 15	4	Soyuz U		P	LC 41	1984 060A			Kosmos 1572	6300	LEO (82.3)	MIL	USSR
	1087	Jun 19	4	Soyuz U		P		1984 061A			Kosmos 1573	6300	LEO (72.8)	MIL	USSR
	1088	Jun 22	3	Soyuz U		P	LC 43/4	1984 064A			Kosmos 1575	6300	LEO (82.3)	MIL	USSR
	1089	Jun 26	4	Soyuz U		P		1984 066A			Kosmos 1576	6700	LEO (67.1)	MIL	USSR
	1090	Jun 29	3	Soyuz U		P		1984 070A			Kosmos 1580	6300	LEO (62.8)	MIL	USSR
	1091	Jul 03	4	Molniya 2BL		P	LC 43/4	1984 071A			Kosmos 1581	1250	EEO (65.2)	MIL	USSR
	1092	Jul 17	14	Soyuz U		B		1984 073A	D	3/3	Soyuz T-12	6850	STA	CIV	USSR
	1093	Jul 19	2	Soyuz U		P	LC 43/4	1984 074A			Kosmos 1582	6300	LEO (82.3)	MIL	USSR
	1094	Jul 24	5	Soyuz U		P		1984 075A			Kosmos 1583	6300	LEO (72.8)	MIL	USSR
	1095	Jul 27	3	Soyuz U		P		1984 076A			Kosmos 1584	6300	LEO (82.3)	MIL	USSR
	1096	Jul 31	4	Soyuz U		B		1984 077A			Kosmos 1585	6700	LEO (64.7)	MIL	USSR
	1097	Aug 02	2	Molniya 2BL		P	LC 16/2	1984 079A			Kosmos 1586	1250	EEO (64.2)	MIL	USSR
	1098	Aug 06	4	Soyuz U		P		1984 082A			Kosmos 1587	6300	LEO (72.8)	MIL	USSR
	1099	Aug 10	4	Molniya ML		P		1984 085A			Molniya 1-61	1600	EEO (63.1)	CIV	USSR
	1100	Aug 14	4	Soyuz U		B	LC 1	1984 086A	D		Progress 23	7020	STA	CIV	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
1101	Aug 16	2	Soyuz U		P LC 41	1984 087A		Kosmos 1590	6300	LEO (82.3)	MIL	USSR
1102	Aug 24	8	Molniya ML		P	1984 089A		Molniya 1-62	1600	EEO (63.7)	CIV	USSR
1103	Aug 30	6	Soyuz U		P LC 43/4	1984 092A		Kosmos 1591	6300	LEO (82.3)	MIL	USSR
1104	Sep 04	5	Soyuz U		P	1984 094A		Kosmos 1592	6000	LEO (72.8)	MIL	USSR
1105	Sep 07	3	Molnyia 2BL		P LC 16/2	1984 096A		Kosmos 1596	1250	EEO (63.8)	MIL	USSR
1106	Sep 13	6	Soyuz U		P LC 43/4	1984 099A		Kosmos 1597	5900	LEO (82.3)	MIL	USSR
1107	Sep 25	12	Soyuz U		P	1984 102A		Kosmos 1599	6700	LEO (67.1)	MIL	USSR
1108	Sep 26	1	Soyuz U		B	1984 103A		Kosmos 1600	6300	LEO (69.9)	MIL	USSR
1109	Oct 04	8	Molniya 2BL		P LC 16/2	1984 107A		Kosmos 1604	1250	EEO (63.2)	MIL	USSR
1110	Nov 14	41	Soyuz U		B	1984 116A		Kosmos 1608	6700	LEO (69.9)	MIL	USSR
1111	Nov 14	0	Soyuz U		P	1984 117A		Kosmos 1609	6300	LEO (72.8)	MIL	USSR
1112	Nov 21	7	Soyuz U		B	1984 119A		Kosmos 1611	6700	LEO (64.7)	MIL	USSR
1113	Nov 29	8	Soyuz U		P	1984 121A		Kosmos 1613	6300	LEO (72.8)	MIL	USSR
1114	Dec 04	5	Molniya ML		P	1984 124A		Molniya 1-63	1600	EEO (63.2)	CIV	USSR
1115	1985 Jan 09	36	Soyuz U		B	1985 002A		Kosmos 1616	6700	LEO (64.8)	MIL	USSR
1116	Jan 16	7	Molniya ML		P LC 43/4	1985 004A		Molniya 3-23	1750	EEO (64.6)	CIV	USSR
1117	Jan 16	0	Soyuz U		B	1985 005A		Kosmos 1623	6300	LEO (70)	MIL	USSR
1118	Feb 06	21	Soyuz U		P	1985 012A		Kosmos 1628	6300	LEO (72.8)	MIL	USSR
1119	Feb 27	21	Soyuz U		B	1985 017A		Kosmos 1630	6700	LEO (64.8)	MIL	USSR
1120	Mar 01	2	Soyuz U		P	1985 019A		Kosmos 1632	6300	LEO (72.8)	MIL	USSR
1121	Mar 25	24	Soyuz U2		B	1985 026A		Kosmos 1643	6700	LEO (64.7)	MIL	USSR
1122	Apr 03	9	Soyuz U		B	1985 027A		Kosmos 1644	6300	LEO (70.3)	MIL	USSR
1123	Apr 16	13	Soyuz U		P LC 41/1	1985 029A		Kosmos 1645	5700	LEO (62.8)	MIL	USSR
1124	Apr 19	3	Soyuz U		P	1985 031A		Kosmos 1647	6700	LEO (82.3)	MIL	USSR
1125	Apr 25	6	Soyuz U		P	1985 032A		Kosmos 1648	6300	LEO (82.3)	MIL	USSR
1126	Apr 26	1	Molniya SOL		B	1985 033A		Prognoz 10 IK	1000	EEO (76.8)	CIV	USSR
1127	May 15	19	Soyuz U		P	1985 036A		Kosmos 1649	6300	LEO (72.8)	MIL	USSR
1128	May 22	7	Soyuz U		P LC 41	1985 038A		Kosmos 1653	6300	LEO (82.3)	MIL	USSR
1129	May 23	1	Soyuz U		B	1985 039A		Kosmos 1654	6700	LEO (64.8)	MIL	USSR
1130	May 29	6	Molniya ML		P LC 43/4	1985 040A		Molniya 3-24	1750	EEO (63.1)	CIV	USSR
1131	Jun 06	8	Soyuz U2		B	1985 043A	D 2/2	Soyuz T-13	6850	STA (Salyut 7)	CIV	USSR
1132	Jun 07	1	Soyuz U		P	1985 044A		Kosmos 1657	6300	LEO (82.2)	MIL	USSR
1133	Jun 11	4	Molniya 2BL		P LC 41/1	1985 045A		Kosmos 1658	1250	EEO (63.7)	MIL	USSR
1134	Jun 13	2	Soyuz U		P	1985 046A		Kosmos 1659	6300	LEO (72.9)	MIL	USSR
1135	Jun 18	5	Molniya 2BL		P LC 16/2	1985 049A		Kosmos 1661	1250	EEO (63.1)	MIL	USSR
1136	Jun 21	3	Soyuz U		B LC 1	1985 051A	D	Progress 24	7020	STA (Salyut 7)	CIV	USSR
1137	Jun 21	0	Soyuz U		P LC 41	1985 052A		Kosmos 1663	6300	LEO (82.3)	MIL	USSR
1138	Jun 26	5	Soyuz U		P	1985 054A		Kosmos 1664	6300	LEO (72.8)	MIL	USSR
1139	Jul 03	7	Soyuz U		P	1985 057A		Kosmos 1665	6300	LEO (72.8)	MIL	USSR
1140	Jul 10	7	Soyuz U		P LC 41/1	1985 059A		Kosmos 1667	5700	LEO (82.3)	MIL	USSR
1141	Jul 15	5	Soyuz U		B	1985 060A		Kosmos 1668	6300	LEO (70.3)	MIL	USSR
1142	Jul 17	2	Molniya ML		P LC 43/4	1985 061A		Molniya 3-25	1750	EEO (62.9)	CIV	USSR
1143	Jul 19	2	Soyuz U		B LC 1	1985 062A	D	Kosmos 1669	7020	STA	CIV	USSR
1144	Aug 02	14	Soyuz U		P	1985 065A		Kosmos 1671	6300	LEO (72.8)	MIL	USSR
1145	Aug 07	5	Soyuz U		P LC 43/4	1985 067A		Kosmos 1672	6300	LEO (82.3)	MIL	USSR
1146	Aug 08	1	Soyuz U		B	1985 068A		Kosmos 1673	6700	LEO (64.8)	MIL	USSR
1147	Aug 12	4	Molniya 2BL		P LC 16/2	1985 071A		Kosmos 1675	1250	EEO (65)	MIL	USSR
1148	Aug 16	4	Soyuz U		P	1985 072A		Kosmos 1676	6300	LEO (67.2)	MIL	USSR
1149	Aug 22	6	Molniya ML		P	1985 074A		Molniya 1-64	1600	EEO (63.2)	CIV	USSR
1150	Aug 29	7	Soyuz U		P LC 41	1985 077A		Kosmos 1678	6300	LEO (82.3)	MIL	USSR
1151	Aug 29	0	Soyuz U		B	1985 078A		Kosmos 1679	6300	LEO (64.8)	MIL	USSR
1152	Sep 06	8	Soyuz U		P LC 41/1	1985 080A		Kosmos 1681	6300	LEO (82.3)	MIL	USSR
1153	Sep 17	11	Soyuz U2		B	1985 081A	D 3/3	Soyuz T-14	6850	STA (Salyut 7)	CIV	USSR
1154	Sep 19	2	Soyuz U		P	1985 083A		Kosmos 1683	6300	LEO (72.8)	MIL	USSR
1155	Sep 24	5	Molniya 2BL		P LC 43/4	1985 084A		Kosmos 1684	1250	EEO (62.7)	MIL	USSR
1156	Sep 26	2	Soyuz U		P	1985 085A		Kosmos 1685	6300	LEO (72.8)	MIL	USSR
1157	Sep 30	4	Molniya 2BL		P LC 16/2	1985 088A		Kosmos 1687	1250	EEO (63.1)	MIL	USSR
1158	Oct 03	3	Vostok M		B	1985 090A		Kosmos 1689	1500	LEO (98)	MIL	USSR
1159	Oct 03	0	Molniya ML		P LC 43/4	1985 091A		Molniya 3-26	1750	EEO (64.3)	CIV	USSR
1160	Oct 16	13	Soyuz U		B	1985 095A		Kosmos 1696	6300	LEO (70.3)	MIL	USSR
1161	Oct 22	6	Molniya 2BL		P LC 43/4	1985 098A		Kosmos 1698	1250	EEO (63.1)	MIL	USSR
1162	Oct 23	1	Molniya ML		B	1985 099A		Molniya 1-65	1600	EEO (64.5)	CIV	USSR
1163	Oct 25	2	Soyuz U		P	1985 101A		Kosmos 1699	6700	LEO (67.1)	MIL	USSR
1164	Oct 28	3	Molniya ML		P	1985 103A		Molniya 1-66	1600	EEO (62.9)	CIV	USSR
1165	Nov 09	12	Molniya 2BL		P LC 41/1	1985 105A		Kosmos 1701	1250	EEO (65.6)	MIL	USSR
1166	Nov 13	4	Soyuz U		P	1985 106A		Kosmos 1702	6300	LEO (72.8)	MIL	USSR
1167	Dec 03	20	Soyuz U		P	1985 111A		Kosmos 1705	6300	LEO (72.8)	MIL	USSR
1168	Dec 11	8	Soyuz U		P	1985 112A		Kosmos 1706	6700	LEO (67.1)	MIL	USSR
1169	Dec 13	2	Soyuz U		P LC 43/4	1985 115A		Kosmos 1708	6300	LEO (82.2)	MIL	USSR
1170	Dec 24	11	Molniya ML		P LC 43/4	1985 117A		Molniya 3-27	1750	EEO (62.9)	CIV	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation	Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1171	Dec 27	3	Soyuz U		P		1985 120A		Kosmos 1713	6300	LEO (62.8)	MIL	USSR
	1172	1986 Jan 08	12	Soyuz U		P		1986 001A		Kosmos 1715	6000	LEO (72.8)	MIL	USSR
	1173	Jan 15	7	Soyuz U		P		1986 004A		Kosmos 1724	6000	LEO (67.1)	MIL	USSR
	1174	Jan 28	13	Soyuz U		B		1986 009A		Kosmos 1728	6000	LEO (70)	MIL	USSR
	1175	Feb 01	4	Molniya 2BL		P	LC 16/2	1986 011A		Kosmos 1729	1250	EEO (62.8)	MIL	USSR
	1176	Feb 04	3	Soyuz U		P		1986 012A		Kosmos 1730	6000	LEO (72.9)	MIL	USSR
	1177	Feb 07	3	Soyuz U2		B		1986 013A		Kosmos 1731	6700	LEO (64.8)	MIL	USSR
	1178	Feb 26	19	Soyuz U		P		1986 020A		Kosmos 1734	6000	LEO (67.1)	MIL	USSR
	1179	Mar 13	15	Soyuz U2		B	LC 1	1986 022A	D(3) 2/2	Soyuz T-15	7000	STA (Mir/ Salyut 7/Mir)	CIV	USSR
	1180	Mar 19	6	Soyuz U2	B15000-010	B	LC 1	1986 023A	D	Progress 25	7000	STA (Mir)	CIV	USSR
F	**1181**	**Mar 26**	**7**	**Soyuz U**		**B**		1986 F02A		Kosmos			MIL	USSR
	1182	Apr 09	14	Soyuz U		B		1986 028A		Kosmos 1739	6000	LEO (64.8)	MIL	USSR
	1183	Apr 15	6	Soyuz U		P		1986 029A		Kosmos 1740	6000	LEO (72.9)	MIL	USSR
	1184	Apr 18	3	Molniya ML		P	LC 41/1	1986 031A		Molniya 3-28	1750	EEO (64.2)	CIV	USSR
	1185	Apr 23	5	Soyuz U2	B15000-009	B	LC 1	1986 032A	D	Progress 26	7000	STA (Mir)	CIV	USSR
	1186	May 14	21	Soyuz U		P		1986 033A		Kosmos 1742	6000	LEO (72.9)	MIL	USSR
	1187	May 21	7	Soyuz U2		B		1986 035A	D	Soyuz TM-1	7000	STA (Mir)	CIV	USSR
	1188	May 21	0	Soyuz U		P	LC 41/1	1986 036A		Kosmos 1744	6000	LEO (62.8)	MIL	USSR
	1189	May 28	7	Soyuz U		P	LC 43/4	1986 040A		Kosmos 1746	6000	LEO (82.3)	MIL	USSR
	1190	May 29	1	Soyuz U		B		1986 041A		Kosmos 1747	6300	LEO (70.4)	MIL	USSR
	1191	Jun 06	8	Soyuz U		B		1986 043A		Kosmos 1756	6000	LEO (64.9)	MIL	USSR
	1192	Jun 11	5	Soyuz U		P		1986 045A		Kosmos 1757	6000	LEO (64.9)	MIL	USSR
	1193	Jun 19	8	Soyuz U		B		1986 048A		Kosmos 1760	6000	LEO (70)	MIL	USSR
	1194	Jun 19	0	Molniya ML		P	LC 41/1	1986 049A		Molniya 3-29	1750	EEO (63.9)	CIV	USSR
	1195	Jul 05	16	Molniya 2BL		P	LC 43/4	1986 050A		Kosmos 1761	1250	EEO (62.8)	MIL	USSR
	1196	Jul 10	5	Soyuz U		B		1986 051A		Kosmos 1762	6300	LEO (82.5)	MIL	USSR
	1197	Jul 17	7	Soyuz U		B		1986 053A		Kosmos 1764	7000	LEO (64.9)	MIL	USSR
	1198	Jul 24	7	Soyuz U		P		1986 054A		Kosmos 1765	6300	LEO (72.9)	MIL	USSR
	1199	Jul 30	6	Molniya ML		P		1986 057A		Molniya 1-67	1600	EEO (63)	CIV	USSR
	1200	Aug 02	3	Soyuz U		P	LC 16	1986 058A		Kosmos 1768	6000	LEO (82.6)	MIL	USSR
	1201	Aug 06	4	Soyuz U2		B		1986 060A		Kosmos 1770	7000	LEO (64.7)	MIL	USSR
	1202	Aug 21	15	Soyuz U		P		1986 063A		Kosmos 1772	6300	LEO (72.9)	MIL	USSR
	1203	Aug 27	6	Soyuz U		B		1986 064A		Kosmos 1773	7000	LEO (64.9)	MIL	USSR
	1204	Aug 28	1	Molniya 2BL		P	LC 16/2	1986 065A		Kosmos 1774	1250	EEO (64.3)	MIL	USSR
	1205	Sep 03	6	Soyuz U		B		1986 066A		Kosmos 1775	6300	LEO (70.4)	MIL	USSR
	1206	Sep 05	2	Molniya ML		P		1986 068A		Molniya 1-68	1600	EEO (63.1)	CIV	USSR
	1207	Sep 17	12	Soyuz U		B		1986 072A		Kosmos 1781	6000	LEO (70.4)	MIL	USSR
F	**1208**	**Oct 03**	**16**	**Molniya 2BL**		**P**	**LC 41/1**	1986 075A		Kosmos 1783	1250	EEO (62.8)	MIL	USSR
	1209	Oct 06	3	Soyuz U		B		1986 077A		Kosmos 1784	7000	LEO (64.8)	MIL	USSR
	1210	Oct 15	9	Molniya 2BL		P	LC 41/1	1986 078A		Kosmos 1785	1250	EEO (64.4)	MIL	USSR
	1211	Oct 20	5	Molniya ML		P	LC 43/4	1986 079A		Molniya 3-30	1750	EEO (62.8)	CIV	USSR
	1212	Oct 22	2	Soyuz U		B		1986 081A		Kosmos 1787	6300	LEO (70)	MIL	USSR
	1213	Oct 31	9	Soyuz U		P	LC 16	1986 084A		Kosmos 1789	6300	LEO (82.6)	MIL	USSR
	1214	Nov 04	4	Soyuz U		P		1986 085A		Kosmos 1790	6300	LEO (72.9)	MIL	USSR
	1215	Nov 13	9	Soyuz U		B		1986 087A		Kosmos 1792	7000	LEO (64.9)	MIL	USSR
	1216	Nov 15	2	Molniya ML		P		1986 089A		Molniya 1-69	1600	EEO (63)	CIV	USSR
	1217	Nov 20	5	Molniya 2BL		P	LC 16/2	1986 091A		Kosmos 1793	1250	EEO (62.8)	MIL	USSR
	1218	Dec 04	14	Soyuz U		B		1986 095A		Kosmos 1804	6300	LEO (70)	MIL	USSR
	1219	Dec 12	8	Molniya 2BL		P	LC 43/4	1986 098A		Kosmos 1806	1250	EEO (62.8)	MIL	USSR
	1220	Dec 16	4	Soyuz U		P		1986 099A		Kosmos 1807	6700	LEO (67.1)	MIL	USSR
	1221	Dec 26	10	Molniya ML		P		1986 103A		Molniya 1-70	1600	EEO (62.9)	CIV	USSR
	1222	Dec 26	0	Soyuz U2		B		1986 102A		Kosmos 1810	7000	LEO (64.8)	MIL	USSR
	1223	1987 Jan 09	14	Soyuz U		B		1987 002A		Kosmos 1811	7000	LEO (64.6)	MIL	USSR
	1224	Jan 15	6	Soyuz U		P		1987 004A		Kosmos 1813	6300	LEO (72.8)	MIL	USSR
	1225	Jan 16	1	Soyuz U2	B15000-011	B	LC 1	1987 005A	D	Progress 27	7000	STA (Mir)	CIV	USSR
	1226	Jan 22	6	Molniya ML		P	LC 41/1	1987 008A		Molniya 3-31	1750	EEO (62.8)	CIV	USSR
	1227	Feb 05	14	Soyuz U2		B		1987 013A	D 2/3	Soyuz TM-2	7000	STA (Mir)	CIV	USSR
	1228	Feb 07	2	Soyuz U		P		1987 014A		Kosmos 1819	6700	LEO (72.8)	MIL	USSR
	1229	Feb 19	12	Soyuz U		P		1987 019A		Kosmos 1822	6300	LEO (73)	MIL	USSR
	1230	Feb 26	7	Soyuz U		P		1987 021A		Kosmos 1824	6700	LEO (67.1)	MIL	USSR
	1231	Mar 03	5	Soyuz U2	I15000-016	B	LC 1	1987 023A	D	Progress 28	7000	STA (Mir)	CIV	USSR
	1232	Mar 11	8	Soyuz U		P		1987 025A		Kosmos 1826	6300	LEO (72.9)	MIL	USSR
	1233	Apr 09	29	Soyuz U		B		1987 032A		Kosmos 1835	7000	LEO (64.8)	MIL	USSR
	1234	Apr 16	7	Soyuz U2		B		1987 033A		Kosmos 1836	7000	LEO (64.8)	MIL	USSR
	1235	Apr 21	5	Soyuz U2	I15000-015	B	LC 1	1987 034A	D	Progress 29	7000	STA (Mir)	CIV	USSR
	1236	Apr 22	1	Soyuz U		P		1987 035A		Kosmos 1837	6300	LEO (82.3)	MIL	USSR
	1237	Apr 24	2	Soyuz U		P	LC 41/1	1987 037A		Kosmos 1841	6300	LEO (62.8)	MIL	USSR
	1238	May 05	11	Soyuz U		B		1987 039A		Kosmos 1843	6000	LEO (70.4)	MIL	USSR
	1239	May 13	8	Soyuz U		B		1987 042A		Kosmos 1845	6700	LEO (70.4)	MIL	USSR
	1240	May 19	6	Soyuz U2	I15000-018	B	LC 1	1987 044A	D	Progress 30	7000	STA (Mir)	CIV	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1241	May 21	2	Soyuz U		P	LC 43/4	1987 045A			Kosmos 1846	6300	LEO (82.3)	MIL	USSR
	1242	May 26	5	Soyuz U		P		1987 046A			Kosmos 1847	7000	LEO (67.2)	MIL	USSR
	1243	May 28	2	Soyuz U		P		1987 047A			Kosmos 1848	6300	LEO (72.9)	MIL	USSR
	1244	Jun 04	7	Molniya 2BL		P	LC 16/2	1987 048A			Kosmos 1849	1250	EEO (63.3)	MIL	USSR
	1245	Jun 12	8	Molniya 2BL		P	LC 43/4	1987 050A			Kosmos 1851	1250	EEO (62.9)	MIL	USSR
F	**1246**	**Jun 18**	**6**	**Soyuz U**		**P**	**LC 43/3**	1987 F04A			Kosmos			MIL	USSR
	1247	Jul 04	16	Soyuz U		P		1987 056A			Kosmos 1863	6300	LEO (72.9)	MIL	USSR
	1248	Jul 08	4	Soyuz U		B		1987 058A			Kosmos 1865	7000	LEO (64.8)	MIL	USSR
S	1249	Jul 09	1	Soyuz U		P		1987 059A			Kosmos 1866	7000	LEO (67.2)	MIL	USSR
	1250	Jul 22	13	Soyuz U2		B	LC 1	1987 063A	D	3/3	Soyuz TM-3	7000	STA (Mir)	CIV	USSR
	1251	Aug 03	12	Soyuz U2	I15000-017	B	LC 1	1987 066A	D		Progress 31	7000	STA (Mir)	CIV	USSR
	1252	Aug 19	16	Soyuz U		P		1987 069A			Kosmos 1872	6700	LEO (72.9)	MIL	USSR
	1253	Sep 03	15	Soyuz U		P		1987 072A			Kosmos 1874	6700	LEO (72.9)	MIL	USSR
	1254	Sep 11	8	Soyuz U2		B		1987 076A			Kosmos 1881	7000	LEO (64.8)	MIL	USSR
	1255	Sep 15	4	Soyuz U		P	LC 43/4	1987 077A			Kosmos 1882	6700	LEO (82.3)	MIL	USSR
	1256	Sep 17	2	Soyuz U		P		1987 081A			Kosmos 1886	6700	LEO (67.1)	MIL	USSR
	1257	Sep 23	6	Soyuz U2	L15000-021	B	LC 1	1987 082A	D		Progress 32	7000	STA (Mir)	CIV	USSR
	1258	Sep 29	6	Soyuz U		P	LC 41/1	1987 083A			Kosmos 1887	6700	LEO (62.8)	MIL	USSR
	1259	Oct 09	10	Soyuz U		B		1987 085A			Kosmos 1889	6700	LEO (70)	MIL	USSR
	1260	Oct 22	13	Soyuz U		P		1987 089A			Kosmos 1893	6700	LEO (67.2)	MIL	USSR
	1261	Nov 11	20	Soyuz U		B		1987 092A			Kosmos 1895	6300	LEO (70.4)	MIL	USSR
	1262	Nov 14	3	Soyuz U		B		1987 093A			Kosmos 1896	6700	LEO (64.8)	MIL	USSR
	1263	Nov 20	6	Soyuz U2	L15000-022	B	LC 1	1987 094A	D		Progress 33	7020	STA (Mir)	CIV	USSR
	1264	Dec 07	17	Soyuz U		B		1987 099A			Kosmos 1899	6300	LEO (70.4)	MIL	USSR
	1265	Dec 14	7	Soyuz U		B		1987 102A			Kosmos 1901	6700	LEO (64.9)	MIL	USSR
	1266	Dec 21	7	Soyuz U2		B	LC 1	1987 104A	D	3/3	Soyuz TM-4	7070	STA (Mir)	CIV	USSR
	1267	Dec 21	0	Molniya 2BL		P	LC 41/1	1987 105A			Kosmos 1903	1250	EEO (63)	MIL	USSR
	1268	Dec 25	4	Soyuz U		B		1987 107A			Kosmos 1905	6300	LEO (70.4)	MIL	USSR
	1269	Dec 26	1	Soyuz U		P	LC 16	1987 108A			Kosmos 1906	6300	LEO (82.6)	MIL	USSR
	1270	Dec 29	3	Soyuz U		P		1987 110A			Kosmos 1907	6300	LEO (72.8)	MIL	USSR
	1271	1988 Jan 20	22	Soyuz U2	L15000-025	B	LC 1	1988 003A	D		Progress 34	7240	STA (Mir)	CIV	USSR
	1272	Jan 26	6	Soyuz U		P		1988 004A			Kosoms 1915	6300	LEO (72.9)	MIL	USSR
	1273	Feb 03	8	Soyuz U		B		1988 007A			Kosmos 1916	7000	LEO (64.9)	MIL	USSR
	1274	Feb 18	15	Soyuz U		P	LC 16	1988 010A			Kosmos 1920	6300	LEO (82.6)	MIL	USSR
	1275	Feb 19	1	Soyuz U		B		1988 011A			Kosmos 1921	6300	LEO (70)	MIL	USSR
	1276	Feb 26	7	Molniya 2BL		P	LC 41/1	1988 013A			Kosmos 1922	1250	EEO (62.9)	MIL	USSR
	1277	Mar 10	13	Soyuz U		P		1988 015A			Kosmos 1923	6300	LEO (72.8)	MIL	USSR
	1278	Mar 11	1	Molniya ML		B		1988 017A			Molniya 1-71	1600	EEO (83)	CIV	USSR
	1279	Mar 17	6	Vostok M		B		1988 021A			IRS 1A	975	SSO	CML	India
	1280	Mar 17	0	Molniya ML		P		1988 022A			Molniya 1-72	1600	EEO (63.6)	CIV	USSR
	1281	Mar 23	6	Soyuz U2	L15000-026	B	LC 1	1988 024A	D		Progress 35	7240	STA (Mir)	CIV	USSR
	1282	Mar 24	1	Soyuz U		P		1988 025A			Kosmos 1935	7000	LEO (67.2)	MIL	USSR
	1283	Mar 30	6	Soyuz U2		B		1988 027A			Kosmos 1936	7000	LEO (64.8)	MIL	USSR
	1284	Apr 11	12	Soyuz U		P		1988 030A			Kosmos 1938	6300	LEO (72.9)	MIL	USSR
	1285	Apr 14	3	Soyuz U		P	LC 41/1	1988 031A			Foton 1	6700	LEO (62.8)	CIV	USSR
	1286	Apr 20	6	Vostok M		B		1988 032A			Kosmos 1939	2000	LEO (97.9)	MIL	USSR
	1287	Apr 27	7	Soyuz U		B		1988 035A			Kosmos 1941	6300	LEO (70.3)	MIL	USSR
	1288	May 12	15	Soyuz U		P		1988 037A			Kosmos 1942	7000	LEO (67.1)	MIL	USSR
	1289	May 13	1	Soyuz U2	L15000-023	B	LC 1	1988 038A	D		Progress 36	7240	STA (Mir)	CIV	USSR
	1290	May 18	5	Soyuz U		B		1988 041A			Kosmos 1944	7000	LEO (64.8)	MIL	USSR
	1291	May 19	1	Soyuz U		B		1988 042A			Kosmos 1945	6300	LEO (70.4)	MIL	USSR
	1292	May 26	7	Molniya ML		B	LC 43/4	1988 044A			Molniya 3-32	1750	EEO (63.4)	CIV	USSR
	1293	May 31	5	Soyuz U		P	LC 41	1988 047A			Kosmos 1951	6300	LEO (82.3)	MIL	USSR
	1294	Jun 07	7	Soyuz U2		B	LC 1	1988 048A	D	3/2	Soyuz TM-5	7000	STA (Mir)	CIV	USSR
	1295	Jun 11	4	Soyuz U		B		1988 049A			Kosmos 1952	6300	LEO (70)	MIL	USSR
	1296	Jun 22	11	Soyuz U		B		1988 054A			Kosmos 1955	7000	LEO (64.8)	MIL	USSR
	1297	Jun 23	1	Soyuz U		P		1988 055A			Kosmos 1956	6300	LEO (82.3)	MIL	USSR
	1298	Jul 07	14	Soyuz U		P	LC 16	1988 057A			Kosmos 1957	6300	LEO (82.6)	MIL	USSR
F	**1299**	**Jul 09**	**2**	**Soyuz U2**		**B**		1988 F02A			Kosmos			MIL	USSR
	1300	Jul 18	9	Soyuz U2	L15000-024	B	LC 1	1988 061A	D		Progress 37	7240	STA (Mir)	CIV	USSR
F	**1301**	**Jul 27**	**9**	**Soyuz U**		**P**	**LC 43/4**	1988 F04A			Kosmos			MIL	USSR
	1302	Aug 08	12	Soyuz U		B		1988 068A			Kosmos 1962	6300	LEO (70)	MIL	USSR
	1303	Aug 12	4	Molniya ML		P		1988 069A			Molniya 1-73	1600	EEO (63.1)	CIV	USSR
	1304	Aug 16	4	Soyuz U		B		1988 070A			Kosmos 1963	6700	LEO (64.8)	MIL	USSR
	1305	Aug 23	7	Soyuz U		B		1988 072A			Kosmos 1964	6300	LEO (70)	MIL	USSR
	1306	Aug 23	0	Soyuz U		P		1988 073A			Kosmos 1965	6300	LEO (82.3)	MIL	USSR
	1307	Aug 29	6	Soyuz U2		B	LC 1	1988 075A	D	3/3	Soyuz TM-6	7070	STA (Mir)	CIV	USSR
	1308	Aug 30	1	Molniya 2BL		P	LC 16/2	1988 076A			Kosmos 1966	1250	EEO (63)	MIL	USSR
	1309	Sep 06	7	Soyuz U		P		1988 079A			Kosmos 1967	6300	LEO (72.9)	MIL	USSR
	1310	Sep 09	3	Soyuz U		P	LC 41	1988 082A			Kosmos 1968	6300	LEO (82.3)	MIL	USSR

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1311	Sep 09	0	Soyuz U2	76048930	B	LC 1	1988 083A	D		Progress 38	7240	STA (Mir)	CIV	USSR
	1312	Sep 15	6	Soyuz U		P		1988 084A			Kosmos 1969	7000	LEO (67.1)	MIL	USSR
	1313	Sep 22	7	Soyuz U		P		1988 088A			Kosmos 1973	6300	LEO (72.9)	MIL	USSR
	1314	Sep 29	7	Molniya ML		P	LC 41/1	1988 090A			Molniya 3-33	1750	EEO (63)	CIV	USSR
	1315	Oct 03	4	Molniya 2BL		P	LC 41/1	1988 092A			Kosmos 1974	1250	EEO (62.9)	MIL	USSR
	1316	Oct 13	10	Soyuz U		P		1988 094A			Kosmos 1976	6300	LEO (72.9)	MIL	USSR
	1317	Oct 25	12	Molniya 2BL		P	LC 41/1	1988 096A			Kosmos 1977	1250	EEO (62.9)	MIL	USSR
	1318	Oct 27	2	Soyuz U		P		1988 097A			Kosmos 1978	6300	LEO (72.9)	MIL	USSR
F	**1319**	**Nov 11**	**15**	**Soyuz U2**		**B**		1988 F05A			Kosmos			MIL	USSR
	1320	Nov 24	13	Soyuz U		P		1988 103A			Kosmos 1981	6300	LEO (62.8)	MIL	USSR
	1321	Nov 26	2	Soyuz U2		B	LC 1	1988 104A	D	3/3	Soyuz TM-7	7000	STA (Mir)	CIV	USSR
	1322	Nov 30	4	Soyuz U		B		1988 105A			Kosmos 1982	6300	LEO (70)	MIL	USSR
	1323	Dec 08	8	Soyuz U		P		1988 107A			Kosmos 1983	6300	LEO (62.8)	MIL	USSR
	1324	Dec 16	8	Soyuz U		P		1988 110A			Kosmos 1984	6300	LEO (62.8)	MIL	USSR
	1325	Dec 22	6	Molniya ML		P	LC 43/3	1988 112A			Molniya 3-34	1750	EEO (62.8)	CIV	USSR
	1326	Dec 25	3	Soyuz U2	Ye15000-029	B	LC 1	1988 114A	D		Progress 39	7240	STA (Mir)	CIV	USSR
	1327	Dec 28	3	Molniya ML		P		1988 115A			Molniya 1-74	1600	EEO (62.8)	CIV	USSR
	1328	Dec 29	1	Soyuz U		B		1988 116A			Kosmos 1986	6700	LEO (64.8)	MIL	USSR
	1329	1989 Jan 12	14	Soyuz U		P	LC 16	1989 002A			Kosmos 1990	6300	LEO (82.9)	MIL	USSR
	1330	Jan 18	6	Soyuz U		B		1989 003A			Kosmos 1991	6300	LEO (70)	MIL	USSR
	1331	Jan 28	10	Soyuz U		B		1989 007A			Kosmos 1993	6500	LEO (64.7)	MIL	USSR
	1332	Feb 10	13	Soyuz U2	Ye15000-032	B	LC 1	1989 008A	D		Progress 40	7250	STA (Mir)	CIV	USSR
	1333	Feb 10	0	Soyuz U		P		1989 010A			Kosmos 2000	6300	LEO (82.4)	MIL	USSR
	1334	Feb 14	4	Molniya 2BL		P	LC 43/3	1989 011A			Kosmos 2001	1250	EEO (62.8)	MIL	USSR
	1335	Feb 15	1	Molniya ML		B		1989 014A			Molniya 1-75	1600	EEO (62.9)	CIV	USSR
	1336	Feb 17	2	Soyuz U		P		1989 015A			Kosmos 2003	6300	LEO (62.8)	MIL	USSR
	1337	Mar 02	13	Soyuz U		P		1989 019A			Kosmos 2005	6500	LEO (62.8)	MIL	USSR
	1338	Mar 16	14	Soyuz U		P		1989 022A			Kosmos 2006	6300	LEO (62.8)	MIL	USSR
	1339	Mar 16	0	Soyuz U2	T15000-034	B	LC 1	1989 023A	D		Progress 41	7250	STA (Mir)	CIV	USSR
	1340	Mar 23	7	Soyuz U2		B		1989 024A			Kosmos 2007	7000	LEO (64.7)	MIL	USSR
	1341	Apr 06	14	Soyuz U		P		1989 029A			Kosmos 2017	6300	LEO (62.8)	MIL	USSR
	1342	Apr 20	14	Soyuz U		P		1989 031A			Kosmos 2018	6500	LEO (62.8)	MIL	USSR
	1343	Apr 26	6	Soyuz U		P	LC 41/1	1989 032A			Foton 2	6200	LEO (62.8)	CIV	USSR
	1344	May 05	9	Soyuz U		P		1989 034A			Kosmos 2019	6300	LEO (62.8)	MIL	USSR
	1345	May 17	12	Soyuz U		B		1989 036A			Kosmos 2020	6500	LEO (64.8)	MIL	USSR
	1346	May 24	7	Soyuz U		B		1989 037A			Kosmos 2021	6500	LEO (64.9)	MIL	USSR
	1347	May 25	1	Soyuz U		P	LC 43/3	1989 038A			Resurs-F 1	6300	LEO (82.3)	CIV	USSR
								1989 038C	A		Pion 1	78	LEO (82.3)	CIV	USSR
								1989 038D	A		Pion 2	78	LEO (82.3)	CIV	USSR
	1348	Jun 01	7	Soyuz U		P		1989 040A			Kosmos 2025	6300	LEO (62.8)	MIL	USSR
	1349	Jun 08	7	Molniya ML		P	LC 43/3	1989 043A			Molniya 3-35	1750	EEO (62.8)	CIV	USSR
	1350	Jun 16	8	Soyuz U		B		1989 047A			Kosmos 2028	6300	LEO (69.9)	MIL	USSR
	1351	Jun 27	11	Soyuz U		P	LC 16	1989 049A			Resurs-F 2	6300	LEO (82.5)	CIV	USSR
	1352	Jul 05	8	Soyuz U		P		1989 051A			Kosmos 2029	6300	LEO (82.4)	MIL	USSR
	1353	Jul 12	7	Soyuz U		P		1989 054A			Kosmos 2030	6500	LEO (67.1)	MIL	USSR
	1354	Jul 18	6	Soyuz U		P	LC 16	1989 055A			Resurs-F 3	6300	LEO (82.5)	CIV	USSR
								1989 055C	A		Pion 3	78	LEO (82.5)	CIV	USSR
								1989 055D	A		Pion 4	78	LEO (82.5)	CIV	USSR
	1355	Jul 18	0	Soyuz U2		B		1989 056A			Kosmos 2031	6500	LEO (50.5)	MIL	USSR
	1356	Jul 20	2	Soyuz U		P		1989 057A			Kosmos 2032	6300	LEO (82.3)	MIL	USSR
	1357	Aug 02	13	Soyuz U		P		1989 060A			Kosmos 2035	6300	LEO (82.5)	MIL	USSR
	1358	Aug 15	13	Soyuz U		P	LC 43/4	1989 063A			Resurs-F 4	6300	LEO (82.3)	CIV	USSR
	1359	Aug 22	7	Soyuz U		P		1989 065A			Kosmos 2036	6300	LEO (62.8)	MIL	USSR
	1360	Aug 23	1	Soyuz U2	T15000-037	B	LC 1	1989 066A	D		Progress M-1	7250	STA (Mir)	CIV	USSR
	1361	Sep 05	13	Soyuz U2		B		1989 071A	D	2/2	Soyuz TM-8	7150	STA (Mir)	CIV	USSR
	1362	Sep 06	1	Soyuz U		P	LC 43/3	1989 073A			Resurs-F 5	6300	LEO (82.3)	CIV	USSR
	1363	Sep 15	9	Soyuz U		P	LC 41/1	1989 075A			Kosmos 2044	6000	LEO (82.3)	MIL	USSR
	1364	Sep 22	7	Soyuz U		B		1989 076A			Kosmos 2045	6300	LEO (69.9)	MIL	USSR
	1365	Sep 27	5	Molniya ML		P		1989 078A			Molniya 1-76	1600	EEO (62.9)	CIV	USSR
	1366	Oct 03	6	Soyuz U		P		1989 082A			Kosmos 2047	6500	LEO (67.1)	MIL	USSR
	1367	Oct 17	14	Soyuz U		P		1989 083A			Kosmos 2048	6300	LEO (62.8)	MIL	USSR
	1368	Nov 17	31	Soyuz U2		B		1989 088A			Kosmos 2049	7000	LEO (64.7)	MIL	USSR
	1369	Nov 23	6	Molniya 2BL		P	LC 16/2	1989 091A			Kosmos 2050	1250	EEO (63)	MIL	USSR
	1370	Nov 28	5	Molniya ML		P	LC 43/3	1989 094A			Molniya 3-36	1750	EEO (62.8)	CIV	USSR
	1371	Nov 30	2	Soyuz U		P		1989 095A			Kosmos 2052	6500	LEO (67.1)	MIL	USSR
	1372	Dec 20	20	Soyuz U2	039	B	LC 1	1989 099A	D		Progress M-2	7250	STA (Mir)	CIV	USSR
	1373	1990 Jan 17	28	Soyuz U		P		1990 003A			Kosmos 2055	6300	LEO (62.8)	MIL	USSR
	1374	Jan 23	6	Molniya ML		P	LC 43/4	1990 006A			Molniya 3-37	1750	EEO (62.7)	CIV	USSR
	1375	Jan 25	2	Soyuz U		P		1990 009A			Kosmos 2057	6500	LEO (62.8)	MIL	USSR
	1376	Feb 11	17	Soyuz U2		B		1990 014A	D	2/2	Soyuz TM-9	7150	STA (Mir)	CIV	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1377	Feb 28	17	Soyuz U2	T15000-040	B	LC 1	1990 020A	D		Progress M-3	7250	STA (Mir)	CIV	USSR
	1378	Mar 22	22	Soyuz U		P		1990 024A			Kosmos 2062	6300	LEO (82.3)	MIL	USSR
	1379	Mar 27	5	Molniya 2BL		P	LC 43/3	1990 026A			Kosmos 2063	1250	EEO (63.6)	MIL	USSR
F	**1380**	**Apr 03**	**7**	**Soyuz U**		**P**	**LC 43**	1990 F02A			Kosmos			MIL	USSR
	1381	Apr 11	8	Soyuz U		P	LC 43/3	1990 032A			Foton 3	6200	LEO (82.8)	CIV	USSR
	1382	Apr 13	2	Soyuz U2		B		1990 033A			Kosmos 2072	6300	LEO (64.8)	MIL	USSR
	1383	Apr 17	4	Soyuz U		P		1990 035A			Kosmos 2073	6300	LEO (82.4)	MIL	USSR
	1384	Apr 26	9	Molniya ML		P		1990 039A			Molniya 1-77	1600	EEO (62.8)	CIV	USSR
	1385	Apr 28	2	Molniya 2BL		P	LC 16/2	1990 040A			Kosmos 2076	1250	EEO (63.2)	MIL	USSR
	1386	May 05	7	Soyuz U2	T15000-041	B	LC 1	1990 041A	D		Progress 42	7150	STA (Mir)	CIV	USSR
	1387	May 07	2	Soyuz U		P		1990 042A			Kosmos 2077	6500	LEO (62.8)	MIL	USSR
	1388	May 15	8	Soyuz U		B		1990 044A			Kosmos 2078	6500	LEO (70)	MIL	USSR
	1389	May 29	14	Soyuz U		P	LC 43/4	1990 047A			Resurs-F 6	6300	LEO (82.3)	CIV	USSR
	1390	Jun 13	15	Molniya ML		P	LC 43/4	1990 052A			Molniya 3-38	1750	EEO (62.9)	CIV	USSR
	1391	Jun 19	6	Soyuz U		P		1990 053A			Kosmos 2083	6300	LEO (82.5)	MIL	USSR
F	**1392**	**Jun 21**	**2**	**Molniya 2BL**		**P**	**LC 43/4**	1990 055A			Kosmos 2084	1250	EEO (62.8)	MIL	USSR
F	**1393**	**Jul 03**	**12**	**Soyuz U**		**P**	**LC 16**	1990 F03A			Kosmos			MIL	USSR
	1394	Jul 11	8	Soyuz U		B		1990 058A			Gamma	7318	LEO (51.6)	CIV	USSR
	1395	Jul 17	6	Soyuz U		P	LC 43/3	1990 060A			Resurs-F 7	6300	LEO (82.3)	CIV	USSR
	1396	Jul 20	3	Soyuz U		P		1990 062A			Kosmos 2086	6300	LEO (82.3)	MIL	USSR
	1397	Jul 24	4	Molniya 2BL		P	LC 16/2	1990 064A			Kosmos 2087	1250	EEO (62.9)	MIL	USSR
	1398	Aug 01	8	Soyuz U2		B		1990 067A	D	2/3	Soyuz TM-10	7150	STA (Mir)	CIV	USSR
	1399	Aug 03	2	Soyuz U		P		1990 069A			Kosmos 2089	6500	LEO (62.8)	MIL	USSR
	1400	Aug 10	7	Molniya ML		P		1990 071A			Molniya 1-78	1600	EEO (62.9)	CIV	USSR
	1401	Aug 15	5	Soyuz U2	T15000-042	B	LC 1	1990 072A	D		Progress M-4	7250	STA (Mir)	CIV	USSR
	1402	Aug 16	1	Soyuz U		P	LC 43/4	1990 073A			Resurs-F 8	6300	LEO (82.3)	CIV	USSR
	1403	Aug 28	12	Molniya 2BL		P	LC 43/4	1990 076A			Kosmos 2097	1250	EEO (63.4)	MIL	USSR
	1404	Aug 31	3	Soyuz U		P		1990 080A			Kosmos 2099	6300	LEO (82.3)	MIL	USSR
	1405	Sep 07	7	Soyuz U		P	LC 16	1990 082A			Resurs-F 9	6300	LEO (82.5)	CIV	USSR
	1406	Sep 20	13	Molniya ML		P	LC 43/4	1990 084A			Molniya 3-39	1750	EEO (62.9)	CIV	USSR
	1407	Sep 27	7	Soyuz U2	T15000-044	B	LC 1	1990 085A	D		Progress M-5	2050	STA (Mir)	CIV	USSR
	1408	Oct 01	4	Soyuz U2		B		1990 087A			Kosmos 2101	6700	LEO (64.8)	MIL	USSR
	1409	Oct 16	15	Soyuz U		P		1990 092A			Kosmos 2102	6700	LEO (62.8)	MIL	USSR
	1410	Nov 16	31	Soyuz U		P		1990 098A			Kosmos 2104	6300	LEO (62.8)	MIL	USSR
	1411	Nov 20	4	Molniya 2BL		P	LC 16/2	1990 099A			Kosmos 2105	1250	EEO (63.4)	MIL	USSR
	1412	Nov 23	3	Molniya ML		P		1990 101A			Molniya 1-79	1600	EEO (62.9)	CIV	USSR
	1413	Dec 02	9	Soyuz U2		B	LC 1	1990 107A	D	3/3	Soyuz TM-11	7150	STA (Mir)	CIV	USSR
	1414	Dec 04	2	Soyuz U		P		1990 109A			Kosmos 2108	6700	LEO (62.8)	MIL	USSR
	1415	Dec 21	17	Soyuz U2		B		1990 113A			Kosmos 2113	7000	LEO (64.8)	MIL	USSR
	1416	Dec 26	5	Soyuz U		P		1990 115A			Kosmos 2120	6300	LEO (82.6)	MIL	USSR
	1417	1991 Jan 14	19	Soyuz U2	T15000-045	B	LC 1	1991 002A	D		Progress M-6	7250	STA (Mir)	CIV	USSR
	1418	Jan 17	3	Soyuz U		P		1991 004A			Kosmos 2121	6300	LEO (82.5)	MIL	USSR
	1419	Feb 07	21	Soyuz U		P		1991 008A			Kosmos 2124	6700	LEO (62.8)	MIL	USSR
	1420	Feb 15	8	Soyuz U		B		1991 011A			Kosmos 2134	6700	LEO (64.7)	MIL	USSR
	1421	Feb 15	0	Molniya ML		P		1991 012A			Molniya 1-80	1600	EEO (53)	CIV	USSR
	1422	Mar 06	19	Soyuz U		P		1991 016A			Kosmos 2136	6300	LEO (62.8)	MIL	USSR
	1423	Mar 19	13	Soyuz U2	R15000-049	B	LC 1	1991 020A	D		Progress M-7	7250	STA (Mir)	CIV	USSR
	1424	Mar 22	3	Molniya ML		P	LC 43/4	1991 022A			Molniya 3-40	1750	EEO (62.9)	CIV	USSR
	1425	Mar 26	4	Soyuz U		P		1991 023A			Kosmos 2138	6700	LEO (56.1)	MIL	USSR
	1426	May 18	53	Soyuz U2		B	LC 1	1991 034A	D	3/3	Soyuz TM-12	7150	STA (Mir)	CIV	USSR
	1427	May 21	3	Soyuz U		P	LC 43/4	1991 035A			Resurs-F 10	6300	LEO (82.3)	CIV	USSR
	1428	May 24	3	Soyuz U		P		1991 036A			Kosmos 2149	6500	LEO (67.1)	MIL	USSR
	1429	May 30	6	Soyuz U2	R15000-050	B	LC 1	1991 038A	D		Progress M-8	7250	STA (Mir)	CIV	USSR
	1430	Jun 18	19	Molniya ML		P		1991 043A			Molniya 1-81	1600	EEO (63)	CIV	USSR
	1431	Jun 28	10	Soyuz U		P	LC 43/3	1991 044A			Resurs-F 11	6300	LEO (82.3)	CIV	USSR
	1432	Jul 09	11	Soyuz U		P		1991 048A			Kosmos 2152	6300	LEO (82.3)	MIL	USSR
	1433	Jul 10	1	Soyuz U2		B		1991 049A			Kosmos 2153	7000	LEO (64.8)	MIL	USSR
	1434	Jul 23	13	Soyuz U		P	LC 43/3	1991 052A			Resurs-F 12	6300	LEO (82.3)	CIV	USSR
	1435	Aug 01	9	Molniya ML		P		1991 053A			Molniya 1-82	1600	EEO (63.2)	CIV	USSR
	1436	Aug 20	19	Soyuz U2	G15000-047	B	LC 1	1991 057A	D		Progress M-9	7250	STA (Mir)	CIV	USSR
	1437	Aug 21	1	Soyuz U		P	LC 43/3	1991 058A			Resurs-F 13	6300	LEO (82.3)	CIV	USSR
	1438	Aug 29	8	Vostok M		B		1991 061A			IRS 1B	980	SSO	CIV	India
	1439	Sep 17	19	Molniya ML		P	LC 43/4	1991 065A			Molniya 3-41	1750	EEO (62.9)	CIV	USSR
	1440	Sep 19	2	Soyuz U		P		1991 066A			Kosmos 2156	6500	LEO (67.1)	MIL	USSR
	1441	Oct 02	13	Soyuz U2		B	LC 1	1991 069A	D	3/3	Soyuz TM-13	7150	STA (Mir)	CIV	USSR
	1442	Oct 04	2	Soyuz U		P	LC 43/4	1991 070A			Foton 4	6200	LEO (62.8)	CIV	USSR
	1443	Oct 09	5	Soyuz U2		B		1991 071A			Kosmos 2163	6500	LEO (64.8)	MIL	USSR
	1444	Oct 17	8	Soyuz U2	15000-055	B	LC 1	1991 073A	D		Progress M-10	7250	STA (Mir)	CIV	USSR
	1445	Nov 20	34	Soyuz U		P		1991 078A			Kosmos 2171	6500	LEO (62.8)	MIL	USSR
	1446	Dec 17	27	Soyuz U		B		1991 085A			Kosmos 2174	6500	LEO (64.8)	MIL	USSR

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad		Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1447	1992 Jan 21	35	Soyuz U		P		1992 001A			Kosmos 2175	6500	LEO (67.1)	MIL	Russia
	1448	Jan 24	3	Molniya 2BL		P	LC 43/4	1992 003A			Kosmos 2176	1250	EEO (64)	MIL	Russia
	1449	Jan 25	1	Soyuz U2	R15000-058	B	LC 1	1992 004A	D		Progress M-11	7250	STA (Mir)	CIV	Russia
	1450	Mar 04	39	Molniya ML		P		1992 011A			Molniya 1-83	1600	EEO (62.9)	CIV	Russia
	1451	Mar 17	13	Soyuz U2		B	LC 1	1992 014A	D	3/3	Soyuz TM-14	7150	STA (Mir)	CIV	Russia
	1452	Apr 01	15	Soyuz U		P		1992 016A			Kosmos 2182	6500	LEO (67.1)	MIL	Russia
	1453	Apr 08	7	Soyuz U2		B		1992 018A			Kosmos 2183	7000	LEO (64.8)	MIL	Russia
	1454	Apr 19	11	Soyuz U2	R15000-059	B	LC 1	1992 022A	D		Progress M-12	7250	STA (Mir)	CIV	Russia
	1455	Apr 29	10	Soyuz U		P	LC 43/4	1992 024A			Resurs-F 14	6300	LEO (69.9)	CIV	Russia
	1456	Apr 29	0	Soyuz U		B		1992 025A			Kosmos 2185	6500	LEO (69.9)	MIL	Russia
	1457	May 28	29	Soyuz U		P		1992 029A			Kosmos 2186	6500	LEO (62.8)	MIL	Russia
	1458	Jun 23	26	Soyuz U		P	LC 43/3	1992 033A			Resurs-F 15	6300	LEO (82.3)	CIV	Russia
	1459	Jun 30	7	Soyuz U2	15000-062	B	LC 31	1992 035A	D		Progress M-13	7250	STA (Mir)	CIV	Russia
	1460	Jul 08	8	Molniya 2BL		P	LC 43/3	1992 040A			Kosmos 2196	1250	EEO (63.3)	MIL	Russia
	1461	Jul 24	16	Soyuz U		P		1992 045A			Kosmos 2203	6500	LEO (62.8)	MIL	Russia
	1462	Jul 27	3	Soyuz U2		B	LC 1	1992 046A	D	3/2	Soyuz TM-15	7150	STA (Mir)	CIV	Russia
	1463	Jul 30	3	Soyuz U		P		1992 048A			Kosmos 2207	6300	LEO (82.3)	MIL	Russia
	1464	Aug 06	7	Molniya ML		P		1992 050A			Molniya 1-84	1600	EEO (62.9)	CIV	Russia
	1465	Aug 12	6	Soyuz U2	U15000-064	B	LC 31	1992 055A	D		Progress M-14	7250	STA (Mir)	CIV	Russia
	1466	Aug 19	7	Soyuz U		P	LC 16	1992 056A			Resurs-F 16	6300	LEO (82.5)	CIV	Russia
								1992 056C	A		Pion-Germes-1	50	LEO (82.5)	CIV	Russia
								1992 056D	A		Pion-Germes-2	50	LEO (82.5)	CIV	Russia
	1467	Sep 22	34	Soyuz U		P		1992 062A			Kosmos 2210	6700	LEO (67.1)	MIL	Russia
	1468	Oct 08	16	Soyuz U		P	LC 43/4	1992 065A			Foton 5	6200	LEO (62.8)	CIV	Russia
	1469	Oct 14	6	Molniya ML		P	LC 43/3	1992 067A			Molniya 3-42	1750	EEO (62.8)	CIV	Russia
	1470	Oct 21	7	Molniya 2BL		P	LC 16/2	1992 069A			Kosmos 2217	1250	EEO (63)	MIL	Russia
	1471	Oct 27	6	Soyuz U2	15000-061	B	LC 31	1992 071A	D		Progress M-15	7250	STA (Mir)	CIV	Russia
	1472	Nov 15	19	Soyuz U		P	LC 16	1992 075A			Resurs 500	6300	LEO (82.5)	CIV	Russia
	1473	Nov 20	5	Soyuz U		P		1992 077A			Kosmos 2220	6500	LEO (67.1)	MIL	Russia
	1474	Nov 25	5	Molniya 2BL		P	LC 43/3	1992 081A			Kosmos 2222	1250	EEO (62.9)	MIL	Russia
	1475	Dec 02	7	Molniya ML		P	LC 43/3	1992 085A			Molniya 3-43	1750	EEO (62.9)	CIV	Russia
	1476	Dec 09	7	Soyuz U2		B		1992 087A			Kosmos 2223	7000	LEO (64.7)	MIL	Russia
	1477	Dec 22	13	Soyuz U2		B		1992 091A			Kosmos 2225	6500	LEO (64.9)	MIL	Russia
	1478	Dec 29	7	Soyuz U		P	LC 43/3	1992 095A			Kosmos 2229	6000	LEO (62.8)	MIL	Russia
	1479	1993 Jan 13	15	Molniya ML		P		1993 002A			Molniya 1-85	1600	EEO (63.4)	CIV	Russia
	1480	Jan 19	6	Soyuz U		P		1993 004A			Kosmos 2231	6500	LEO (67.1)	MIL	Russia
	1481	Jan 24	5	Soyuz U2		B	LC 1	1993 005A	D	2/3	Soyuz TM-16	7150	STA (Mir)	CIV	Russia
	1482	Jan 26	2	Molniya 2BL		P	LC 16/2	1993 006A			Kosmos 2232	1250	EEO (62.9)	MIL	Russia
	1483	Feb 21	26	Soyuz U2	U15000-068	B	LC 1	1993 012A	D		Progress M-16	7250	STA (Mir)	CIV	Russia
	1484	Mar 31	38	Soyuz U2	N15000-069	B	LC 1	1993 019A	D		Progress M-17	7250	STA (Mir)	CIV	Russia
	1485	Apr 02	2	Soyuz U		P		1993 021A			Kosmos 2240	6500	LEO (82.9)	MIL	Russia
	1486	Apr 06	4	Molniya 2BL		P	LC 43/4	1993 022A			Kosmos 2241	1250	EEO (63.6)	MIL	Russia
	1487	Apr 21	15	Molniya ML		P	LC 43/4	1993 025A			Molniya 3-44	1750	EEO (62.8)	CIV	Russia
F	**1488**	**Apr 27**	**6**	**Soyuz U**		**B**	**LC 31**	1993 028A			Kosmos 2243	6300	LEO (70.3)	MIL	Russia
	1489	May 21	24	Soyuz U		P	LC 16	1993 033A			Resurs-F 17	6300	LEO (82.5)	CIV	Russia
	1490	May 22	1	Soyuz U2		B	LC 1	1993 034A	D		Progress M-18	7250	STA (Mir)	CIV	Russia
	1491	May 26	4	Molniya ML		P		1993 035A			Molniya 1-86	1600	EEO (62.9)	CIV	Russia
	1492	Jun 25	30	Soyuz U		P	LC 16	1993 040A			Resurs-F 18	6300	LEO (82.5)	CIV	Russia
	1493	Jul 01	6	Soyuz U2		B	LC 1	1993 043A	D	3/2	Soyuz TM-17	7150	STA (Mir)	CIV	Russia
S	1494	Jul 14	13	Soyuz U		P		1993 045A			Kosmos 2259	6500	LEO (67.1)	MIL	Russia
	1495	Jul 22	8	Soyuz U		P		1993 047A			Kosmos 2260	6300	LEO (82.2)	MIL	Russia
	1496	Aug 04	13	Molniya ML		P	LC 43/3	1993 049A			Molniya 3-45	1750	EEO (62.8)	CIV	Russia
	1497	Aug 10	6	Molniya 2BL		P	LC 16/2	1993 051A			Kosmos 2261	1250	EEO (63)	MIL	Russia
	1498	Aug 10	0	Soyuz U	N15000-634	B	LC 1	1993 052A	D		Progress M-19	7250	STA (Mir)	CIV	Russia
	1499	Aug 24	14	Soyuz U		P	LC 16	1993 053A			Resurs-F 19	6300	LEO (82.5)	CIV	Russia
	1500	Sep 07	14	Soyuz U2		B		1993 057A			Kosmos 2262	6500	LEO (64.8)	MIL	Russia
	1501	Oct 11	34	Soyuz U	77044270	B	LC 1	1993 064A	D		Progress M-20	7250	STA (Mir)	CIV	Russia
	1502	Nov 05	25	Soyuz U2		B	LC 1	1993 071A			Kosmos 2267	6500	LEO (70.4)	MIL	Russia
	1503	Dec 22	47	Molniya ML		P	LC 43	1993 079A			Molniya 1-87	1600	EEO (62.7)	CIV	Russia
	1504	1994 Jan 08	17	Soyuz U2		B	LC 1	1994 001A	D	3/2	Soyuz TM-18	7150	STA (Mir)	CIV	Russia
	1505	Jan 28	20	Soyuz U	N15000-635	B	LC 1	1994 005A	D		Progress M-21	7250	STA (Mir)	CIV	Russia
	1506	Mar 17	48	Soyuz U		P	LC 43	1994 018A			Kosmos 2274	6500	LEO (67.1)	MIL	Russia
	1507	Mar 22	5	Soyuz U	76032992	B	LC 1	1994 019A	D		Progress M-22	7250	STA (Mir)	CIV	Russia
	1508	Apr 28	37	Soyuz U2		B	LC 31	1994 025A			Kosmos 2280	7000	LEO (70.3)	MIL	Russia
	1509	May 22	24	Soyuz U	76024355	B	LC 1	1994 031A	D		Progress M-23	7250	STA (Mir)	CIV	Russia
	1510	Jun 07	16	Soyuz U		P	LC 16	1994 032A			Kosmos 2281	6300	LEO (82.5)	MIL	Russia
	1511	Jun 14	7	Soyuz U		P	LC 43/3	1994 033A			Foton 9	6200	LEO (62.8)	CIV	Russia
	1512	Jul 01	17	Soyuz U2		B	LC 1	1994 036A	D	2/3	Soyuz TM-19	7150	STA (Mir)	CIV	Russia
	1513	Jul 20	19	Soyuz U		P	LC 43	1994 042A			Kosmos 2283	6500	LEO (67.1)	MIL	Russia
	1514	Jul 29	9	Soyuz U		B	LC 31	1994 044A			Kosmos 2284	6500	LEO (70.3)	MIL	Russia

B = Baikonur; P = Plesetsk; LC 1 and LC 31 are at Baikonur. LC 16, LC 41, and LC 43 are at Plesetsk.

T = Test Flight; **F** = Failure; **P** = Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1515	Aug 05	7	Molniya 2BL		P	LC 16/2	1994 048A			Kosmos 2286	1250	EEO (62.8)	MIL	Russia
	1516	Aug 23	18	Molniya ML		P	LC 43	1994 051A			Molniya 3-46	1750	EEO (62.7)	CIV	Russia
	1517	Aug 25	2	Soyuz U	N15000-636	B	LC 1	1994 052A	D		Progress M-24	7250	STA (Mir)	CIV	Russia
	1518	Oct 03	39	Soyuz U2		B	LC 1	1994 063A	D	3/3	Soyuz TM-20	7150	STA (Mir)	CIV	Russia
	1519	Nov 11	39	Soyuz U	Ya15000-638	B	LC 1	1994 075A	D		Progress M-25	7250	STA (Mir)	CIV	Russia
	1520	Dec 14	33	Molniya ML		P	LC 43/4	1994 081A			Molniya 1-88	1600	EEO (62.8)	CIV	Russia
	1521	Dec 29	15	Soyuz U2		B	LC 31	1994 088A			Kosmos 2305	7000	LEO (64.9)	MIL	Russia
	1522	1995 Feb 15	48	Soyuz U	Ya15000-641	B	LC 1	1995 005A	D		Progress M-26	7250	STA (Mir)	CIV	Russia
	1523	Feb 16	1	Soyuz U		P	LC 43	1995 006A			Foton 10	6200	LEO (62.8)	CIV	Russia
	1524	Mar 14	26	Soyuz U2		B	LC 1	1995 010A	D	3/2	Soyuz TM-21	7150	STA (Mir)	CIV	Russia
	1525	Mar 22	8	Soyuz U		P	LC 43	1995 014A			Kosmos 2311	6500	LEO (67.2)	MIL	Russia
	1526	Apr 09	18	Soyuz U		B	LC 1	1995 020A	D		Progress M-27	7250	STA (Mir)	CIV	Russia
	1527	May 24	45	Molniya 2BL		P	LC 16/2	1995 026A			Kosmos 2312	1250	EEO (63.2)	MIL	Russia
	1528	Jun 28	35	Soyuz U		P	LC 43	1995 031A			Kosmos 2314	6500	LEO (67.1)	MIL	Russia
	1529	Jul 20	22	Soyuz U		B	LC 1	1995 036A	D		Progress M-28	7250	STA (Mir)	CIV	Russia
	1530	Aug 02	13	Molniya 2BL	N15000-294 10M127S	P	LC 43/3	1995 039A			Interbol 1	1250	EEO (71.2)	CIV	Russia
								1995 039AF	A		Magion 4	58	EEO (63.1)	CIV	Czech Rep.
	1531	Aug 09	7	Molniya ML	PVB77031-674	P	LC 43/3	1995 042A			Molniya 3-47	1750	EEO (62.9)	CIV	Russia
	1532	Sep 03	25	Soyuz U2		B	LC 1	1995 047A	D	3/3	Soyuz TM-22	7150	STA (Mir)	CIV	Russia
	1533	Sep 26	23	Soyuz U		P	LC 43/4	1995 050A			Resurs-F 20	6300	LEO (82.3)	CIV	Russia
	1534	Sep 29	3	Soyuz U		B	LC 31	1995 051A			Kosmos 2320	7000	LEO (64.9)	MIL	Russia
	1535	Oct 08	9	Soyuz U	V15000-645	B	LC 1	1995 053A	D		Progress M-29	7250	STA (Mir)	CIV	Russia
	1536	Dec 18	71	Soyuz U	647	B	LC 1	1995 070A	D		Progress M-30	7250	STA (Mir)	CIV	Russia
	1537	Dec 28	10	Molniya 2BL		B	LC 31	1995 072A			IRS 1C	1330	SSO	CIV	India
								1995 072B	A		Skipper	230	SSO	CIV	USA
	1538	1996 Feb 21	55	Soyuz U	651	B	LC 1	1996 011A	D	2/3	Soyuz TM-23	7150	STA (Mir)	CIV	Russia
	1539	Mar 14	22	Soyuz U		P	LC 43/4	1996 016A			Kosmos 2331	6500	LEO (67.1)	MIL	Russia
	1540	May 05	52	Soyuz U		B	LC 1	1996 028A	D		Progress M-31	7250	STA (Mir)	CIV	Russia
F	**1541**	**May 14**	**9**	**Soyuz U**	**PVB78051-368**	**B**	**LC 31**	1996 F02A			Kosmos			MIL	Russia
F	**1542**	**Jun 20**	**37**	**Soyuz U**		**P**	**LC 16**	1996 F04A			Kosmos	6500		MIL	Russia
	1543	Jul 31	41	Soyuz U		B	LC 1	1996 043A	D		Progress M-32	7250	STA (Mir)	CIV	Russia
	1544	Aug 14	14	Molniya ML		P	LC 43/4	1996 045A			Molnyia 1-89	1600	EEO (62.9)	CIV	Russia
	1545	Aug 17	3	Soyuz U		B	LC 1	1996 047A	D	3/3	Soyuz TM-24	7150	STA (Mir)	CIV	Russia
	1546	Aug 29	12	Molniya 2BL		P	LC 43/3	1996 050B			Interbol 2	1300	EEO (62.9)	CIV	Russia
								1996 050A	A		Victor	30	LEO (62.8)	CIV	Argentina
								1996 050C	A		Magion 5	62	EEO (62.8)	CIV	Czech Rep.
	1547	Oct 24	56	Molniya ML	PVB71612-697	P	LC 43/4	1996 060A			Molniya 3-48	1750	EEO (62.9)	CIV	Russia
	1548	Nov 19	26	Soyuz U		B	LC 1	1996 066A	D		Progress M-33	7250	STA (Mir)	CIV	Russia
	1549	Dec 24	35	Soyuz U	PVB15000-050	P	LC 43/4	1996 073A			Bion 11	6000	LEO (62.8)	CIV	Russia
	1550	1997 Feb 10	48	Soyuz U		B	LC 1	1997 003A	D	3/2	Soyuz TM-25	7250	STA (Mir)	CIV	Russia
	1551	Apr 06	55	Soyuz U		B	LC 1	1997 014A	D		Progress M-34	7156	STA (Mir)	CIV	Russia
	1552	Apr 09	3	Molniya 2BL	PVB76032-647	P	LC 16/2	1997 015A			Kosmos 2340	1800	EEO (63.3)	MIL	Russia
	1553	May 14	35	Molniya 2BL		P	LC 43/4	1997 022A			Kosmos 2342	1800	EEO (63)	MIL	Russia
	1554	May 15	1	Soyuz U		B	LC 31	1997 024A			Kosmos 2343	6500	LEO (64.9)	MIL	Russia
	1555	Jul 05	51	Soyuz U		B	LC 1	1997 033A	D		Progress M-35	7150	STA (Mir)	CIV	Russia
	1556	Aug 05	31	Soyuz U		B	LC 1	1997 038A	D	2/3	Soyuz TM-26	7250	STA (Mir)	CIV	Russia
	1557	Sep 24	50	Molniya ML		P	LC 43/4	1997 054A			Molniya 1-90	1600	EEO (62.9)	CIV	Russia
	1558	Oct 05	11	Soyuz U		B	LC 1	1997 058A	D		Progress M-36	7195	STA (Mir)	CIV	Russia
	1559	Oct 09	4	Soyuz U		P	LC 43/3	1997 060A			Foton 11	6200	LEO (62.8)	CIV	Russia
	1560	Nov 18	40	Soyuz U		P		1997 072A			Resurs F-1M	5717	LEO (82.3)	CIV	Russia
	1561	Dec 15	27	Soyuz U		P		1997 080A			Kosmos 2348	6600	LEO (67.1)	MIL	Russia
	1562	Dec 20	5	Soyuz U		B	LC 1	1997 081A	D		Progress M-37	7040	STA (Mir)	CIV	Russia
	1563	1998 Jan 29	40	Soyuz U		B	LC 1	1998 004A	D	3/3	Soyuz TM-27	7150	STA (Mir)	CIV	Russia
	1564	Feb 17	19	Soyuz U		B	LC 31	1998 009A			Kosmos 2349	6600	LEO (70.4)	MIL	Russia
	1565	Mar 14	25	Soyuz U		B	LC 1	1998 015A	D		Progress M-38	7007	STA (Mir)	CIV	Russia
	1566	May 07	54	Molniya 2BL		P	LC 16/2	1998 027A			Kosmos 2351	1900	EEO (63)	MIL	Russia
	1567	May 14	7	Soyuz U		B	LC 1	1998 031A	D		Progress M-39	7450	STA (Mir)	CIV	Russia
	1568	Jun 24	41	Soyuz U		P	LC 43/3	1998 038A			Kosmos 2358	6600	LEO (67.1)	MIL	Russia
	1569	Jun 25	1	Soyuz U		B	LC 31	1998 039A			Kosmos 2359	6700	LEO (64.9)	MIL	Russia
	1570	Jul 01	6	Molniya ML		P	LC 43/3	1998 040A			Molniya 3-49	1600	EEO (62.8)	CIV	Russia
	1571	Aug 13	43	Soyuz U		B	LC 1	1998 047A	D	3/3	Soyuz TM-28	7150	STA (Mir)	CIV	Russia
	1572	Sep 28	46	Molniya ML		P	LC 43/3	1998 054A			Molniya 1-91	1600	EEO (62.9)	CIV	Russia
	1573	Oct 25	27	Soyuz U		B	LC 1	1998 062A	D		Progress M-40	7450	STA (Mir)	CIV	Russia
	1574	1999 Feb 09	107	Soyuz U/Ikar	ST01	B	LC 1	1999 004A	M		Globalstar 36	450	LEO (52)	CML	USA
								1999 004B	M		Globalstar 23	450	LEO (52)	CML	USA
								1999 004C	M		Globalstar 38	450	LEO (52)	CML	USA
								1999 004D	M		Globalstar 40	450	LEO (52)	CML	USA
	1575	Feb 20	11	Soyuz U	M15000-662	B	LC 1	1999 007A	D	3/3	Soyuz TM-29	7150	STA (Mir)	CIV	Russia
	1576	Mar 15	23	Soyuz U/Ikar	ST02	B	LC 1	1999 012A	M		Globalstar 22	450	LEO (52)	CML	USA

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload; D = Docking

FLIGHT HISTORY

	Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
							1999 012B	M		Globalstar 41	450	LEO (52)	CML	USA
							1999 012C	M		Globalstar 46	450	LEO (52)	CML	USA
							1999 012D	M		Globalstar 37	450	LEO (52)	CML	USA
1577	Apr 02	18	Soyuz U		B	LC 1	1999 016A	D		Progress M-41	7250	STA (Mir)	CIV	Russia
1578	Apr 15	13	Soyuz U/Ikar	ST03	B	LC 1	1999 019A	M		Globalstar 19	450	LEO (52)	CML	USA
							1999 020B	M		Globalstar 42	450	LEO (52)	CML	USA
							1999 020C	M		Globalstar 44	450	LEO (52)	CML	USA
							1999 020D	M		Globalstar 45	450	LEO (52)	CML	USA
1579	Jul 08	84	Molniya ML		P	LC 43/3	1999 036A			Molniya 3-50	1250	EEO (62.8)	CIV	Russia
1580	Jul 16	8	Soyuz U	?-667	B	LC 1	1999 038A	D		Progress M-42	7150	STA (Mir)	CIV	Russia
1581	Aug 18	33	Soyuz U		P	LC 43/4	1999 044A			Kosmos 2365	6700	LEO (67.1)	MIL	Russia
1582	Sep 09	22	Soyuz U		P	LC 43/4	1999 048A			Foton 12	6410	LEO (62.8)	CIV	Europe
1583	Sep 22	13	Soyuz U/Ikar	ST04 S15000-061	B	LC 1	1999 049A	M		Globalstar 30	450	LEO (52)	CML	USA
							1999 049B	M		Globalstar 50	450	LEO (52)	CML	USA
							1999 049C	M		Globalstar 55	450	LEO (52)	CML	USA
							1999 049D	M		Globalstar 58	450	LEO (52)	CML	USA
1584	Sep 28	6	Soyuz U		P	LC 43/4	1999 054A			Resurs F-1M 2	6300	LEO (82.3)	CIV	Russia
1585	Oct 18	20	Soyuz U/Ikar	ST05	B	LC 1	1999 058A	M		Globalstar 31	450	LEO (52)	CML	USA
							1999 058B	M		Globalstar 56	450	LEO (52)	CML	USA
							1999 058C	M		Globalstar 57	450	LEO (52)	CML	USA
							1999 058D	M		Globalstar 59	450	LEO (52)	CML	USA
1586	Nov 22	35	Soyuz U/Ikar	ST06	B	LC 1	1999 062A	M		Globalstar 29	450	LEO (52)	CML	USA
							1999 062B	M		Globalstar 34	450	LEO (52)	CML	USA
							1999 062C	M		Globalstar 39	450	LEO (52)	CML	USA
							1999 062D	M		Globalstar 61	450	LEO (52)	CML	USA
1587	Dec 27	35	Molniya M		P	LC 16/2	1999 073A			Kosmos 2368	1900	EEO (62.8)	MIL	Russia
1588	2000 Feb 01	36	Soyuz U		B	LC 1	2000 005A	D		Progress M1-1	7290	STA (Mir)	CIV	Russia
1589	Feb 08	7	Soyuz U/ Fregat	ST07/079	B	LC 31	2000 009A			DumSat	110	LEO (64.9)	CML	Russia
							2000 009B	A		IRDT	1110	LEO (64.8)	CML	Germany
1590	Mar 20	41	Soyuz U/ Fregat	ST08	B	LC 31	2000 015A			Dumsat	2382	EEO (64.9)	CIV	Russia
1591	Apr 04	15	Soyuz U		B	LC 1	2000 018A	D	2/2	Soyuz TM-30	7000	STA (Mir)	CIV	Russia
1592	Apr 25	21	Soyuz U		B	LC 1	2000 021A	D		Progress M1-2	7250	STA (Mir)	CML	Russia
1593	May 03	8	Soyuz U		B	LC 1	2000 023A			Kosmos 2370	6700	LEO (64.8)	MIL	Russia
1594	Jul 16	74	Soyuz U/ Fregat	ST09/069	B	LC 31	2000 041A	M		Samba (Cluster II FM7 C3)	1200	EEO (65)	CIV	Europe
							2000 041B	M		Salsa (Cluster II FM6 C2)	1200	EEO (65)	CIV	Europe
1595	Aug 06	21	Soyuz U		B	LC 1	2000 044A	D		Progress M1-3 (ISS-1P)	7100	STA (ISS)	CIV	Russia
1596	Aug 09	3	Soyuz U/ Fregat	ST10/070	B	LC 31	2000 045A	M		Rumba (Cluster II FM5 C1)	1200	EEO (65)	CIV	Europe
							2000 045B	M		Tango (Cluster II FM8 C4)	1200	EEO (65)	CIV	Europe
1597	Sep 29	51	Soyuz U		B	LC 31	2000 058A			Komos 2373	6600	LEO (70.4)	MIL	Russia
1598	Oct 16	17	Soyuz U		B	LC 1	2000 064A	D		Progress M-43	7250	STA (Mir)	CML	Russia
1599	Oct 31	15	Soyuz U		B	LC 1	2000 070A	D	3/3	Soyuz TM-31 (ISS-2R/1S)	7250	STA (ISS)	CIV	Russia
1600	Nov 16	16	Soyuz U		B	LC 1	2000 073A	D		Progress M1-4 (ISS-2P)	7285	STA (ISS)	CIV	Russia
1601	2001 Jan 24	69	Soyuz U	254	B	LC 1	2001 003A	D		Progress M1-5	2677	STA (Mir)	CIV	Russia
1602	Feb 26	33	Soyuz U		B	LC 1	2001 008A	D		Progress M-44 (ISS-3P)	7150	STA (ISS)	CIV	Russia
1603	Apr 28	61	Soyuz U		B	LC 1	2001 017A	D	3/3	Soyuz TM-32 (ISS-2S)	7000	STA (ISS)	CIV	Russia
1604	May 20	22	Soyuz FG		B	LC 1	2001 021A	D		Progress M1-6 (ISS-4P)	7150	STA (ISS)	CIV	Russia
1605	May 29	9	Soyuz U		P	LC43/4	2001 022A			Kosmos 2377	7200	LEO (67.1)	MIL	Russia
1606	Jul 20	52	Molniya M		P	LC43/4	2001 030A			Molniya 3		EEO (62.8)	MIL	Russia
1607	Aug 21	32	Soyuz U		B	LC 1	2001 036A	D		Progress M-45 (ISS-5P)	7150	STA (ISS)	CIV	Russia
1608	Sep 14	24	Soyuz U		B	LC 1	2001 041A	D		Progress M-CO1 (ISS-4R)	7130	STA (ISS)	CIV	Russia
1609	Oct 21	37	Soyuz U		B	LC 1	2001 048A	D	3/3	Soyuz TM-33 (ISS-3S)	7250	STA (ISS)	CIV	Russia
1610	Oct 25	4	Molniya 2BL		P	LC 43/3	2001 050A			Molniya 3-52	1250	EEO (62.8)	MIL	Russia

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FLIGHT HISTORY

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation		Pad	Payload Designation		Crew Up/Dn	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1611	Nov 26	32	Soyuz FG		B	LC 1	2001 051A	D		Progress M1-7 (ISS-6P)	7150	STA (ISS)	CIV	Russia
	2000							2001 051B	A		Kolibiri	27	LEO (51.6)	NGO	Russia
	1612	2002 Feb 25	91	Soyuz U		P	LC 43/3	2002 008A			Kosmos 2387	6700	LEO (67.1)	MIL	Russia
	1613	Mar 21	24	Soyuz FG		B	LC 1	2002 013A	D		Progress M1-8 (ISS-7P)		STA (ISS)	CIV	Russia
	1614	Apr 01	11	Molniya M		P	LC 43/3	2002 017A			Kosmos 2388	1900	EEO (62.93)	MIL	Russia
	1615	Apr 25	24	Soyuz U		B	LC 1	2002 020A	D	3/3	Soyuz TM-34 (ISS-4S)		STA (ISS)	CIV	Russia
	1616	Jun 26	62	Soyuz U		B	LC 1/5	2002 033A	D		Progress M-46 (ISS-8P)	7290	STA (ISS)	CIV	Russia
	1617	Sep 25	91	Soyuz FG		B		2002 045A	D		Progress M1-9 (ISS-9P)		STA (ISS)	CIV	Russia
F	**1618**	**Oct 15**	**20**	**Soyuz U**		**P**	**LC 43/3**	2002 F02A			Foton M-1	6425	LEO ()	CIV	Russia
	1619	Oct 30	15	Soyuz FG		B	LC 1	2002 050A	D	3/3	Soyuz TMA-1 (ISS-5S)	7250	STA (ISS)	CIV	Russia
	1620	Dec 24		Molniya M		P	LC 16/2	2002 059A			Kosmos 2393		EEO (62.3)	MIL	Russia
	1621	2003 Feb 02	40	Soyuz U	E15000-680	B	LC 1/5	2003 006A	D		Progress M-47 (ISS-10P)	7150	STA (ISS)	CIV	Russia
	1622	Apr 02	59	Molniya M		P		2003 011A			Molniya 1-92	1660	EEO (63)	MIL	Russia
	1623	Apr 26	24	Soyuz FG	E15000-005	B	LC1/5	2003 016A	D	2/3	Soyuz TMA-2 (ISS-6S)	7136	STA (ISS)	CIV	Russia
	1624	Jun 02	37	Soyuz FG/ Fregat	005/1005	B	LC 31	2003 022A			Mars Express	1042	Mars	CIV	Europe
						A					Beagle 2	60	Mars	CIV	Europe
	1625	Jun 08	6	Soyuz U		B	LC 1	2003 025A	D		Progress M1-10 (ISS-11P)	7250	STA (ISS)	CIV	Russia
	1626	Jun 19	11	Molniya M		P	LC43/3	2003 029A			Molniya 3-53	1600	EEO (63)	MIL	Russia
	1627	Aug 12	54	Soyuz U		B	LC31/6	2003 035A			Kosmos 2399	6750	LEO (65)	MIL	Russia
	1628	Aug 29	17	Soyuz U		B	LC 1	2003 039A	D		Progress M-48 (ISS-12P)	7283	STA (ISS)	CIV	Russia
	1629	Oct 18	50	Soyuz FG		B	LC 1	2003 047A	D	3/3	Soyuz TMA-3 (ISS-7S)	7200	STA (ISS)	CIV	Russia
	1630	Dec 27	70	Soyuz FG/ Fregat	084 / 1006	B	PL31/6	2003 059A			AMOS 2	1374	GTO	CML	Israel

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Flight History

Failure Descriptions:

F	1958 Apr 27	Sputnik	1958 F03	Vehicle destroyed at T+95 s because of first-stage pogo effects.
F	1958 Sep 23	Vostok L	1958 F15	Vehicle broke up 93 s after launch, because of pogo effects.
F	1958 Oct 11	Vostok L	1958 F17	Vehicle broke up 104 s after launch, because of pogo effects.
F	1958 Dec 04	Vostok L	1958 F20	Second-stage engine failed 245 s after launch.
F	1959 Jul 18	Vostok L	1959 F06	Second-stage control system failure 153 s after launch.
F	1960 Apr 15	Vostok L	1960 F05	Third-stage engine shutdown 726 s after launch.
F	1960 Apr 19	Vostok L	1960 F06	Strap-on booster Block B engine acheived only 75% thrust at ignition, and broke away from the core at T+0.4 s.
F	1960 Jul 28	Vostok L	1960 F09	Strap-on booster Block B engine failed at T+17 s, booster broke away, and the vehicle exploded.
F	1960 Oct 10	Molniya	1960 F12	During second-stage flight, resonance vibration occurred on the third stage causing damage to the gyro, causing loss of control at T+309 s and automatic shutdown at T+324 s.
F	1960 Oct 14	Molniya	1960 F14	Third-stage engine failed to ignite because of a LOX leak that froze kerosene in the fuel pump inlet.
F	1960 Dec 22	Vostok L	1960 F19	Third-stage malfunction 432 s after launch.
F	1961 Feb 04	Molniya	1961 002A	An electrical transformer, which powered the command timer, failed to initiate fourth-stage ignition.
F	1961 Dec 11	Vostok K	1961 F14	Third stage shut down early.
F	1962 Jun 01	Vostok	1961 F06	Strap-on booster Block B engine shut down at T+2 s and broke away from vehicle because of power failure in engine control system.
F	1962 Aug 25	Molniya	1962 040A	One of four solid settling motors on the fourth stage failed to fire. The stage began to tumble 3 s after ignition and shutdown after only 45 s when propellant slosh cut off flow to the engines.
F	1962 Sep 01	Molniya	1962 043A	Fourth stage failed to ignite.
F	1962 Sep 12	Molniya	1962 045A	Third-stage vernier engine exploded at shutdown when LOX valve failed to close. Fourth stage proceeded but failed at ignition because of oxidizer pump cavitation.
F	1962 Oct 24	Molniya	1962 057A	Fourth stage failed 16 s after ignition when loss of lubrication caused a turbopump shaft to jam.
F	1962 Nov 04	Molniya	1962 062A	An anomaly in the second stage pressurization system caused cavitation in the LOX pump after T+262 and fuel pump after T+292. The second stage continued to function, but vibration damaged a fuse in the fourth stage that prevented it from igniting.
F	1963 Jan 04	Molniya L	1963 001A	A power transformer failure prevented fourth-stage ignition.
F	1963 Feb 03	Molniya L	1963 F01	Failure of the fourth-stage gyro and torque sensor at T+105 caused loss of control at T+295.
F	1963 Jul 10	Vostok	1963 F10	Strap-on booster Block V hydrogen peroxide valve in gas generator shut down at T-2 s because of power failure in engine control system, causing engine shutdown. Vehicle lifted off, Block V strap-on broke loose, destroying launch vehicle.
F	1963 Nov 11	Molniya	1963 044A	Fourth stage lost attitude control during parking orbit coast and ignited with incorrect attitude.
F	1963 Nov 28	Vostok	1963 F15	Third-stage engine failed to ignite.
F	1964 Feb 19	Molniya M	1964 F01	A third-stage LOX leak froze fuel in the fuel feedline causing an explosion.
F	1964 Mar 21	Molniya M	1964 F03	Third-stage vernier engines did not produce full thrust when ignited at T+290 s because of lox valve failure. The main engine shut down prematurely at T+486 s and the vehicle did not achieve orbit.
F	1964 Mar 27	Molniya M	1964 014A	A power failure in fourth-stage ACS caused loss of control in parking orbit. .
F	1964 Apr 20	Molniya M	1964 F05	Power failure caused failure of control system and third-stage engine shutdown at T+340 s.
F	1964 Jun 04	Molniya	1964 F07	Second-stage throttle servomotor failed at T+104 s and the vehicle crashed.
S	1964 Oct 28	Kosmos 50	1964 070A	Satellite exploded into 97 pieces on eighth day in orbit.
S	1965 Feb 22	Kosmos 57	1965 012A	Satellite exploded in orbit.
F	1965 Mar 12	Molniya L	1965 018A	Fourth stage failed to ignite after the transformer failed.
F	1965 Apr 10	Molniya L	1965 F05	Third-stage engine failed because of failure of the nitrogen pressurization system for the LOX tank.
F	1965 Jul 13	Vostok	1965 F08	Pitch control malfunctioned during first-stage flight, causing automatic flight termination.
F	1965 Nov 23	Molniya M	1965 094A	Third-stage engine combustion chamber exploded at T+528 s because of rupture of fuel line, leading to abnormal stage separation, which prevented Block L fourth stage from operating.
F	1966 Mar 01	Molniya M	1966 017A	Block L fourth-stage attitude control failed in park orbit, preventing stage ignition.
F	1966 Mar 27	Molniya M	1966 F04	Vehicle failed during ascent.
F	1966 May 17	Voskhod	1966 F06	Failure to orbit.
F	1966 Jun 17	Voskhod	1966 054A	Fourth stage failed to ignite.
F	1966 Sep 16	Vostok	1966 F09	Strap-on Block D engine failed.
F	1966 Dec 14	Soyuz	1966 F12	Launch aborted at engine start because of failure of core-stage oxidizer valve. As the support structures were being put back in place for de-fueling, the launch escape system triggered the solid propellant escape rockets, carrying the Soyuz capsule away from the pad, but igniting the third-stage propellants in the process. The launch vehicle was destroyed in an explosion which severly damaged the pad and killed one person. The escape system had not been shut down after the abort and had sensed a 7-deg change in vehicle attitude, either because the support structure struck the vehicle causing it to veer, or because the earth's rotation as measured by the inertial gyros had rotated the vehicle out of its initial orientation.
F	1967 Apr 12	Voskhod	1967 033A	Failure.
F	1967 Jun 20	Voskhod	1967 F06	Failure to orbit.
F	1967 Jul 21	Voskhod	1967 F08	Failure to orbit.
F	1967 Sep 01	Voskhod	1967 F09	Failed to orbit.
F	1968 Feb 07	Molniya M	1968 F01	Stage 3 engine shut down prematurely at T+524 s because of excessive gas generator fuel consumption.
F	1969 Feb 01	Vostok M	1969 F03	Second-stage failure.
F	1970 Jul 21	Voskhod	1970 F06	Failure to orbit.
F	1970 Aug 22	Molniya M	1970 065A	Fourth-stage engine ignited, but shut down prematurely.
F	1971 Mar 05	Voskhod	1971 F03	Failure to orbit.
F	1971 Jun 25	Voskhod	1971 F05	Failure to orbit.
F	1971 Aug 19	Voskhod	1971 F09	Failure to orbit.

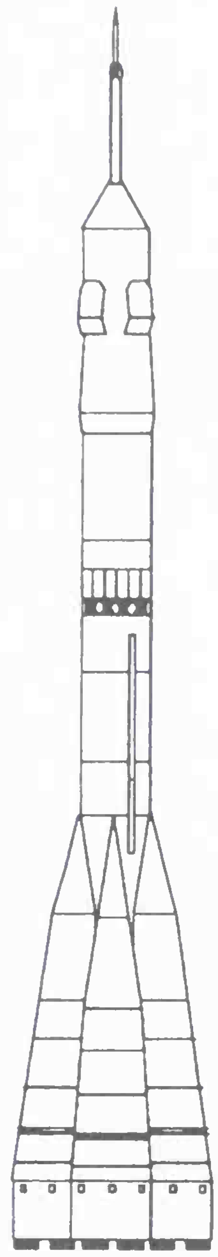
Flight History

Failure Descriptions (continued):

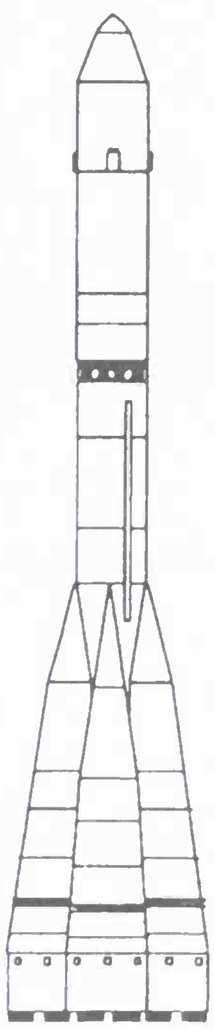
F	1971 Dec 03	Voskhod	1971 F12	Failure to orbit.
F	1972 Mar 31	Molniya M	1972 023A	Fourth-stage engine ignited, but shut down prematurely.
F	1972 Sep 02	Voskhod	1972 F05	Failure to orbit.
F	1973 Jul 04	Voskhod	1973 F05	Failure to orbit.
F	1974 Apr 12	Voskhod	1974 F02	Failure to orbit.
F	1974 May 23	Soyuz U	1974 F03	Failure to orbit.
S	1974 Aug 26	Soyuz 15	1974 067A	Failed to rendezvous successfully with Salyut 3.
F	1974 Aug 30	Voskhod	1974 F06	Failure to orbit.
F	1975 Apr 05	Soyuz	1975 F02	The core stage failed to fully separate after burnout. The additional mass prevented the third-stage burn from performing properly. Since the escape tower had already been jetisonned, the cosmonauts had to separate their capsule from the rocket and reentered roughly 1600 km downrange, near the Chinese border.
F	1976 Jul 01	Molniya M	1976 062A	Fourth-stage engine ignited, but shut down prematurely.
S	1976 Jul 22	Kosmos 844	1976 072A	Satellite exploded, 294 pieces.
F	1976 Sep 01	Molniya M	1976 088A	Fourth-stage engine ignited, but shut down prematurely.
F	1976 Oct 04	Soyuz U	1976 F02	Failure to orbit.
F	1977 Feb 22	Soyuz U	1977 F01	Failure to orbit.
F	1977 Aug 10	Soyuz U	1977 F03	Failure to orbit.
F	1978 Dec 23	Vostok M	1978 121A	Failure.
F	1979 Feb 16	Soyuz U	1979 F01	Failure to orbit.
S	1979 Apr 10	Soyuz 33	1979 029A	Failed to dock with Salyut 6.
S	1979 Aug 28	Kosmos 1124	1979 077A	Satellite exploded 9 September 1979 into eight pieces.
F	1979 Oct 12	Soyuz U	1979 F04	Failure to orbit.
F	1980 Feb 12	Molniya M	1980 013A	The satellite and fourth stage were tracked in a 62.8 deg, 212×570 km parking orbit prior for delivery to a highly eliptical orbit, but subsequently neither the fourth stage, nor the satellite, nor any debris could be found. It is suspected that a fourth-stage failure accidentally deorbited the spacecraft and upper stage.
F	1980 Mar 18	Vostok M		30 min prior to launch, an explosion destroyed the vehicle and killed 51 people. The investigation commission blamed a member of the ground crew who had not properly fixed a LOX leak in the third-stage fueling systems. However, it was later recognized that incompatible materials in the hydrogen peroxide loading lines caused decomposition of the hydrogen peroxide in the vehicle tanks, resulting in the explosion. Following the disaster, vehicles and ground systems were modified to prevent future explosions. The modified verions of Vostok, Soyuz and Molniya have PVB appended to their article numbers (Pozharo-Vzryvo-Bezopasnaya, "Fire-Explosion-Proof").
F	1980 Apr 18	Molniya M	1980 031A	Fourth-stage engine ignited, but shut down prematurely.
F	1981 Mar 28	Soyuz U	1981 F02	Failure to orbit.
F	1981 Sep 11	Molniya M	1981 088A	Fourth-stage engine ignited, exploding partway through burn.
F	1982 May 15	Soyuz U	1982 F02	Failure to orbit.
F	1982 Jun 12	Soyuz U	1982 F03	Failure to orbit.
F	1982 Dec 08	Molniya M	1982 115A	Fourth-stage engine ignited, exploding partway through burn.
S	1983 Apr 20	Soyuz T-8	1983 035A	Failed to dock with Salyut 7.
F	1983 Jul 08	Molniya 2BL	1983 070A	Upper stage exploded.
F	1983 Sep 26	Soyuz U	1983 F02	At T–90 s, prior to launch, a fire started at the base of the vehicle. Shortly before the launch vehicle exploded, the escape tower fired, pulling the Soyuz capsule to a height of 950 m. The capsule then parachuted to a landing roughly 2.5 km from the pad, saving the cosmonauts. Fuel and debris continued to burn on the pad for 20 h.
F	1986 Mar 26	Soyuz U	1986 F02	Failure to orbit.
F	1986 Oct 03	Molniya 2BL	1986 075A	Fourth-stage engine ignited, but shut down prematurely.
F	1987 Jun 18	Soyuz U	1987 F04	Failure to orbit.
S	1987 Jul 09	Kosmos 1866	1987 059A	Engine failed to shut down, propellant depleted prematurely; exploded 26 July 1987.
F	1988 Jul 09	Soyuz U2	1988 F02	Failure to orbit.
F	1988 Jul 27	Soyuz U	1988 F04	Failure to orbit.
F	1988 Nov 11	Soyuz U2	1988 F05	Failure to orbit.
F	1990 Apr 03	Soyuz U	1990 F02	Failure to orbit.
F	1990 Jun 21	Molniya 2BL	1990 055A	Fourth-stage failure.
F	1990 Jul 03	Soyuz U	1990 F03	Failure to orbit.
F	1993 Apr 27	Soyuz U	1993 028A	Spacecraft was delivered to planned orbit, but at the intended separation time an explosion occurred in the upper stage, damaging the spacecraft.
F	1996 May 14	Soyuz U	1996 F02	Payload shroud broke up 49 s into flight. The flight continued until separation of the strap-on boosters, when the vehicle veered off course causing the main engines to automatically shut down and the vehicle to crash.
F	1996 Jun 20	Soyuz U	1996 F04	Payload shroud failed 50 s into flight, causing flight termination and impact 8 km from the pad. Investigation concluded that this and the previous failure were due to a defective manufacturing process that resulted in weaker glue bonds between layers of glass-reinforced-plastic in the fairing structure.
F	2002 Oct 15	Soyuz U	2002 F02	Contamination in the hydrogen peroxide system of the Block D strap-on booster caused an engine failure about 8–9 s after liftoff. The booster broke away from the vehicle and crashed near the pad. The safety system shut down the remaining engines at T+29 s and the vehicle crashed about 1 km from the pad. (Shutdown may have occurred at T+20 s with impact at T+29 s.) One soldier was killed and eight others injured by a subsequent explosion when they were sent to extinguish fires.

Vehicle Design

Overall Vehicle

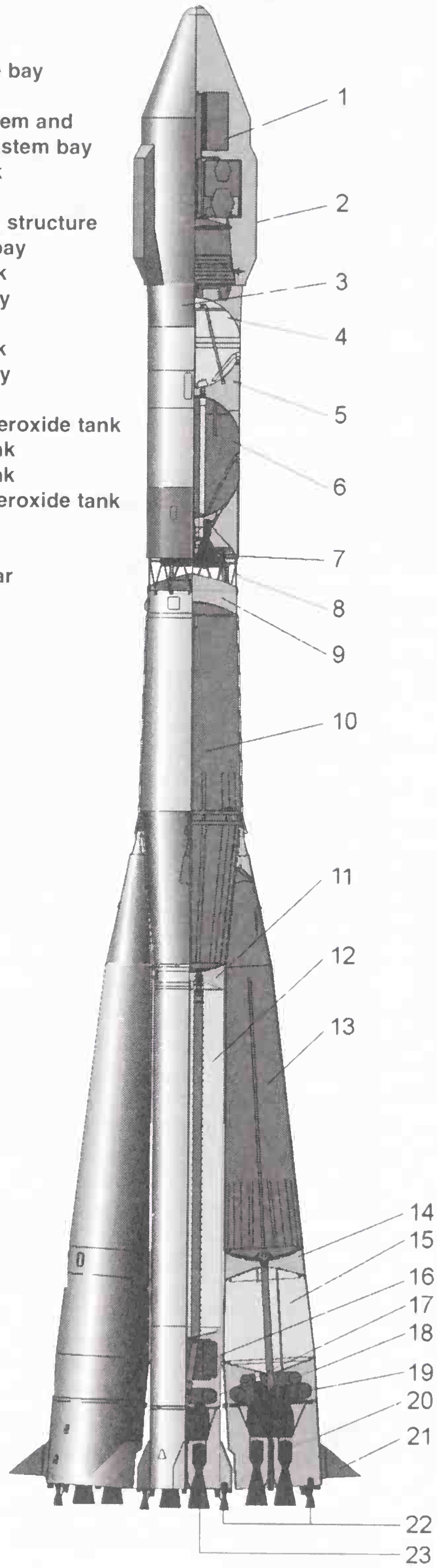


Soyuz U or FG



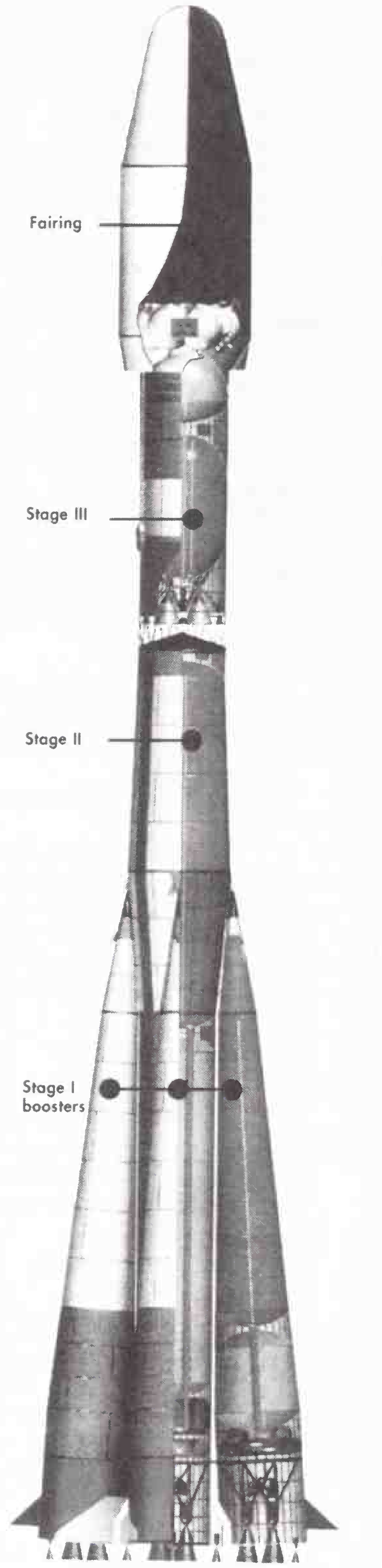
Molniya

1. Spacecraft
2. Fairing
3. Intermediate bay
4. Fuel tank
5. Control system and telemetry system bay
6. Oxygen tank
7. Main engine
8. Intermediate structure
9. Equipment bay
10. Oxygen tank
11. Intertank bay
12. Fuel tank
13. Oxygen tank
14. Intertank bay
15. Fuel tank
16. Hydrogen peroxide tank
17. Nitrogen tank
18. Nitrogen tank
19. Hydrogen peroxide tank
20. Main engine
21. Air rudder
22. Steering gear
23. Main engine



Courtesy Starsem.

Soyuz

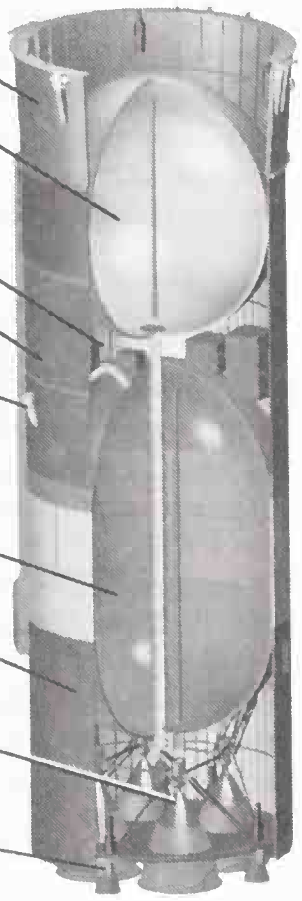


Courtesy Starsem.

Soyuz 2

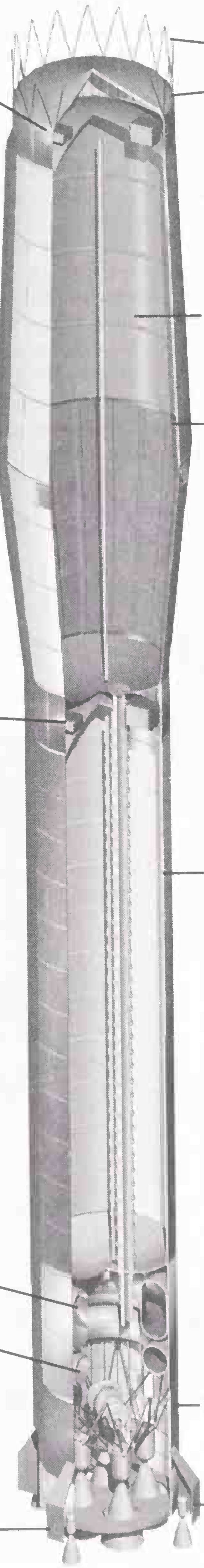
	Molniya	Soyuz U or FG	Soyuz 2
Height	43.44 m (142.5 ft)	50.67 m (166.2 ft) w/ Soyuz capsule	46.1 m (151.2 ft)
Gross Liftoff Mass	305 t (672 klbm)	310 t (683 klbm)	305 t (672 kblm)
Thrust at Liftoff	4031.9 kN (906.41 klbf)	Soyuz U: 4031.9 kN (906.41 klbf) Soyuz FG: 4144 kN (931.8 klbf)	4144.7 kN (931.77 klbf)

Vehicle Design



Courtesy Starsem.

Soyuz Third Stage, Block I



Courtesy Starsem.

Soyuz Second Stage, Block A

Courtesy Starsem.

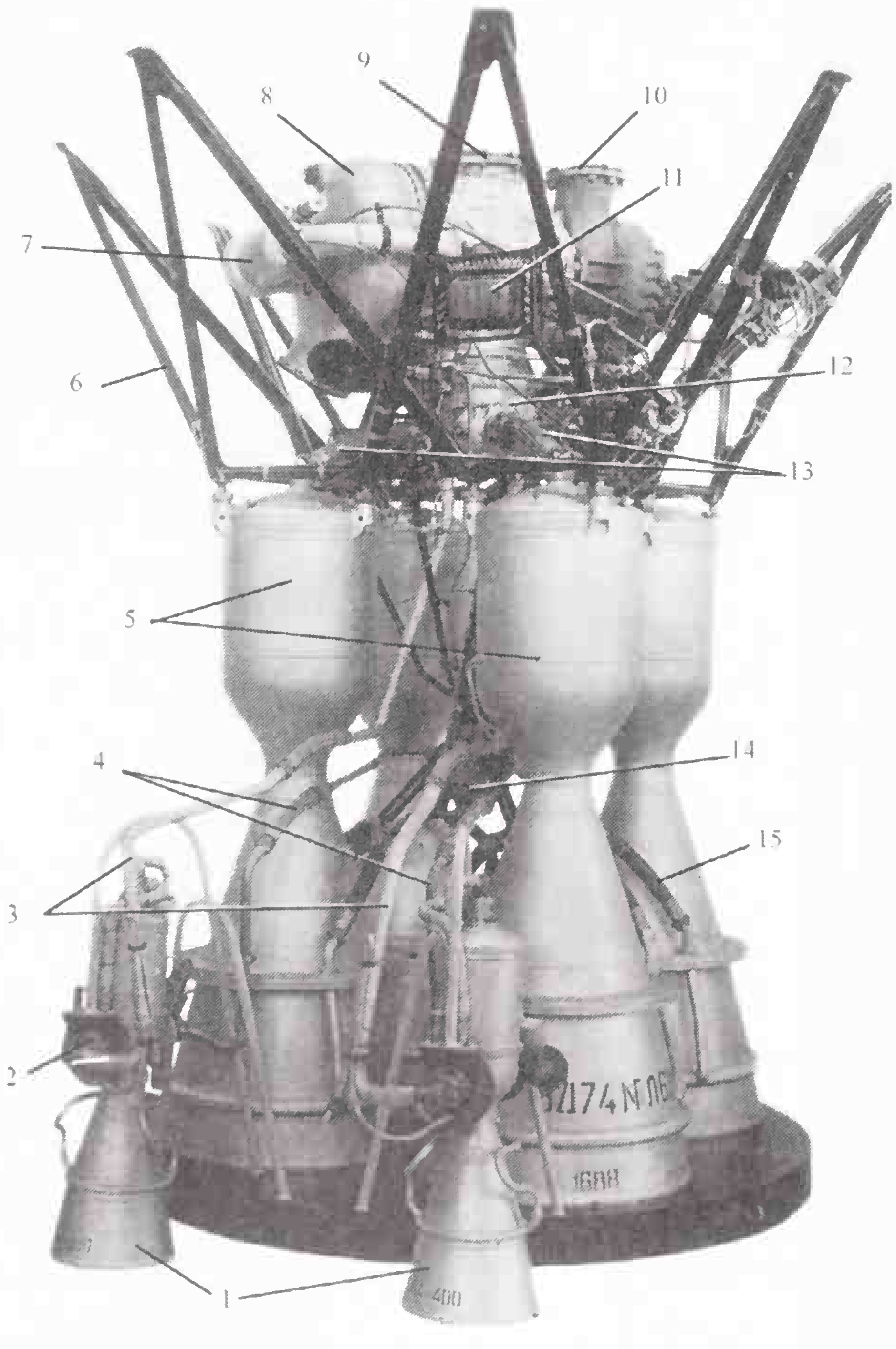
Soyuz First Stage, B, V, G, D Blocks

VEHICLE DESIGN

Stages 1 and 2

Four conical strap-on boosters are clustered around a core stage for all Soyuz family configurations. While the boosters and core stage all ignite for liftoff, the boosters are designated in the Russian nomenclature as the first stage, and the lowest core stage is designated as the second stage. This is counterintuitive to Western analysts who have often labeled the core stage the first stage. The conical boosters transmit thrust to the core stage through a ball and socket joint at the top of the cone. Aft struts connect the boosters to the core. All stages used in the Soyuz family (with the recent exception of two new upper stages) use LOX and kerosene propellant. The second stage can use either standard grade kerosene referred to as T-1, or a synthetic hydrocarbon called syntin, which is based on cyclopropane and is slightly denser and provides more thrust. Manned flights used the Soyuz U2 variant, which was distinguished from the common Soyuz U version by the use of engines modified to burn syntin for higher performance. However, in the mid 1990s use of syntin ended because of the high cost of production.

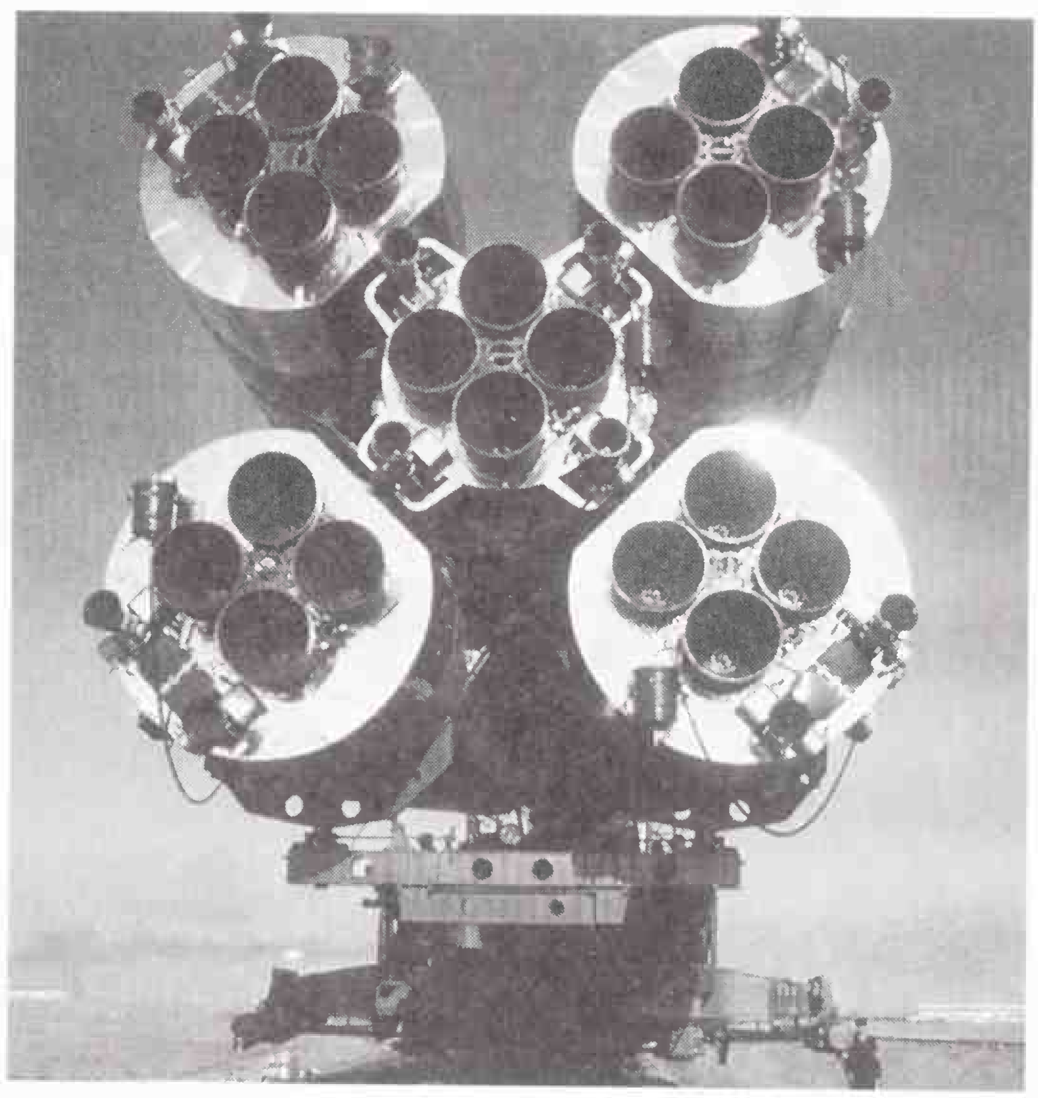
The first and second stages are powered by the closely related RD-107 and RD-108 engines on the Soyuz U and Molniya-M, or the modernized RD-107A and RD-108A engines on the Soyuz FG and Soyuz 2. The engines use an open cycle gas generator powered by catalytic decomposition of hydrogen peroxide in an 82% solution to power two main turbopumps on a single shaft and two auxiliary pumps. The two main pumps (one for oxidizer and one for fuel) feed propellant to four thrust chambers. One auxiliary pump supplies the hydrogen peroxide to the gas generator, while the other is used to force liquid nitrogen through a heat exchanger, and send the resulting nitrogen gas to pressurize the fuel tanks. The hydrogen peroxide and liquid nitrogen are stored in toroidal tanks just above the engines of the booster and second stage. The RD-107 booster engines have two small gimbaled vernier nozzles to provide pitch, yaw, and roll control. The second stage RD-108 engine has four of these verniers, and also operates at a lower chamber pressure and temperature to provide longer burn time. Counting the boosters and core stage, there are 20 main engine thrust chambers and 12 vernier engine thrust chambers firing at liftoff of every Soyuz or Molniya flight. Considering the high flight rate of the Soyuz/Molniya family, this indicates that the Soviet Union produced an average of three Soyuz main engine thrust chambers per day for more than a decade. The main engines can be throttled for at least two different levels of thrust. The engines operate at less than full throttle on the launch pad to verify successful operation before liftoff. The booster engines (but not the second stage engines) are also throttled back to about 84% of full throttle before separation. To launch the new, heavier Soyuz TMA series spacecraft to the International Space Station, the new Soyuz FG has RD-107A and RD-108A engines with modified injectors and an adjusted mixture ratio to increase thrust and specific impulse. These provide a 5% increase in payload capability. The same engines will be used on Soyuz 2.



LPRE RD-107A: 1 —vernier chambers, 2 —unit of the vernier chamber turn, 3 —vernier chambers oxidizer pipelines, 4 —vernier chambers fuel pipelines; 5 —main chambers; 6 —frame of attachment to the LV; 7 —steam and gas generator; 8 —turbine; 9 —oxidizer pump, 10 —fuel pump, 11 —throttling system pressure transducer, 12 —oxidizer main valve; 13 —main chambers oxidizer pipelines; 14 —fuel main valve; 15 — main chamber fuel pipeline

Courtesy Starsem.

RD-107A Engine



Courtesy Space Commerce Corporation.

View of 20 main and 12 vernier chambers on the base of the vehicle.

Vehicle Design

Stages 1 and 2

Note: The following data come from a variety of sources, which often disagree slightly and express values in different ways (for example, they may count the mass of hydrogen peroxide in the propellant mass, only in the gross mass, or not at all). Additionally, vehicles used for different missions have slightly different hardware configurations and propellant loads. For example, the reported propellant mass of the Block A stage ranges from 86.5 to 95.4 t (190.7–210.3 klbm). These differences and inconsistencies make it very difficult to assemble a consistent data set. Wherever possible, data directly from TsSKB or Starsem were used. The propellant mass fraction calculation treats the gas generator H_2O_2 as part of the propellant.

	Soyuz U / Molniya		Soyuz FG and Soyuz 2	
	Block B, V, G and D (First-Stage Strap-on Boosters)	**Block A (Second-Stage Core)**	**Block B, V, G, and D (First-Stage Strap-on Boosters)**	**Block A (Second-Stage Core)**
Dimensions				
Length	19.6 m (64.3 ft)	28 m (92 ft)	19.6 m (64.3 ft)	28 m (92 ft)
Diameter	2.68 (8.8 ft)	2.15–2.95 m (7.1–9.7 ft)	2.68 (8.8 ft)	2.15–2.95 m (7.1–9.7 ft)
Mass				
Propellant Mass	39.2 t (86.4 klbm)	90.1 t (198.6 klbm)	39.2 t (86.4 klbm)	90 t (198 klbm)
Other Fluids	H_2O_2: 1.19 t (2.62 klbm) LN_2: 280 kg (620 lbm)	H_2O_2: 2.6 t (5.73 klbm) LN_2: 520 kg (1145 lbm)	H_2O_2: 1.19 t (2.62 klbm) LN_2: 280 kg (620 lbm)	H_2O_2: 2.6 t (5.73 klbm) LN_2: 520 kg (1145 lbm)
Inert Mass	3784 kg (8342 lbm)	6875 kg (15,160 lbm)	3810 kg (8342 lbm)	Soyuz FG: 6550 kg (14,440 lbm) Soyuz 2: 6450 kg (14,220 lbm)
Gross Mass	44.5 t (98.1 klbm)	101.9 t (224.6 klbm)	44.4 t (98.0 klbm)	99.4–99.5 t (219.1–219.3 klbm)
Propellant Mass Fraction	0.91	0.91	0.91	0.93
Structure				
Type	Skin-stringer	Skin-stringer	Skin-stringer	Skin-stringer
Material	Aluminum	Aluminum	Aluminum	Aluminum
Propulsion				
Engine Designation	RD-107 (NPO Energomash)	RD-108 (NPO Energomash)	RD-107A (NPO Energomash)	RD-108A (NPO Energomash)
Number of Engines	1 engine with 4 thrust chambers and 2 verniers	1 engine with 4 thrust chambers and 4 verniers	1 engine with 4 thrust chambers and 2 verniers	1 engine with 4 thrust chambers and 4 verniers
Propellant	LOX/kerosene T-1	LOX/kerosene T-1	LOX/kerosene T-1	LOX/kerosene T-1
Average Thrust	Sea level: 813.2 kN (182.8 klbf) Vacuum: 991.7 kN (222.9 klbf)	Sea level: 778.9 kN (175.1 klbf) Vacuum: 997.1 kN (224.1 klbf)	Sea level: 838.3 kN (188.5 klbf) Vacuum: 1021.1 kN (229.5 klbf)	Sea level: 791.7 kN (178.0 klbf) Vacuum: 989.8 kN (222.5 klbf)
Isp	Sea Level: 245 s Vacuum: 310 s	Sea level: 264 Vacuum: 311	Sea level: 262 s Vacuum: 319 s	Sea level: 255 s Vacuum: 319s
Chamber Pressure	58.5 bar (848 psi)	51.0 bar (740 psi)	?	?
Nozzle Expansion Ratio	?	?	?	
Propellant Feed System	Gas generator turbopump	Gas generator turbopump	Gas generator turbopump	Gas generator turbopump
Mixture Ratio (O/F)	2.47:1	2.39:1	?	?
Throttling Capability	84% or 100%	2 levels	84% or 100%	2 levels
Restart Capability	No	No	No	No
Tank Pressurization	Warm nitrogen gas	Warm nitrogen gas	Warm nitrogen gas	Warm nitrogen gas
Attitude Control				
Pitch, Yaw	2 verniers per booster hydraulically gimbaled ±45 deg, and steerable aerofins	4 hydraulically gimbaled verniers	2 verniers per booster hydraulically gimbaled ±45 deg, and steerable aerofins	4 hydraulically gimbaled verniers
Roll	2 verniers per booster hydraulically gimbaled ±45 deg, and steerable aerofins	4 hydraulically gimbaled verniers	2 verniers per booster hydraulically gimbaled ±45 deg, and steerable aerofins	4 hydraulically gimbaled verniers
Staging				
Nominal Burn Time	120 s	286 s	120 s	286 s
Shutdown Process	Command shutdown	Command shutdown	Command shutdown	Command shutdown
Stage Separation	Pyrotechnic bolts, aerodynamic separation	Pyrotechnic bolts, stage 3 ignition	Pyrotechnic bolts, aerodynamic separation	Pyrotechnic bolts, stage 3 ignition

Vehicle Design

Third Stage

The Block I third stage of Soyuz and Molniya consists of two independent tanks for LOX and kerosene, an intermediate bay at the top of the stage, which supports the payload fairing and/or upper stage, an intertank section, which houses the avionics, and a tail section, which houses the engine. Following jettison of the second stage, the interstage structure around the third-stage engine separates into three parts and is also jettisoned. The oxygen tank is pressurized by oxygen heated to gas in the engine heat exchanger. The fuel tank is pressurized by cooled exhaust gas from the engine gas generator. The majority of the exhaust gas is expelled through four steering verniers that can each rotate ±40 deg in one plane. Following release of the upper stage or payload, the third stage backs away by venting residual oxygen from the tank through a forward pointing nozzle.

The largest modifications for Soyuz 2 are made in the third stage. The old RD-0110 engine is to be replaced by the RD-0124 engine, which has been in development since 1993. It is designed to have dimensions, interfaces, and thrust similar to the RD-0110, but by using an oxygen-rich staged-combustion cycle the specific impulse is increased by almost 35 s. The new engine alone provides a 950-kg (2100-lbm) increase in performance for Soyuz 2. Although this engine was designed specifically for Soyuz, it has also been selected for Aurora and Angara as well.

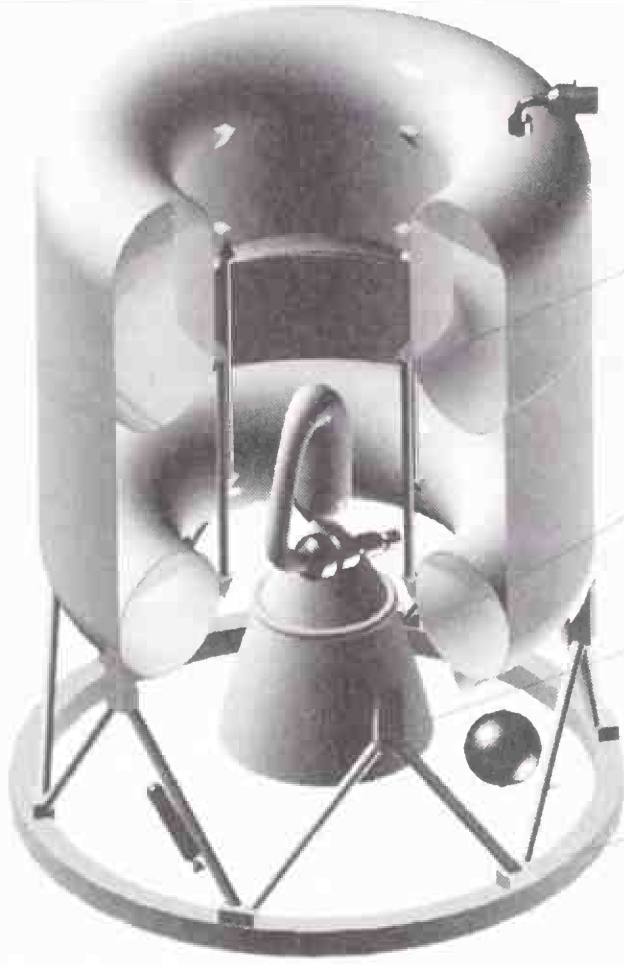
Because the oxygen-rich cycle of the RD-0124 requires more LOX than the older engine, the tanks of the third stage must be modified accordingly. To maintain the same overall dimensions, the upper kerosene tank will be given a more compact ellipsoidal bulkhead while the lower LOX tank will be stretched to take advantage of the additional space. The RD-0124 engine will not be implemented until the second step of the Soyuz 2 project – the Soyuz 2-1B configuration. However, the tank modifications will be made first, on the Soyuz 2-1A. The third stage will also be enhanced structurally to handle the higher loads imparted by the larger payload fairing planned for the Soyuz 2.

	Block I (Soyuz U, FG and Molniya)	Upgraded Block I (Soyuz 2-1B)
Dimensions		
Length	6.7 m (22.0 ft)	6.7 m (22.0 ft)
Diameter	2.66 m (8.73 ft)	2.66 m (8.73 ft)
Mass		
Propellant Mass	21.4–22.9 t (47.2–50.5 klbm)	22.8 t (50.3 klbm)
Inert Mass	2355 kg (5190 lbm)	2470 kg (5445 lbm)
Gross Mass	25.2 t (55.6 klbm)	25.3 t (55.8 klbm)
Propellant Mass Fraction	0.91	0.90
Structure		
Type	Skin-stringer	Skin-stringer
Material	Aluminum	Aluminum
Propulsion		
Engine Designation	RD-0110/RD-461 (KB Khimavtomatiki)	RD-0124 (KB Khimavtomatiki)
Number of Engines	1 with 4 thrust chambers and 4 verniers	1 with 4 thrust chambers
Propellant	LOX/kerosene	LOX/kerosene
Average Thrust	298 kN (67 klbf) including 6 kN (1.4 klbf) verniers	294.3 kN (66.16 klbf)
Isp	325 s	359 s
Chamber Pressure	68 bar (986 psi)	162 bar (2350 psi)
Nozzle Expansion Ratio	82.2:1	?
Propellant Feed System	Gas generator turbopump	Staged combustion turbopump
Mixture Ratio (O/F)	2.2:1	2.6:1
Throttling Capability	Two levels	Two levels?
Restart Capability	No	No?
Tank Pressurization	Oxidizer: heated oxygen gas Fuel: cooled gas generator exhaust	Oxidizer: heated oxygen gas? Fuel: cooled gas generator exhaust?
Attitude Control		
Pitch, Yaw	4 verniers with single axis gimbal ±40 deg	Main engine nozzle gimbal
Roll	4 verniers with single axis gimbal ±40 deg	Main engine nozzle gimbal
Staging		
Nominal Burn Time	230–250 s	300 s
Shutdown Process	Command shutdown	Command shutdown
Stage Separation	Spring ejection	Spring ejection

Vehicle Design

Upper Stages

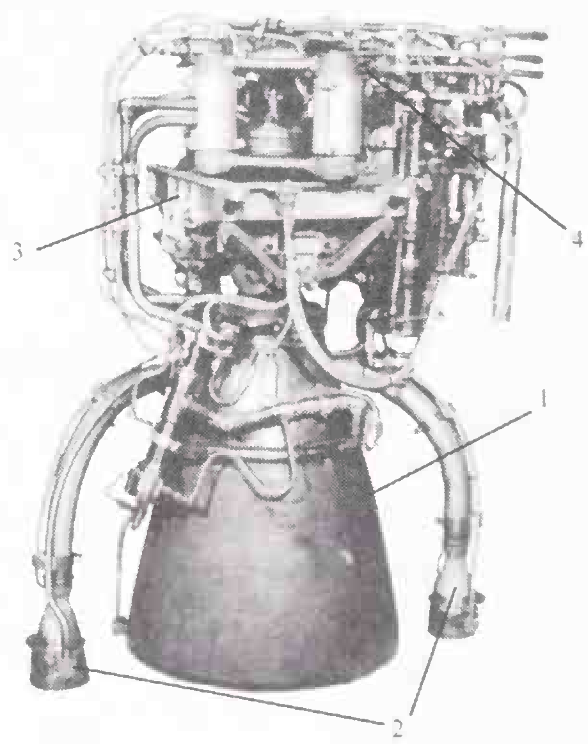
The Block L stage (also called Molniya) has been used as the fourth stage of the Molniya launch vehicle since 1960. Its presence is the primary difference between the Molniya and the three-stage Soyuz U. It was originally used to send early Soviet planetary spacecraft to the moon, Mars, and Venus, although this use ended with the introduction of the larger Proton K/Block D launch vehicle. It is now used to deliver payloads to highly elliptical Earth orbits. The stage consists of a pair of toroidal LOX and kerosene tanks, with the engine located in the center. The upper oxidizer tank and lower fuel tank are separated by a cylindrical shell welded to the tanks. In this space between the tanks, the power supply, control system, and other components are mounted. Attitude control is provided by gimbaling the main engine, and by four vernier engines. Before main engine ignition, two solid rocket motors are fired to settle the propellants, and are then jettisoned. The Block L has demonstrated mediocre reliability and is the primary cause of Molniya failures. In addition, its main engine cannot be restarted. Therefore, new upper stages are being introduced to address these limitations. An upgraded version, the Block LM, has been designed, which would use the restartable 11D58M engine from Proton's Block DM upper stage, and would burn LOX/syntin. However, there is no indication that this stage will be adopted, and it appears the Block L will instead be replaced by the Fregat stage.



Courtesy Starsem.

Molniya Block L Stage

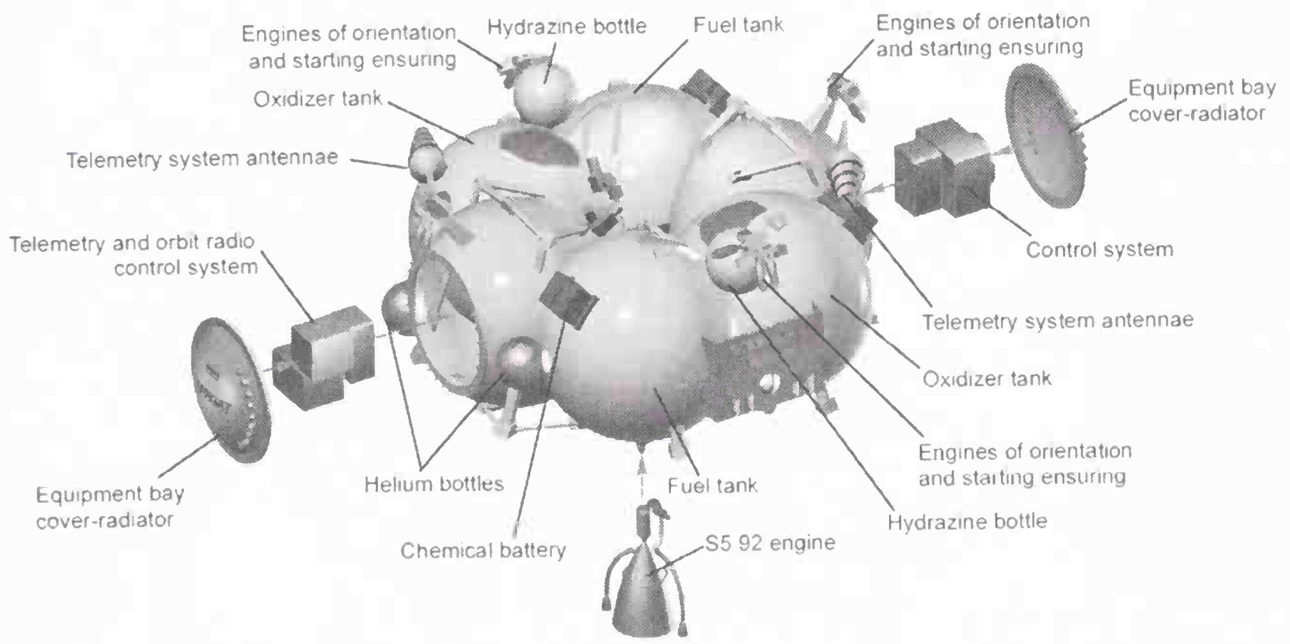
The Fregat is a more capable stage first introduced in 2000. It is capable of multiple restarts and has enabled Soyuz for the first time to perform more complex and higher-energy missions such as launching GTO satellites or planetary probes. The Fregat upper stage is derived from a spacecraft propulsion system, in this case the propulsion unit used on the Phobos and Mars-96 interplanetary spacecraft. The S5.92 main engine has been used on 27 interplanetary missions, including lunar landers, Mars orbiters, and missions to Venus such as the Vega spacecraft. The main engine can operate at two thrust levels. High thrust is used for large delta-V maneuvers, and low thrust is used when more precise impulse is required. The stage structure consists of six truncated spheres forming a ring with the main engine in the center. Two spheres are oxidizer tanks, two are fuel tanks, and two are pressurized avionics and equipment compartments. The avionics systems use off-the-shelf components to reduce the development cost and schedule. The guidance and control system is developed from systems on Zenit, and the telemetry system is in use in other spacecraft. Lavochkin Science and Production Association, manufacturer of the Fregat stage, has proposed its use on other launch vehicles, such as Proton and Zenit. However, adoption on these vehicles appears unlikely in the near future.



Engine S5.92: 1 — main chamber; 2 — exhaust nozzles;
3 — hinge ensuring plane-parallel motion; 4 — turbopump assembly

Courtesy Starsem.

S5.92 Engine



Courtesy Starsem.

Soyuz Fregat Stage

Vehicle Design

	Block L (Molniya Fourth Stage)	Fregat (Soyuz U and Soyuz ST)
Dimensions		
Length	3.2 m (10.5 ft)	1.5 m (4.9 ft)
Diameter	2.4 m (7.9 ft)	3.35 (11.0 ft)
Mass		
Propellant Mass	5500 kg (21,100 lbm)	5350 kg (11,800 lbm) plus 85 kg (187 lbm) ACS hydrazine
Inert Mass	1160 kg (2560 lbm)	1000 kg (2000 lbm)
Gross Mass	6660 kg +700 kg ullage motors and other jettisoned mass (14,500 + 1540 lbm)	6435 kg (14,190 lbm)
Propellant Mass Fraction	0.83	0.83
Structure		
Type	Monocoque?	Monocoque
Material	Aluminum	Aluminum
Propulsion		
Engine Designation	11D33/S1-5400 (RSC Energia?)	S5.92 (KB Khimash)
Number of Engines	1 engine with 4 verniers	1
Propellant	LOX/kerosene	UDMH/N_2O_4
Average Thrust	66.7 kN (15 klbf)	Main engine: High thrust: 19.6 kN (4.4 klbf) Low thrust: 13.7 kN (3.08 klbf) Exhaust nozzles: High thrust: 360 N (87 lbf) Low thrust: 190 N (43 lbf)
Isp	340 s	High thrust: 327 s Low thrust: 316 s
Chamber Pressure	54.5 bar (790 psi)	High thrust: 98 bar (1420 psi) Low thrust: 68.5 bar (990 psi)
Nozzle Expansion Ratio	?	?
Propellant Feed System	Gas-generator turbopump	Pressure fed
Mixture Ratio (O/F)	?	2.0:1
Throttling Capability	No	Two levels
Restart Capability	No	20 restarts
Tank Pressurization	?	Helium
Attitude Control		
Pitch, Yaw	Main engine gimbal	12 ACS thrusters
Roll	3 axis verniers	12 ACS thrusters
Staging		
Nominal Burn Time	250 s	877 s
Shutdown Process	Command shutdown	Command shutdown
Stage Separation	Spring ejection of payload	Springs?

Attitude Control System

Soyuz/Molniya

When the boosters and second stage are firing, steering is provided by gimbaled vernier engines. There are two verniers on each first-stage booster, and four on the second-stage core. The verniers are hydraulically gimbaled, and can swivel ±45 degrees in one plane. Additional control authority while inside the lower atmosphere is provided by four steerable aerodynamic fins (also referred to as rudders), one at the base of each booster. Interestingly, these fins are some of the only components that must be added to the vehicle at the launch pad. The third stage is steered using four 6-kN (1.4-klbf) vernier engines that are fed from the main engine gas generator exhaust gasses. These can each rotate ±40 deg.

Block L (Molniya fourth stage)

The S1-5400 engine can be gimbaled in two axes to provide pitch and yaw control. Roll control during burns and three-axis control during coast phases are provided by a set of thrusters fed by exhaust gas from the gas generator.

Fregat

During main engine burns, pitch and yaw are controlled by moving the S5.92 main engine. However, it does not pivot around a gimbal bearing like conventional engines. Because the Fregat is a very short stage with the engine in the middle, its center of gravity is located near the top of the engine, and therefore engine gimbaling would not produce a significant moment arm. Instead, the engine is mounted on a hinge that allows the whole engine

VEHICLE DESIGN

to move laterally in a flat horizontal plane. As a result, the engine thrust is always oriented axially, but can be offset laterally from the centerline of the stage, thus creating a moment arm to provide control. In addition to the main engine, Fregat has four sets of three thrusters each, which are used for propellant settling before main engine ignition, and three-axis control during coast phases. The combined system is referred to as the Propulsion Subsystem for Attitude Control and Engine Ignition (PSACEI). The PSACEI thrusters are 50 N (11 lbf) monopropellant units, with a specific impulse of 225 s. They are fueled with 85 kg (187 lbm) of hydrazine stored in two tanks.

Avionics

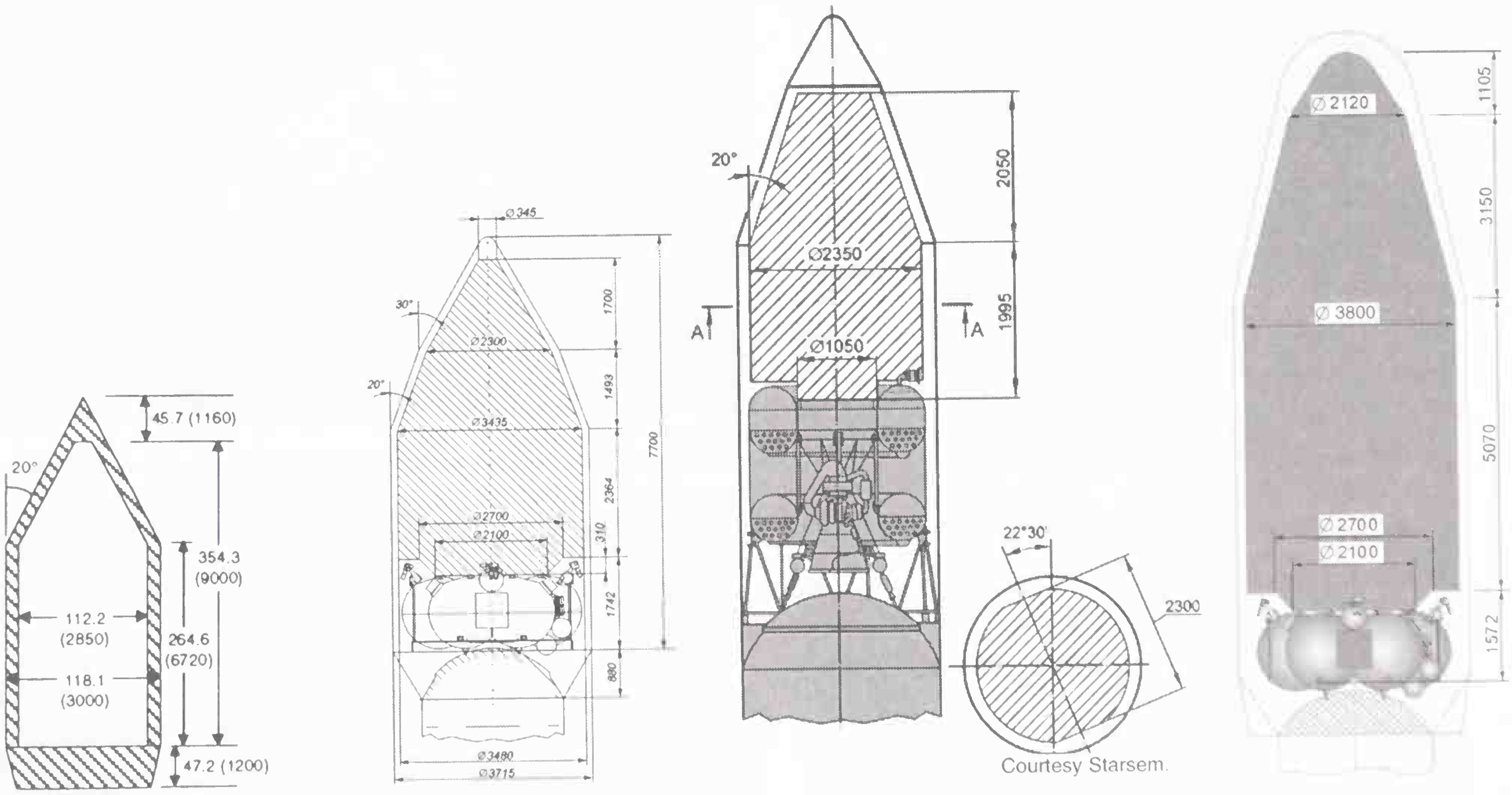
On the Soyuz, one system located in the equipment bay at the top of the second stage controls the vehicle through second-stage flight. After second-stage shutdown, control is handed off to a second control system contained in the intertank section of the Block I third stage in the case of Soyuz, or on the Block L stage for Molniya. The Soyuz/Molniya control system is a relatively primitive analog system. An autostabilizer (AS) performs both attitude stabilization and in plane pitch steering. The AS is not capable of out-of-plane yaw steering. Steering commands are implemented by the programmable pulse generator, which sends electrical commands to vehicle systems to set the pitch attitude and target velocity. The PPG is programmed before launch by encoding the desired commands onto movie film. The control system also contains propellant management systems, a telemetry system, and both AC and DC power supplies. Flight safety is provided by an onboard system that shuts down the main engines if the attitude is determined to be incorrect. Russian launch vehicles do not typically have a ground-controlled command destruct system.

The Fregat stage has a more modern avionics suite including a digital computer and inertial measurement platform. This enables it to adjust planned maneuvers to correct for errors in the performance of the lower three stages, improving injection accuracy. Fregat can also perform out-of-plane yaw maneuvers to reach nonstandard inclinations. The advanced avionics of the Fregat have already proved useful. On the first operational flight of Fregat, carrying Cluster 2 satellites for ESA, the Soyuz third stage shut down 3 s early because of a manufacturing fault. The Fregat flight software adjusted to correct for the error and reached the target orbit safely.

The Soyuz 2 will have new avionics that for the first time incorporate a computer, a digital control system, and an inertial measurement system – standard elements on most other launch vehicles. These new components will give Soyuz 2 the capability for yaw steering, allowing it to reach nonstandard inclinations. The system will also be capable of controlling less aerodynamically stable configurations, allowing the use of larger payload fairings. Downrange zones reserved for the impact of stages can be reduced because of improved accuracy. The improved control system will also weigh 200 kg (440 lbm) less. Changes in the telemetry system will save another 160 kg (350 lbm). The new system will be produced within Russia, eliminating dependence on suppliers in Ukraine.

Payload Fairing

A variety of payload fairings are used on the Soyuz and Molniya, and several new designs have recently been developed or are being designed. In addition to those shown below, a 3.0-m (10-ft) diameter, 16.8-m (55.1-ft) high fairing with an escape tower is used for manned flights of the Soyuz spacecraft. A 2.7-m (8.8-ft) diameter, 9-m (29.5-ft) high fairing is used for Bion, Photon, and Resurs-F spacecraft. A 3.3-m- (10.8-ft-) diameter, 9.5-m- (31-ft-) high fairing is used for Kosmos spacecraft. The new Type S fairing, shown below, is the widest yet, to hold the Fregat upper stage. For commercial flights of the Soyuz 2, Starsem offers the even larger 4-m-diam Type ST fairing, which is a Russian-built fairing similar to the Ariane 4 fairing. Acoustic blankets, radio windows, and access doors are all optional on the Type S and Type ST fairings.



Courtesy Starsem.

	Progress capsule fairing (Soyuz U)	Type S (Soyuz/Fregat)	Molniya (Molniya)	Type ST (Soyuz 2)
Length	11.36 m (37.3 ft)	7.91 m (26.0 ft)	7.8 m (25.6 ft)	11.4 m (37.4)
Primary Diameter	3.00 m (9.8 ft)	3.72 m (12.2 ft)	2.7 m (8.9 ft)	4.1 m (13.5)
Mass	4500 kg (9900 lbm)	?	?	?
Sections	2	2	2	?
Structure	Skin-stringer?	?	Skin-stringer?	?
Material	Aluminum	Aluminum	Aluminum	Carbon fiber reinforced plastic

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	Type S: 3435 mm (135.2 in.) Type ST: 3800 mm (149.6 in.)
Maximum Cylinder Length	Type S: 2364 mm (93.1 in.) Type ST: 5070 mm (199.6 in.)
Maximum Cone Length	Type S: 3193 mm (125.7 in.) Type ST: 4448 mm (175.1 in.)
Payload Adapter Interface Diameter	Mission specific
Payload Integration	
Nominal Mission Schedule Begins	T–12 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	?
On-Pad Storage Capability	10 days unfueled, 27 h for a fueled vehicle
Last Access to Payload	T–50 m
Environment	
Maximum Axial Load	4.8 *g*
Maximum Lateral Load	1.0 *g*
Minimum Lateral/Longitudinal Payload Frequency	15/25 Hz
Maximum Acoustic Level	Type S: 125 dB at 250 Hz Type ST: 136 dB at 250 Hz
Overall Sound Pressure Level	Type S: 140 dB Type ST: 141 dB
Maximum Flight Shock	Depends on adapter
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	2.5 kPa/s (0.36 psi/s)
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	Soyuz, 260×900 km (140×485 nmi), 51.8 deg orbit: ±9 km (5 nmi) perigee ±46 km (24.8 nmi) apogee ±0.05 deg inclination Soyuz/Fregat, 1000 km (540 nmi) circular orbit: ±10 km (5.4 nmi) semimajor axis ±1 deg inclination
Attitude Accuracy (3 sigma)	Orientation uncontrolled by Soyuz, controlled only with Ikar or Fregat upper stages
Nominal Payload Separation Rate	Mission specific
Deployment Rotation Rate Available	Spin available only with Fregat upper stage
Loiter Duration in Orbit	1 h for Molniya Block L, 48 h for Fregat?
Maneuvers (Thermal/Collision Avoidance)	Collision avoidance only
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Soyuz has carried multiple spacecraft such as Globalstar spacecraft in sets of four on mission-specific dispensers. Comanifest has not generally been performed in the past.
Auxiliary Payloads	Standardized opportunities for auxiliary payloads have not been reported. The Molniya Block L stage has surplus volume for auxiliary payloads in the center of its toroidal oxygen tank. The joint U.S./Russian Skipper spacecraft was carried in this volume.

Production and Launch Operations

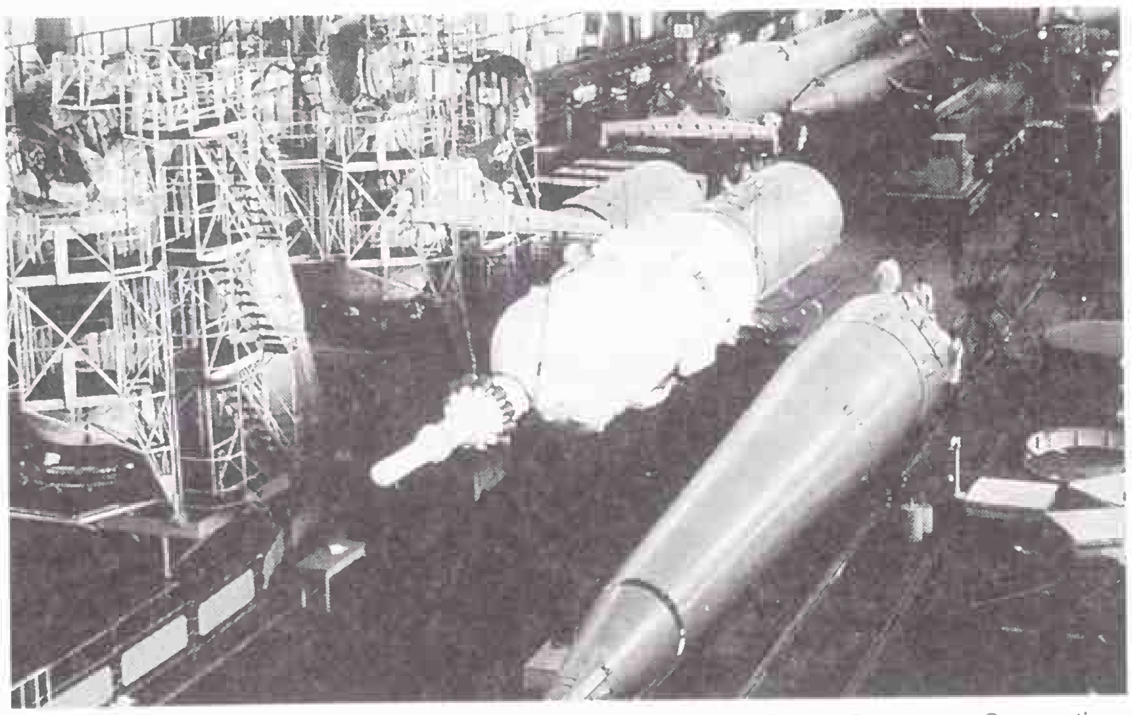
Production

The Soyuz vehicles are produced in Samara, Russia, by the Samara Space Center (SSC). SSC consists of two closely related, but distinct organizations that have been involved in Soyuz production almost since the beginning. The Central Specialized Design Bureau—Tsentralnoe Spetsiliairovannoe KB (TsSKB) was established in Samara as a branch of Sergei Korolev's OKB-1 design bureau, to oversee production of the R-7 missile and related space launch vehicles at the Aviation Plant #1 factory. TsSKB has responsibility for the design of Soyuz/Molniya, and also works on recoverable scientific and reconnaissance satellites, communications spacecraft, and other space projects. TsSKB became independent in 1975. The Aviation Plant #1 became the MZ Progress factory. MZ Progress worked closely with TsSKB on Soyuz, but has also performed production work for a number of other launch vehicles (including the Energia heavy-lift vehicle) and spacecraft. To cooperate more closely to commercialize Soyuz, TsSKB and the Progress factory have formed the Samara Space Center, which represents both parties.

Production of the Soyuz/Molniya has been very high in the past, peaking at a rate of more than one complete vehicle per week in the 1970s and 1980s. Many of the program subcontractors are located in Samara or close by. The RD-107/108 engines are built from an NPO Energomash design at the Frunze factory in Samara. While Soyuz production has dropped to about a dozen per year because of decreased government demand, a capacity to build 25–30 vehicles per year is still present. The following organizations are involved in Soyuz production.

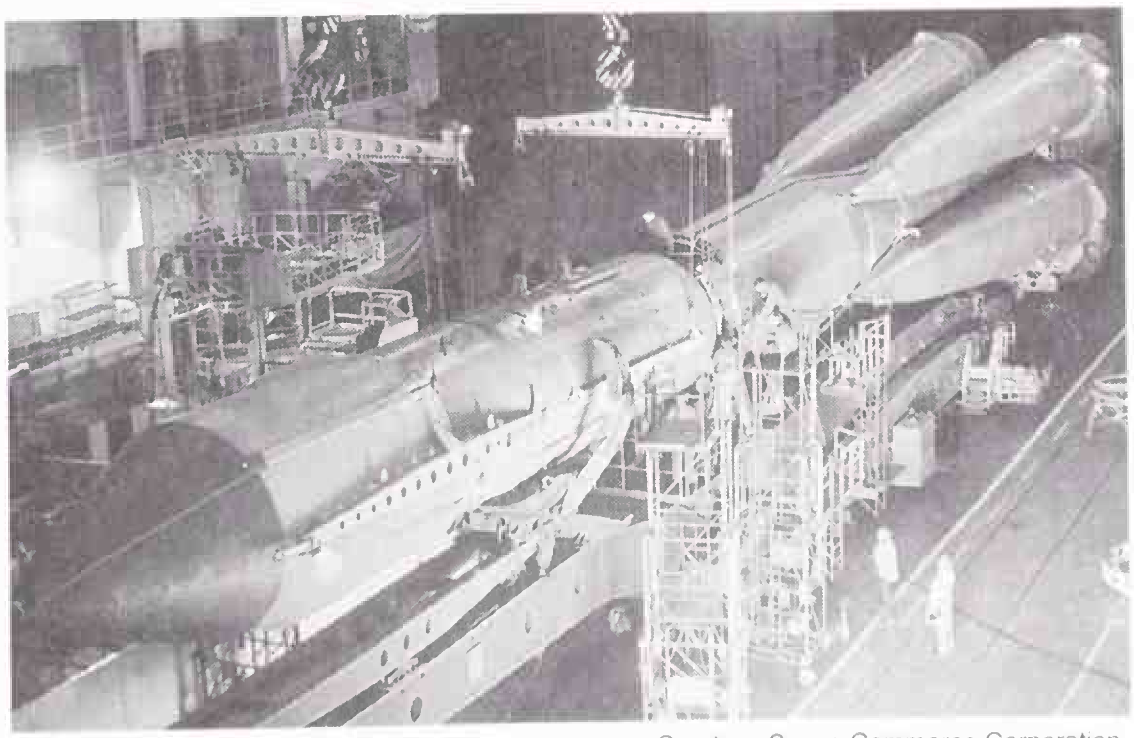
Organization	Responsibility
Samara Space Center—TsSKB and Progress factory	Prime contractor for Soyuz/Molniya vehicles and manufacture and integration of stages
NPO Lavochkin	Fregat and Block L upper stage, Type S payload fairing
KB Khimash	Fregat stage main engine
KVoronyezh Mechanical Factory	RD-0110 third stage engine
AO Motorostroitel, Frunze Factory	First and second stage engines
PA Kommunar	Control systems
EADS	Payload adapter/dispensers

Launch Facilities



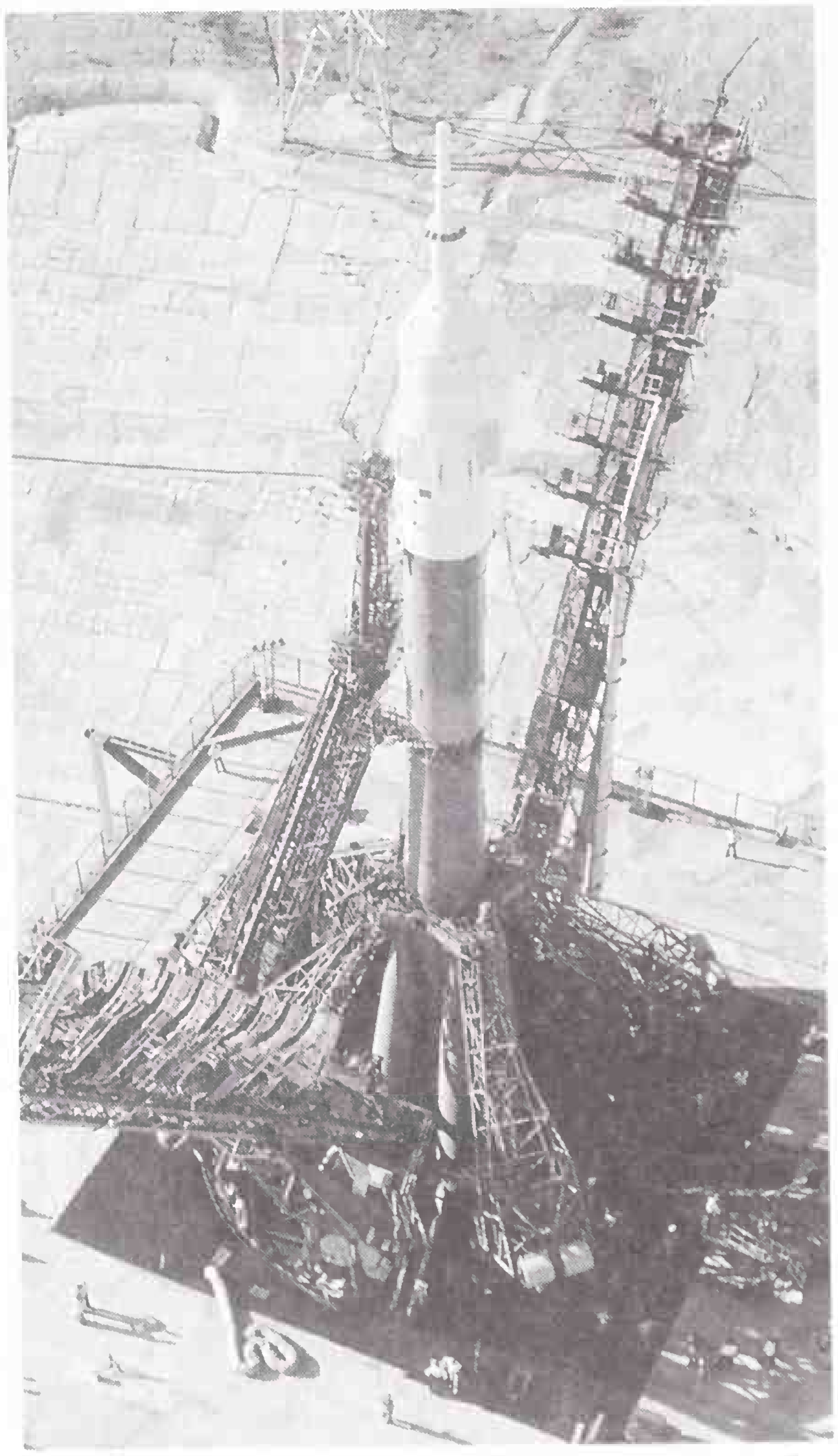
Courtesy Space Commerce Corporation.

Delivery of Vehicle to Assembly Building in Stages



Courtesy Space Commerce Corporation.

Horizontal Integration of Stages



Courtesy Space Commerce Corporation.

View of Soyuz Vehicle on Launch Pad

Production and Launch Operations

Launch Operations—Baikonur, Plesetsk

Soyuz is launched from both Baikonur and Plesetsk. Molniya is launched only from Plesetsk. There are two operational pads at Baikonur: Pad 5 at LC 1 and Pad 6 at LC 31. There are three active pads at Plesetsk: LC 16 Pad 2, and LC 43 Pad 3 and 4. The oldest Soyuz complex at Plesetsk, LC 41, has apparently been deactivated. Launch operations at both cosmodromes are similar and are described together below. For maps and background information on both facilities, please see the Spaceports chapter.

After arrival at the cosmodrome, the payload undergoes checkout and fueling. It is then transported to the space vehicle assembly building to be integrated with the launch vehicle, the components of which have already been transported to the cosmodrome by rail. The integration process can follow one of two approaches, depending on payload requirements. In one option, the payload, adapter, interstage, and, for Molniya, the Block L stage, are integrated vertically. The two halves of the payload fairing are then attached to either the interstage (Soyuz) or to the Block L stage (Molniya) and connected together. This assembly (referred to as the "head block" in Russian terminology) is then rotated to a horizontal position, moved to a transporter/erector railroad car, and integrated horizontally with the rest of the launch vehicle. In the other option, the payload, interstage, Block L (Molniya), and Block I are integrated vertically, after which the assembly is rotated to a horizontal position and the complete payload fairing is rolled into place over the payload. This stack is then transferred to a transporter/erector car and attached to the previously assembled core stage and strap-on boosters. Throughout this process, temperatures within the vehicle assembly building called the MIK are maintained between 5° and 35° C (41–95° F). Once the payload is integrated with the launch vehicle, temperatures within the payload fairing can be maintained between 15° and 25° C (59–77° F).

Following integrated testing in the MIK, the launch vehicle is transported horizontally to the launch pad. The maximum speed during the 5-km (3.1-m) rail trip is 10 km/h. Upon arrival at the pad, the vehicle is raised into a vertical position and installed onto the pad. During the erection process, payload thermal control is interrupted for not more than 3.5 h.

The launch pad consists of a fixed rectangular platform, in the center of which is a square pit. Once in place on the pad, the vehicle rests inside this pit on a rotating structure, with its aft end approximately 7 m (23 ft) below the level of the launch platform. The rotating structure allows the vehicle to be aligned to the proper flight azimuth before launch. Attached to the rotating structure are four supporting trusses, along with service towers and umbilical masts. The support trusses act to secure the vehicle in position before launch; they are held in place by the weight of the vehicle itself. The entire launch pad structure is suspended over a large flame trench.

Direct access to the launch vehicle (above the level of the launch platform) and payload is gained through service towers positioned around the vehicle; these are withdrawn 50 min before launch. Retractable platforms permit access to the launch vehicle below the level of the launch platform until one hour before launch. Payload fairing environmental control is halted 40 min before launch; by launch time, the temperature inside the fairing can vary from –35° to +50° C (–31° to 122° F). An umbilical mast, supplying electrical, pneumatic, and hydraulic connections to the Block I and (for Molniya) Block L, as well as the payload, is detached 10 min before ignition, at which time these vehicle elements are switched to internal power. A cable mast connected to the control system of the first-stage core and the strap-on boosters is disconnected at ignition. After 5–7 cm (2–3 in.) of vehicle vertical motion, the support trusses retract under the influence of large counterweights, like petals to allow the vehicle to clear the pad.

The entire processing flow, from launch vehicle and payload arrival at the cosmodrome until launch, requires 121 h, including 18 h for on-pad activities. The Soyuz vehicles are able to perform under severe weather conditions, including dense fog, wind, rain, snow, and the wide temperature variations experienced at Baikonur (–40° to 50° C). For example, in March 1988, a Vostok launcher carrying an Indian satellite lifted off the pad at Baikonur on schedule in the midst of a driving snowstorm.

Flight Sequence

No diagram available.

Event	Time, s
Main Engine Ignition	–2 ?
Liftoff	0
First-Stage Separation	118
Fairing Separation	150–170
Second-Stage Separation	286
Third-Stage Shutdown	520–540

Vehicle Upgrade Plans

Kourou Launch Site

European and Russian agencies and companies have been negotiating for several years to launch Soyuz from the European equatorial spaceport at Kourou. The advantages of low-latitude launch would enable Soyuz to launch up to 3000 kg (6600 lbs) to GTO. The idea has been controversial in Europe, where some officials have been concerned that Soyuz would reduce demand for Ariane 5, or that the limited market for small GTO satellites would not justify the investment in infrastructure. Others argued that Soyuz would provide Arianespace with more flexibility for handling small satellites, for which Ariane 5 is not optimized, and could therefore coexist with Ariane. Another attraction is the possibility that human spaceflight launches could be performed from Kourou. To persuade ESA, Russia tied the Kourou question to participation in broader aerospace cooperation including airliners and reusable launch vehicle development. Europeans also feared that if they didn't provide an equatorial launch site, someone else would (see the entry for Aurora). In June 2002 ESA governments gave permission for the plan, although it would take several more months to line up funding. The cost of the project is about 275 million ($270 million). Arianespace hopes to price launches at 40 million ($39 million), with the first launches taking place around 2005.

Yamal/Aurora

In 1996 RSC Energia and the Progress Central Design Bureau began studying advanced derivatives of Soyuz 2. In 1999 this concept was reworked and emerged as the Yamal program, with the goal of producing a modular family of vehicles that could launch payloads in a variety of sizes. Yamal became RSC Energia's alternative to Khrunichev's Angara program. The smallest vehicle, Polyot, would be air launched from an AN-124-100 cargo aircraft. It is proposed by the Vozdushny Start ("Air Launch") corporation, of which Energia holds a 40% stake. Polyot is intended to launch up to 3000 kg (6600 lbm) for $10 million. The Yamal family was also to include a Soyuz-based medium-class launch vehicle, which became Aurora, and larger Proton-class vehicles based on Aurora with enhanced strap-on boosters.

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San Diego, CA 92101

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Like many recent Russian launch vehicle programs, development of Yamal was stifled by a lack of domestic funding. A key milestone came when the Asia Pacific Space Centre (APSC) decided to invest in the project. During the mid 1990s APSC had promoted a plan to launch Soyuz rockets from Woomera, Australia, and a project to launch Angara from Australia's Christmas Island in the Indian Ocean (not to be confused with proposals for space launches from the other Christmas Island or Kiritimati, Republic of Kiribati, in the Pacific Ocean). APSC eventually decided to proceed with development of the Christmas Island launch site using the Aurora as its launch system. The combined development of the vehicle and the launch site is estimated to cost AUS$800 million (US$433 million). In 2001 the Australian government agreed to invest AUS$100 million (US$54 million) for infrastructure on the island. Neither the Russian government nor the Russian industrial partners are providing a significant share of the funding. Russian executives estimate Aurora could be priced at $60–70 million per launch. Launch services on Aurora are offered through APSC.

Aurora would replace the RD-108A engine on the Soyuz second stage with a much higher performance NK-33 engine. A four-chamber RD-0124 engine (developed for the third stage of Soyuz 2) would be used on the second stage as a vernier engine. The LOX tank diameter is increased to 3.4 m (11.2 ft) to accommodate the higher mixture ratio of the staged-combustion engines. The third stage would be based on the Vostok Block E stage rather than the Soyuz Block I and would also be enlarged to 3.4 m (11.2 ft) in diameter. The new Corvet fourth stage would be used for injection in GTO. It resembles a smaller version of the Block DM stage, with nearly spherical LOX tank, a toroidal kerosene tank, and 11D58M engine with an extendible nozzle for increased performance.

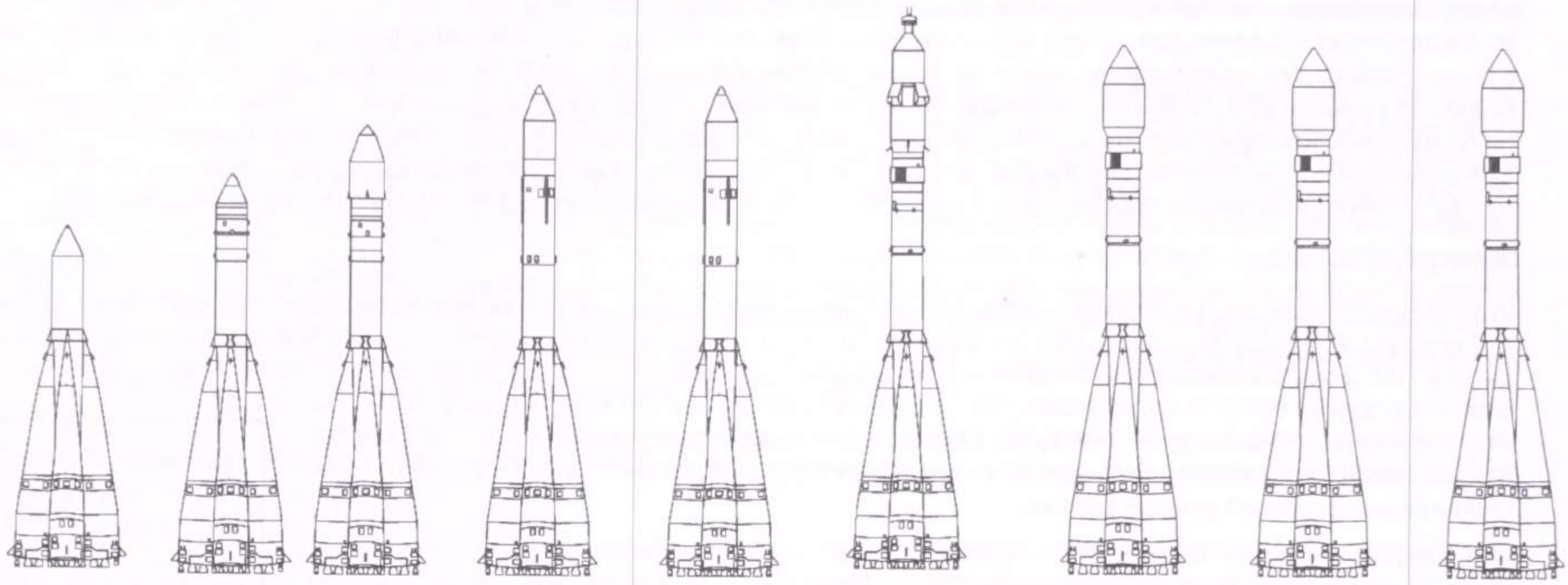
The combination of these performance improvements and the near-equatorial launch site will provide Aurora with much greater performance than the existing Soyuz. Maximum payload to a 200-km (108-nmi) 11-deg orbit would be 12,000 kg (26,500 lbm). If launched from Baikonur, Aurora could carry 10,600 kg (23,400 lbm) to a 200 km (108 nmi), 51.6 deg orbit. Up to 4500 kg (9900 lbm) can be delivered to GTO at 11 deg, or 2100 kg (4600 lbm) to GEO.

Aurora could be operational from Christmas Island by 2005. Earlier test flights may be launched from Baikonur at LC 250; the launch complex used for the maiden launch of the Energia heavy launch vehicle. However, the downturn in the commercial launch market has made the future of Aurora uncertain. The French have also attempted to quash the project, seeing it as a competitor to their own plans to provide near-equatorial launches of Soyuz from Kourou through Starsem and Arianespace.

Vehicle History

Vehicle Evolution

	Out of Production				Current Production				In Development



Vehicle	**Sputnik**	**Luna**	**Vostok**	**Voskhod**	**Molniya**	**Soyuz**	**Soyuz U/ Fregat**	**Soyuz FG**	**Soyuz 2**
Period of Service	1957–1958	1958–1960	1958–1991	1963–1976	1960–Present	1965–Present	2000–Present	2001–Present	2005

Vehicle History

Vehicle Description

Name	Article Number	Sheldon Name	DoD Designation	Description
Sputnik	8K71O or 8K71PS	A	SL-1	First space launch vehicle, essentially test configuration of R-7 ICBM. Used for two flights carrying Sputnik 1 and 2. Consisted of one core stage plus four strap-on boosters.
Sputnik	8A91	A	SL-2	Upgraded version of 8K71 designed specifically for space launch, with improved engines. Carried 1327 kg (2925 lbm) Sputnik 3.
Vostok, Luna	8K72	A-1	SL-3	Upgraded with new Block E third stage. Luna version had short fairing for launching the first lunar probes.
Vostok K	8K72K	A-1	SL-3	Similar to 8K72, with improved engines on lower stages and RO-7 engine replacing RO-5 on Block E third stage.
Vostok 2	8A92	A-1	SL-3	Improved version of Vostok with upgraded engines, control systems.
Vostok 2M	8A92M	A-1	SL-3	Version of 8A92 specifically for polar and sun-synchronous missions, with improved Block E avionics, modified fairing.
?	11A59	A-1-m	SL-5	Variation of Vostok without Block E third stage, Polyot spacecraft propulsion system performed orbit injection. Used for only two flights in 1963 and 1964.
Molniya	8K78	A-2-e	SL-6	Derivative of Vostok with larger Block I third stage replacing Block E, and new Block L fourth stage.
Molniya-M	8K78M	A-2-e	SL-6	Upgraded version of Molniya. Available with versions of the Block L upper stage designated ML or 2BL.
Voskhod	11A57	A-2	SL-4	Similar to Molniya without Block L fourth stage. Typically grouped with Soyuz, which it closely resembled. Used for upgraded Voskhod manned capsule and military spacecraft.
Soyuz	11A511	A-2	SL-4	Similar to Molniya without Block L stage.
Soyuz	11A510	A-2?	SL-10	Maneuverable upper stage in place of the Block I stage, flown twice in 1965 and 1966.
Soyuz L	11A511L	A-2	SL-4	Variation of Soyuz.
Soyuz M	11A511M	A-2	SL-4	Variation of Soyuz.
Soyuz U	11A511U	A-2	SL-4	Modern variation of Soyuz.
Soyuz U2	11A511U2	A-2	SL-4	Variation of Soyuz U with upgraded second-stage engine burning syntin synthetic propellant, used for manned missions until 1995.
Soyuz U/Ikar	11A511U?	None	SL-4	Soyuz U with added Ikar fourth stage used for injection into higher orbits.
Soyuz U/Fregat	11A511U?	None	SL-4	Soyuz U with new Fregat fourth stage based on planetary spacecraft propulsion system.
Soyuz FG	11A511U-FG	None	SL-4	Upgrade of Soyuz U with improved engines.
Soyuz 2	None to date	None	None to date	Upgrade of Soyuz FG with digital avionics and new third-stage engine.

Historical Summary

On 4 October 1957 the Soviets startled the world with the launch of Sputnik. The launch vehicle was based on the R-7, or Semyorka, ICBM, adapted for injection of a satellite into LEO. The satellite was called the PS (preliminary satellite), though Korolev's launch team referred to it as the SP (Sergei Pavlovich) in his honor. The satellite was composed of a pressurized, highly polished, spherical steel ball, a thermometer, a silver-zinc battery, four whip antennas, and a radio transmitter. The satellite transmitted its beeps for three weeks, reentering the atmosphere 4 January 1958 after 92 days in orbit.

The Soyuz/Molniya family of launch vehicles is based on the original R-7 ICBM (designated SS-6 in the West, code-named Sapwood by NATO), which was first flown in August 1957. The R-7 was unsuccessful as an ICBM, with only four being deployed operationally.

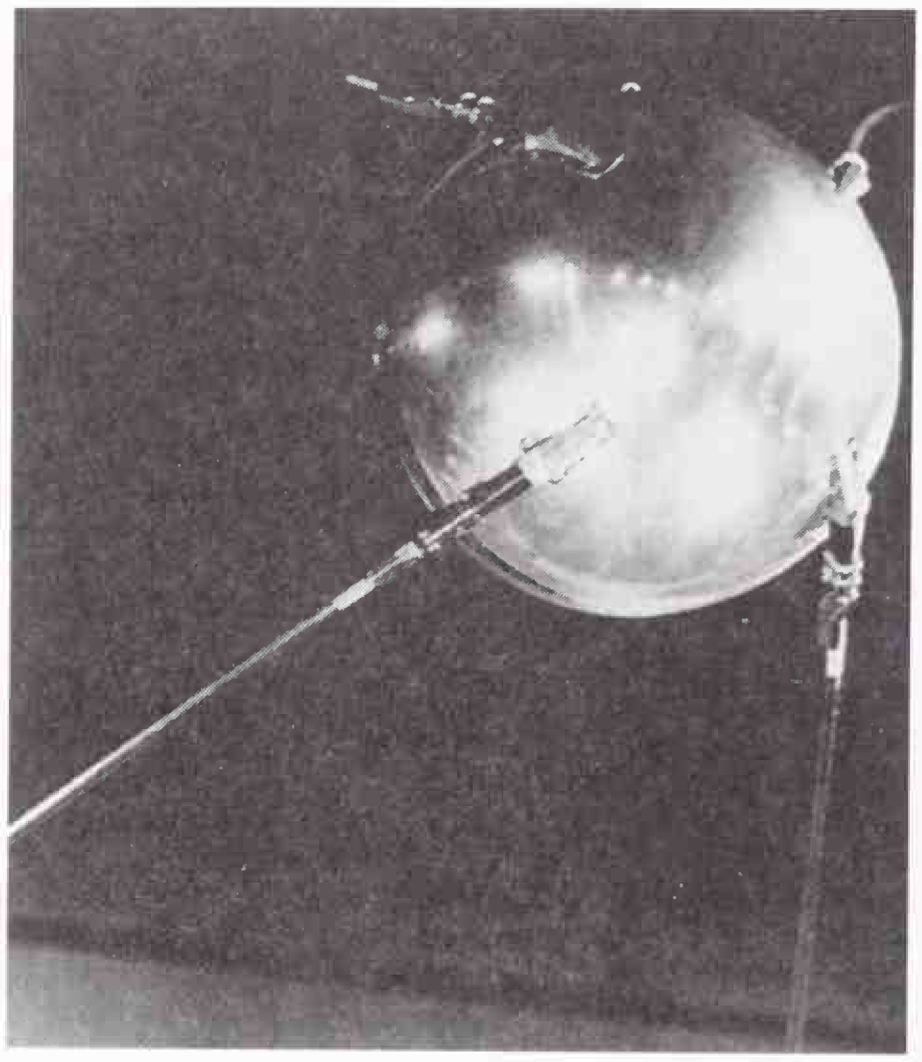
The R-7 evolved to become the Sputnik (satellite), Vostok (east), Voskhod (sunrise), Molniya (lightning), and Soyuz (union) launch vehicles, and each of these had several variants. The Soyuz, and earlier versions of the Vostok and Voskhod, have launched 90 flights of humans as of 30 June 1999. After 40 years, this family of rockets has long been the most-launched rocket family with upgrades being introduced that will continue its production into the next century.

The original vehicle designated Sputnik (A, or SL-1 in the West) is classified as a two-stage vehicle. The vehicle consisted of a central core with four strap-on boosters. The Russians class this as a two-stage assembly with the engines of the strap-ons and the central sustainer igniting simultaneously at liftoff.

Vehicle History

Sergei Korolev's OKB-1 design bureau, which has since evolved into RSC Energia, was responsible for the development of the original Sputnik launch vehicle. This vehicle launched the first Sputnik satellites in 1957 and 1958. The third and largest Sputnik satellite had a mass of 1327 kg (2925 lbm) and was delivered to a 226×1881 km (122×1016 nmi) orbit, by an upgraded version of the first two launch vehicles. All Sputnik vehicle launches were from Baikonur.

In 1958, Dmitri Kozlov, a close associate of Korolev, was placed in charge of establishing the production facility in the city of Kuibyshev (now Samara), which lies about 880 km (550 mi) southeast of Moscow on the Volga River. On 23 July 1959, the Serial Production Design Department was officially established at the MZ Progress factory in Kuibyshev, and one year later it was incorporated into OKB-1 as its Kuibyshev branch office. By 1961, all design, engineering, flight testing, and operations responsibility for the boosters had been transferred from OKB-1 in Moscow to Kuibyshev. In 1974, this branch was split out into a separate organization and named Tsentralnoe Spetsializirovannoe KB (TsSKB or Central Specialized Design Bureau), with Dmitri Kozlov as chief designer. Today, TsSKB, with Kozlov still serving as chief designer, reports directly to the Russian Space Agency. Production of the Soyuz/Molniya-class boosters still takes place at the MZ Progress factory in Samara. Currently, MZ Progress production facilities are shared by RSC Energia and TsSKB.



Courtesy NASA.

Sputnik 1

Over the years, the desire for increased launch vehicle capability led to the development of a series of more capable rockets based on the original R-7 vehicle design. The first step was to add a third stage, called the Block E, to the basic R-7 vehicle two-stage configuration. Three failed launch attempts were made in 1958, with the first successful launch of a three-stage vehicle taking place in 1959. This vehicle was designated Vostok (A-1, SL-3). It consisted of an R-7 core stage and strap-on boosters with a third stage attached by a truss structure. The initial application of the Vostok was to launch the Luna moon payloads; it was later used to launch the manned Vostok capsule and first generation recoverable Kosmos satellites. Use of the Vostok rapidly declined in the mid 1980s, and production of the vehicle was halted in 1985. On 17 March 1988, the first Soviet commercial launch occurred aboard a Vostok with an Indian remote-sensing spacecraft, IRS-1A. The final Vostok flight carried India's IRS-1B into SSO from Baikonur on 29 August 1991. During its more than 30 years of active use, Vostok underwent several systems upgrades, including a major redesign in 1964.

The Molniya launch vehicle (A-2-e, SL-6) was introduced in 1960 with its first successful launch in 1961. It dramatically increased the capability of the Soviet space program to launch high-energy planetary and lunar missions and place payloads into highly elliptical Earth orbits. Molniya was used to launch Luna (4–14) and Venera (1–8) missions and the Soviet Union's first three Mars probes. Following the development of the much larger Proton (D-1-e, SL-12) in the mid 1960s, Molniya's planetary launch missions dwindled. The Molniya incorporates two new core stages in addition to the R-7 core and strap-on motors, resulting in a four-stage vehicle. Molniya's Block I second core stage is three times the size of Vostok's, and its main engine generates five times the thrust. The small fourth core stage (designated Block L) was restartable on orbit, allowing for additional mission flexibility. Molniya was launched exclusively from Baikonur until 1970, when launches from Plesetsk began.

The three-stage Soyuz (A-2, SL-4) premiered in 1963. This vehicle employs the same basic core stage and strap-on motors as Vostok and Molniya and a third stage very similar to that of Molniya. The Soyuz Block I provides a considerable increase over Vostok's payload capability to LEO. Since 1964 all Soviet and Russian human missions have relied on the Soyuz, but the largest program it supports involves military and civilian recoverable photographic reconnaissance flights. Significant upgrades to the vehicle took place in 1973 and 1982. It is used today for launching manned Soyuz spacecraft, Progress cargo spacecraft, and biological research satellites, as well as Kosmos observation satellites. It also launched the Voskhod manned spacecraft.

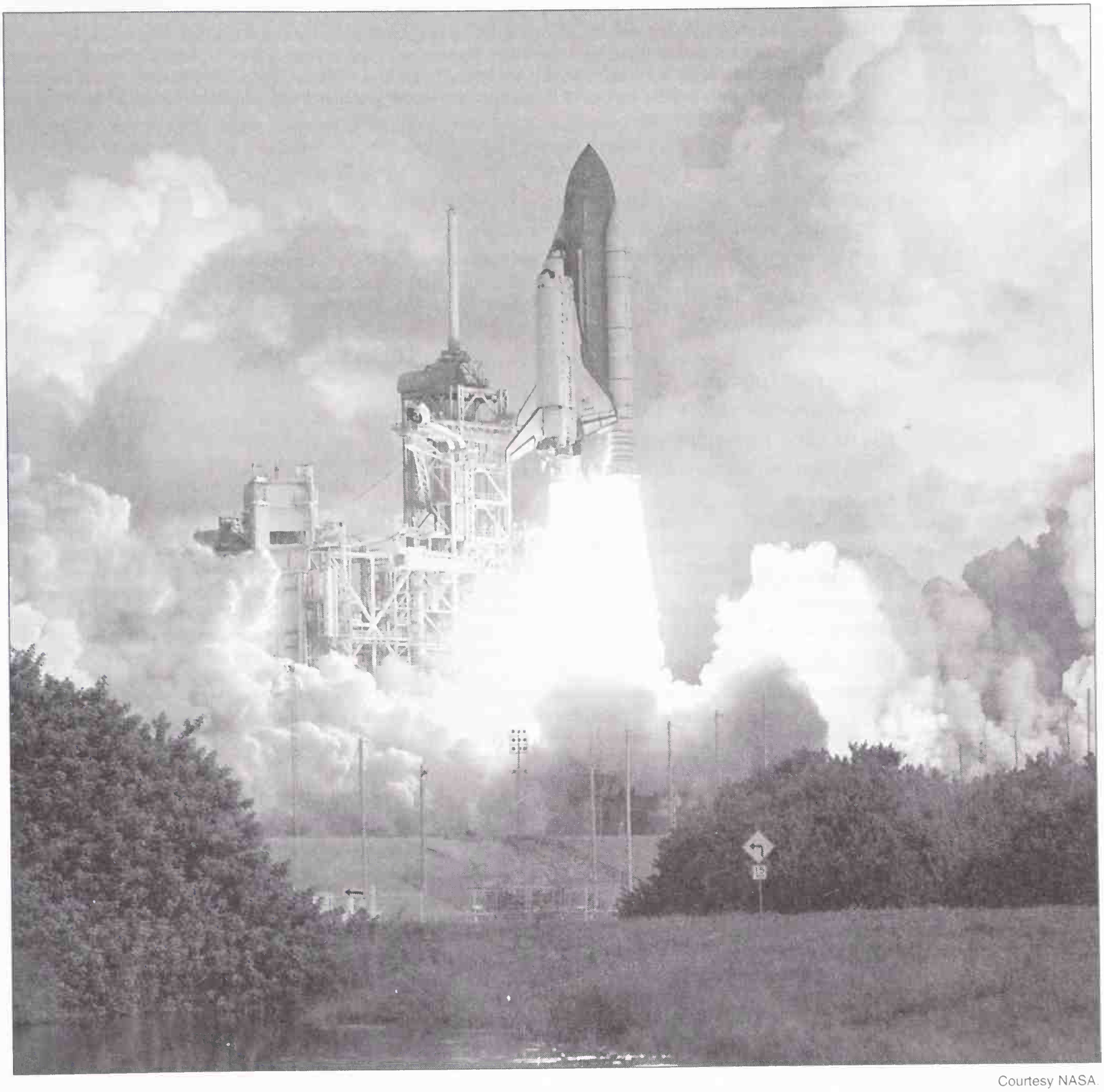
Although the Soyuz/Molniya-class boosters have remained essentially unchanged in external appearance since their inception, their systems have been subjected to a constant stream of modifications over the years. In addition, subtle differences exist between superficially identical components. For example, the basic core stage and strap-on boosters may vary slightly from vehicle to vehicle in such details as propellant loading and main engine thrust, depending on the vehicle version and mission application. Also, the third stages used by Soyuz and Molniya, although very similar to one another in propellant capacity and engine performance, differ in the upper-stage control system, which resides on the third stage for the Soyuz and on the fourth stage for Molniya. Soyuz can also employ a number of different payload fairings depending on the mission at hand. The Soyuz/Molniya-class launch vehicles, until recently, have exclusively used LOX and kerosene propellants in all stages. However, this is changing with the recent introduction of the Ikar and Fregat upper stages, both of which use UDMH/N_2O_4 propellent.

In 1996 TsSKB–Progress, the Russian Space Agency, Aerospatiale, and Arianespace entered into a joint venture, named Starsem, to market the Soyuz and Molniya rockets commercially to customers other than the Russian government. Starsem is headquartered near Paris. Starsem has invested $35 million at Baikonur to offer processing facilities comparable to those found at other world-class launch sites. Starsem has been participating in studies with ESA and Arianespace to investigate the business opportunities of launching the Soyuz from the European spaceport in Kourou, French Guiana.

The arrival of Western customers and funding has allowed several Soyuz upgrade programs to reach fruition. First was the new Ikar upper stage, which was used in 1999 to launch a series of satellites for Globalstar. Ikar was a temporary solution until the more capable Fregat stage became operational the following year. Fregat was funded specifically to launch ESA's Cluster satellites. Fregat gives Soyuz the capability to address the commercially lucrative GTO market but also to launch future European planetary probes such as Mars Express. Next to arrive was the Soyuz FG in 2001. It has improved injectors on the first and second stage engines. In the future, the Soyuz 2 will offer significant increases in payload and fairing volume.

The high launch rates and relatively short times between failures and the next launch attempt suggest both a robust vehicle design and an effective system for failure investigation and resolution, hallmarks of a system that will continue to fly for many years to come.

SPACE SHUTTLE



Courtesy NASA.

NASA's Space Shuttle is the world's first reusable launch vehicle and has been used for all U.S. human spaceflight programs since 1981. It is able to carry heavy payloads to LEO and remain in orbit for up to two weeks before returning to Earth. Atlantis is shown lifting off on mission STS-106 in 2000.

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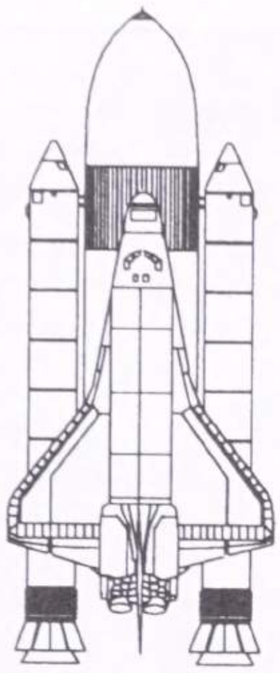
Space Shuttle

General Description



Space Shuttle

Summary

The Space Shuttle is the world's first reusable space launch system. It consists of a reusable delta-wing spaceplane called an orbiter; two solid-propellant rocket boosters, which are recovered and reused; and an expendable external tank containing liquid propellants for the orbiter's three main engines. The Space Shuttle carries astronauts to orbit to perform a wide variety of missions such as spacecraft deployment and recovery, research in pressurized modules in the cargo bay, repair, and space station support.

Status

Operational. First launch in 1981.

Origin

United States

Key Organizations

Marketing Organization	Not marketed
Launch Service Provider	NASA
Prime Contractor	United Space Alliance

Primary Missions

Human spaceflight supporting the International Space Station program

Estimated Launch Cost

$450–750 million average cost, depending on flight rate

Spaceport

Launch Site	Kennedy Space Center LC-39 A and B
Location	28.5° N, 81.0° W
Available Inclinations	28.5–57 deg
Landing Site	Kennedy Space Center Runway 15/33
Landing Site	Edwards AFB, various runways

Performance Summary

Payload support structures are considered part of the payload mass in the values below.

204 km (110 nmi), 28.5 deg	28,800 kg (63,500 lbm)
200 km (108 nmi), 90 deg	No capability
Space Station Orbit: 407 km (220 nmi), 51.6 deg	18,300 kg (40,300 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	No capability
GTO: 185×35,786 km (100×19,323 nmi), 28.7 deg	No capability (requires separate upper stage)
Geostationary Orbit	No capability (requires separate upper stage)

Flight Record (through 31 December 2003)

Total Orbital Flights	113
Launch Vehicle Successes	109
Launch Vehicle Partial Failures	2
Launch Vehicle Failures	2

Nominal Flight Rate

3–6 per year

Nomenclature

The Space Shuttle was originally called the Space Transportation System, and thus the STS abbreviation is often used to refer to the Space Shuttle. The abbreviation is also used in the flight numbering system. Currently, Space Shuttle flights are designated with a mission number such as STS-90, indicating the expected flight sequence when the mission was first planned. Because Shuttle manifests are frequently shuffled, these numbers do not always indicate the order in which the missions actually fly. Between 1984 and 1986, missions were indicated with a different numbering system, such as STS-51L. The first digit indicated the planned year of flight, the second indicated the launch site to be used (1 for Kennedy Space Center and 2 for Vandenberg AFB), and the letter indicated the expected order.

Each orbiter has both a given name and an orbiter vehicle (OV) number. Both are listed below. The names are taken from names of famous sailing ships used in the exploration of Earth, except for *Enterprise*, which is named after the starship from the television series *Star Trek*.

Enterprise	OV-101	Test orbiter used only for atmospheric and landing tests and fit verification and practice tests at launch sites.
Challenger	OV-099	Originally built as structural test article, it became second operational orbiter. It was destroyed in flight in 1986.
Columbia	OV-102	The first operational orbiter, heavier than later orbiters, but had extended duration orbiter (EDO) capability for 16-day missions. It was destroyed on descent in 2003.
Discovery	OV-103	The third operational orbiter.
Atlantis	OV-104	The fourth operational orbiter.
Endeavour	OV-105	The replacement for *Challenger*. It also has EDO capability for 16-day missions.

Cost

Because the Space Shuttle is not available commercially, there is no set launch service price. The cost per flight to NASA is also not well characterized because most costs are not accounted for on a mission by mission basis. Flight costs are effectively determined by two factors. A large annual fixed cost, which is independent of the flight rate, pays for overhead expenses such as program infrastructure and the large number of specialized employees who operate and maintain the vehicles. A smaller marginal cost covers specific expenses incurred on each flight, such as the costs of the external tank and solid motors. Per flight costs may thus be expressed as either average costs (total annual costs divided by the number of missions), or as marginal costs (only the cost of performing an additional mission, beyond the overhead expenses that are incurred whether or not the mission takes place). The Shuttle program budget has ranged from $3.0 to $3.2 billion per year between 1999 and 2002, and has flown four to seven missions per fiscal year. Therefore, average costs per mission have ranged from $450 million to $750 million, depending on how many missions are flown each year. There have been several estimates of marginal costs per flight, ranging from $60 million to more than $100 million. Two major contributors to the marginal cost per flight are the cost of replacing the external tank (ET) and SRBs. NASA currently procures the motors for the SRBs from ATK Thiokol under a $2.4 billion contract to provide 35 flight sets (70 motors). This is equivalent to $68 million per flight set. In 2000 NASA gave a contract to Lockheed Martin for 35 ETs. The contract was initially worth $1.15 billion, but was increased to $1.49 billion in 2002 when the production rate was decreased. This works out to $42.6 million per tank. So, ET and SRB production represents a combined $111 million per flight set of hardware. These contracts may include some fixed costs, so it is difficult to determine the true marginal cost impact.

Availability

Following the *Challenger* accident, the Space Shuttle was restricted to carrying only those payloads that require the unique capabilities of the Space Shuttle or the presence of a human crew. The Space Shuttle is therefore not available for conventional commercial satellites. Commercial customers for payloads such as microgravity experiments that require a human presence may use the Space Shuttle, but these represent a small portion of payloads. Most Space Shuttle missions support NASA's human spaceflight program. The Space Shuttle also performs deployment of very heavy NASA spacecraft that will not fit on other launch vehicles or smaller satellites that require both deployment and recovery. Availability of the Space Shuttle for these types of missions has been significantly reduced during construction of the International Space Station (ISS), because most flights are dedicated to ISS. While the U.S. military was originally a partner in the Space Shuttle program, military payloads have been off-loaded to expendable vehicles such as Titan IV. The last dedicated military Shuttle mission occurred in 1992.

Recently, the Space Shuttle has demonstrated a typical flight rate of 5–6 missions per year. A flight rate below about 5 per year is considered a safety risk, because it would be difficult to maintain operational proficiency. A flight rate above 8–10 per year would require additional investment in new infrastructure.

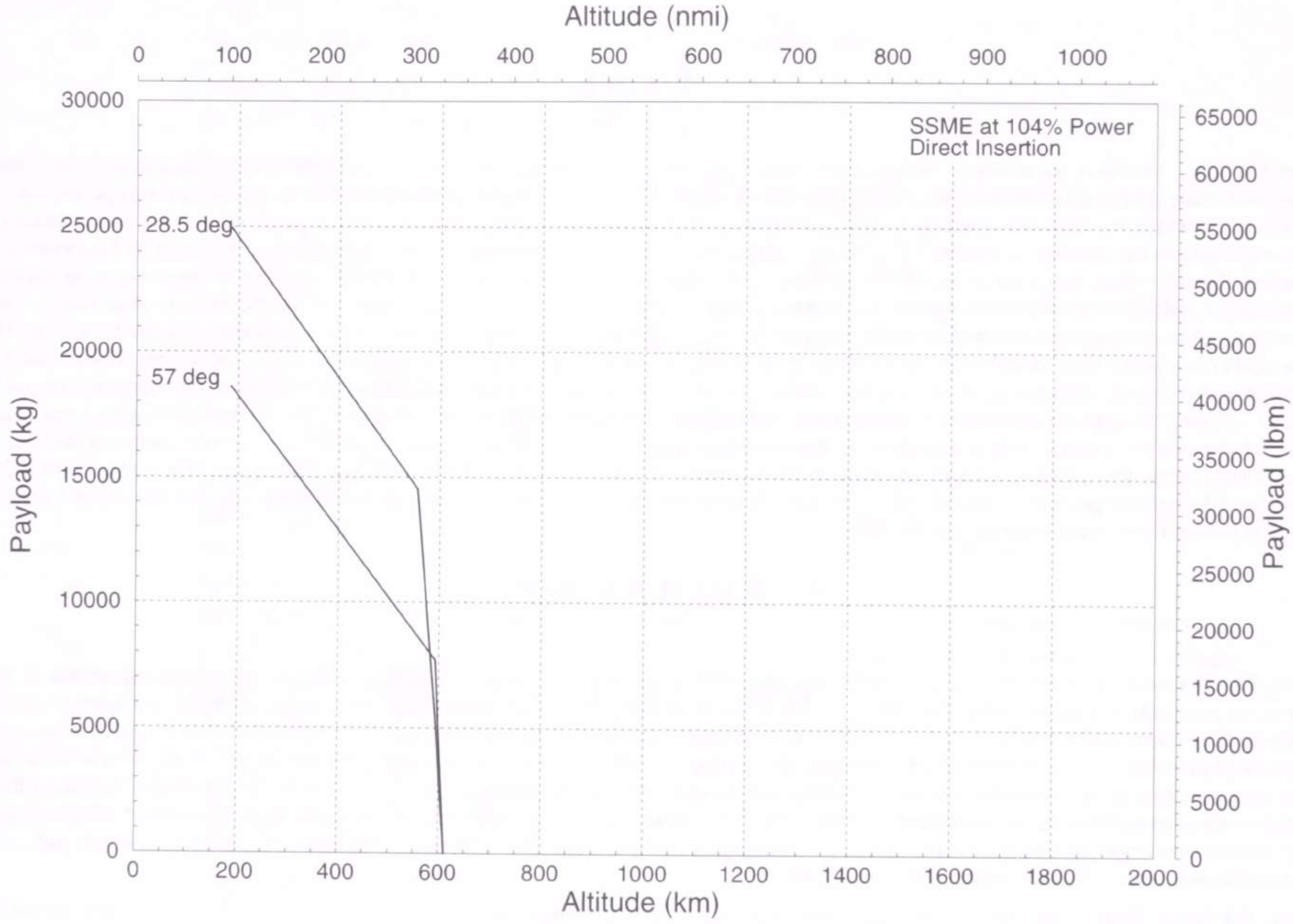
In February 2003 *Columbia* was lost during reentry. NASA postponed all Shuttle flights during the accident investigation and while it implements the recommendations of the Columbia Accident Investigation Board. At the time of publication, NASA indicated a return to flight would be possible in the March–April 2005 period. The Shuttle flight rate after the program recovers is uncertain but is likely to be in the range of only 4–5 flights per year.

Performance

The Space Shuttle can reach orbits between 28.5 deg and 57 deg when launched from Kennedy Space Center (KSC), Florida. Slightly higher inclinations may be achieved under special circumstances by using a dogleg maneuver. The highest inclination achieved on a Shuttle mission to date was 62 deg on STS 36 in 1990. Payload capability can also be restricted by the maximum cargo weight allowed during landing.

The most recent performance curves available date from 1999 and are shown. However, upgrades since then have increased performance. Shuttle payload capabilities for several specific reference missions are given. Each additional crew member above the normal number planned for that mission type reduces the payload capability by 225 kg (500 lbm).

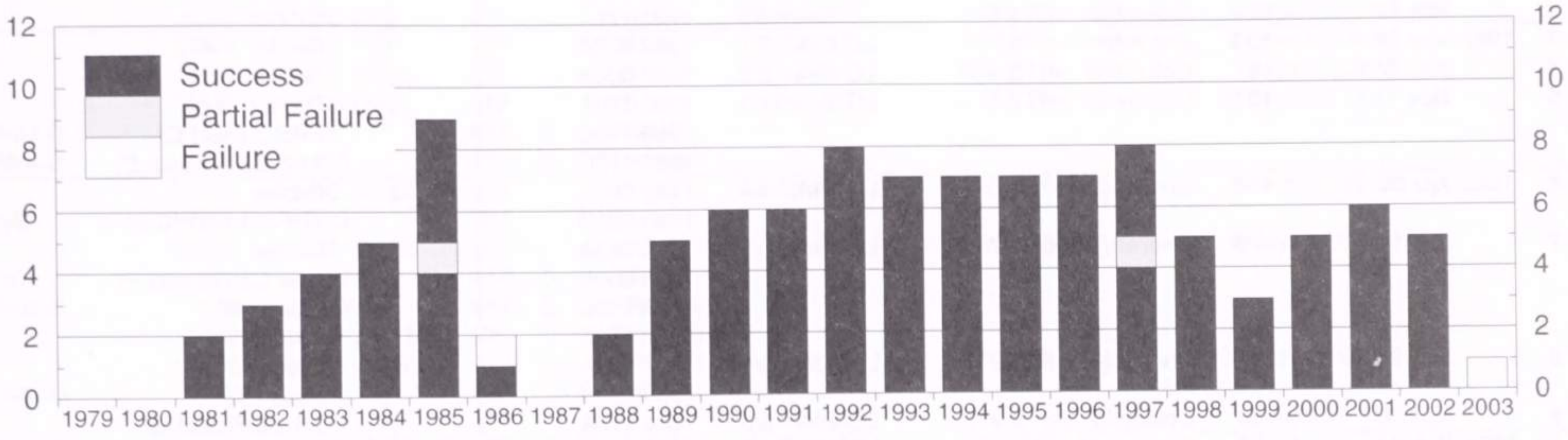
Spacecraft Deployment Mission	204 km (110 nmi), 28.45 deg	28,800 kg (63,500 lbm)
Science Platform	278 km (150 nmi), 28.45 deg	27,575 kg (60.800 lbm)
Spacecraft Servicing Mission	592 km (320 nmi), 28.45 deg	18,400 kg (40,600 lbm)
ISS Mission	407 km (220 nmi) 51.6 deg	18,300 kg (40,300 lbm)



Space Shuttle: Performance to LEO from Kennedy Space Center

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)

Total Orbital Flights	113
Launch Vehicle Successes	109
Launch Vehicle Partial Failures	2
Launch Vehicle Failures	2

Year	Total/Failures	OV 099 *Challenger* T/F	OV 102 *Columbia* T/F	OV 103 *Discovery* T/F	OV 104 *Atlantis* T/F	OV 105 *Endeavour* T/F
Total	113/2	10/1	28/1	30/0	26/0	19/0
1981	2/0		2			
1982	3/0		3			
1983	4/0	3	1			
1984	5/0	3		2		
1985	9/0	3		4	2	
1986	2/1	1/1	1			
1987	0/0					
1988	2/0			1	1	
1989	5/0		1	2	2	
1990	6/0		2	2	2	
1991	6/0		1	2	3	
1992	8/0		2	2	2	2
1993	7/0		2	2		3
1994	7/0		2	2	1	2
1995	7/0		1	2	2	2
1996	7/0		3		2	2
1997	8/0		3	2	3	
1998	5/0		1	2		2
1999	3/0		1	2		
2000	5/0			1	2	2
2001	6/0			2	2	2
2002	5/0		1		2	2
2003	1/1		1/1			

Flight History

		Date (UTC)	Launch Interval (days)	Model	Flight Designation	Pad	Crew Up/Dn	Payload Designation		P/L up	P/L down	Payload Name
T	1	1981 Apr 12	—	Columbia	STS 1	LC 39A	2/2	1981 034A		⇑	⇓	Orbiter
T	2	Nov 12	214	Columbia	STS 2	LC 39A	2/2	1981 111A		⇑	⇓	Orbiter
T	3	1982 Mar 22	130	Columbia	STS 3	LC 39A	2/2	1982 022A		⇑	⇓	Orbiter
T	4	Jun 27	97	Columbia	STS 4	LC 39A	2/2	1982 065A		⇑	⇓	Orbiter
	5	Nov 11	137	Columbia	STS 5	LC 39A	4/4	1982 110A		⇑	⇓	Orbiter
								1982 110B	C	⇑		SBS 3 (SBS C)
								1982 110C	C	⇑		Anik C3 (Telesat E)
	6	1983 Apr 04	144	Challenger	STS 6	LC 39A	4/4	1983 026A		⇑	⇓	Orbiter
S								1983 026B		⇑		TDRS 1 (TDRS A)
	7	Jun 18	75	Challenger	STS 7	LC 39A	5/5	1983 059A		⇑	⇓	Orbiter
								1983 059B	C	⇑		Anik C2 (Telesat F)
								1983 059C	C	⇑		Palapa B1
								1932 059F		⇑	⇓	SPAS 01
	8	Aug 30	73	Challenger	STS 8	LC 39A	5/5	1983 089A		⇑	⇓	Orbiter
								1983 089B		⇑		Insat 1B
	9	Nov 28	90	Columbia	STS 9	LC 39A	6/6	1983 116A		⇑	⇓	Orbiter/Spacelab
	10	1984 Feb 03	67	Challenger	41-B	LC 39A	5/5	1984 011A		⇑	⇓	Orbiter
S								1984 011B	C	⇑		Westar 6
								1984 011C		⇑		IRT (SPAS 1A)
S								1984 011D	C	⇑		Palapa B2
	11	Apr 06	63	Challenger	41-C	LC 39A	5/5	1984 034A		⇑	⇓	Orbiter
								1984 034B		⇑		LDEF
								1980 014A	R			Solar Maximum Mission [Repair]
	12	Aug 30	146	Discovery	41-D	LC 39A	6/6	1984 093A		⇑	⇓	Orbiter
								1984 093B	C	⇑		SBS 4 (SBS D)
								1984 093C	C	⇑		Leasat 2 (Syncom 402)
								1984 093D	C	⇑		Telstar 3C
	13	Oct 05	36	Challenger	41-G	LC 39A	7/7	1984 108A		⇑	⇓	Orbiter
								1984 108B		⇑		ERBS
	14	Nov 08	34	Discovery	51-A	LC 39A	5/5	1984 113A		⇑	⇓	Orbiter
								1984 113B	C	⇑		Anik D2 (Telesat H)
								1984 113C	C	⇑		Leasat 1 (Syncom 401)
								1984 011B	R		⇓	Westar 6
								1984 011D	R		⇓	Palapa B2
	15	1985 Jan 24	77	Discovery	51-C	LC 39A	5/5	1985 010A		⇑	⇓	Orbiter
								1985 010B		⇑		USA 8
	16	Apr 12	78	Discovery	51-D	LC 39A	7/7	1985 028A		⇑	⇓	Orbiter
								1985 028B	C	⇑		Anik C1 (Telesat I)
S								1985 028C	C	⇑		Leasat 3 (Syncom 403)
	17	Apr 29	17	Challenger	51-B	LC 39A	7/7	1985 034A		⇑	⇓	Orbiter/Spacelab 3
								1985 034B		⇑		Nusat
	18	Jun 17	49	Discovery	51-G	LC 39A	7/7	1985 048A		⇑	⇓	Orbiter
								1985 048B	C	⇑		Morelos 1 (Morelos A)
								1985 048C	C	⇑		Arabsat 1B
								1985 048D	C	⇑		Telstar 303
								1985 048E		⇑	⇓	Spartan 101
P	**19**	**Jul 29**	**42**	**Challenger**	**51-F**	**LC 39A**	**7/7**	1985 063A		⇑	⇓	Orbiter/Spacelab 2
								1985 063B		⇑	⇓	PDP
	20	Aug 27	29	Discovery	51-I	LC 39A	5/5	1985 076A		⇑	⇓	Orbiter
								1985 076B	C	⇑		Optus 1 (Aussat 1)
								1985 076C	C	⇑		ASC 1
								1985 076D	C	⇑		Leasat 4 (Syncom 404)
								1985 028C	R			Leasat 3 [Repair]
	21	Oct 03	37	Atlantis	51-J	LC 39A	5/5	1985 092A		⇑	⇓	Orbiter
								1985 092B	C	⇑		DSCS 302 (USA 11)
								1982 092C	C	⇑		DSCS 303 (USA 12)
	22	Oct 30	27	Challenger	61-A	LC 39A	8/8	1985 104A		⇑	⇓	Orbiter/SpaceLab D1
								1985 104B		⇑		GLOMR
	23	Nov 27	28	Atlantis	61-B	LC 39A	7/7	1985 109A		⇑	⇓	Orbiter
								1985 109B	C	⇑		Morelos 2 (Morelos B)
								1985 109C	C	⇑		Optus 2 (Aussat 2)
								1985 109D	C	⇑		Satcom Ku-2
								1985 109E		⇑		OEX
	24	1986 Jan 12	46	Columbia	61-C	LC 39A	7/7	1986 003A		⇑	⇓	Orbiter
								1986 003B		⇑		Satcom Ku-1

Launch complexes LC-39A and B are at the Kennedy Space Center.

Some Space Shuttle missions were military in nature and full details have not been disclosed.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

R = Rendezvous with spacecraft already in orbit; D = Docking with space station [station identified in brackets under payload column]

Flight History

Upper Stage	Mass up (kg)	Orbit (Incl)	Market	Country	Date down	Landing	Duration
	4911	LEO (40)	CIV	USA	Apr 14	EAFB Rwy 23	2 days 06:20:32
	8520	LEO (38)	CIV	USA	Nov 14	EAFB Rwy 23	2 days 06:13:13
	10304	LEO (38)	CIV	USA	Mar 30	WSMR Rwy 17	8 days 00:04:45
	11113	LEO (28.5)	CIV	USA	Jul 04	EAFB Rwy 22	7 days 01:09:40
	14555	LEO (28.5)	CIV	USA	Nov 16	EAFB Rwy 22	5 days 02:14:26
PAM-D	1117	GTO	CML	USA			
PAM-D	1230	GTO	CML	Canada			
	21312	LEO (28.5)	CIV	USA	Apr 09	EAFB Rwy 22	5 days 23:00:42
IUS	2268	GTO	CIV	USA			
	16844	LEO (28.5)	CIV	USA	Jun 24	EAFB Rwy 15	6 days 02:23:59
PAM-D	1238	GTO	CML	Canada			
PAM-D	1200	GTO	CML	Indonesia			
	1442	LEO (28.5)	CIV	Germany			
	13646	LEO (28.5)	CIV	USA	Sep 05	EAFB Rwy 22	6 days 01:08:43
PAM-D	1152	GTO	CIV	India			
	15093	LEO (57)	CIV	USA	Dec 08	EAFB Rwy 17	10 days 07:47:24
	15367	LEO (28.5)	CIV	USA	Feb 11	KSC Rwy 15	7 days 23:15:55
PAM-D	1200	LEO (28.5)	CML	USA			
	91	LEO (28.4)	CIV	USA			
PAM-D	1200	LEO (28.5)	CML	Indonesia			
	17362	LEO (28.5)	CIV	USA	Apr 13	EAFB Rwy 17	6 days 23:40:07
	3625	LEO (28.5)	CIV	USA			
		LEO (28.5)	CIV	USA			
	21559	LEO (28.5)	CIV	USA	Sep 05	EAFB Rwy 17	6 days 00:56:04
PAM-D	1117	GTO	CML	USA			
PKM	3230	GTO	CML	USA			
PAM-D	3423	GTO	CML	USA			
	10647	LEO (57)	CIV	USA	Oct 13	KSC Rwy 33	8 days 05:23:33
	226	LEO (57)	CIV	USA			
	20556	LEO (28.5)	CIV	USA	Nov 16	KSC Rwy 15	7 days 23:44:56
PAM-D	1238	GTO	CML	Canada			
PKM	3230	GTO	CML	USA			
		Retrieve	CML	USA			
		Retrieve	CML	Indonesia			
		LEO (28.5)	MIL	USA	Jan 27	KSC Rwy 15	3 days 01:33:23
IUS			MIL	USA			
	16254	LEO (28.5)	CIV	USA	Apr 19	KSC Rwy 33	6 days 23:55:23
PAM-D	1238	GTO	CML	Canada			
PKM	3230	LEO (28.5)	CML	USA			
	14250	LEO (57)	CIV	USA	May 06	EAFB Rwy 17	7 days 00:08:46
	52	LEO (57)	CIV	USA			
	20180	LEO (28.5)	CIV	USA	Jun 24	EAFB Rwy 23	7 days 01:38:52
PAM-D	1140	GTO	CML	Mexico			
PAM-D	1270	GTO	CML	Saudi Arabia			
PAM-D	3423	GTO	CML	USA			
	1008	LEO (28.4)	CIV	USA			
	15608	LEO (49.5)	CIV	USA	Aug 06	EAFB Rwy 23	7 days 22:45:26
	285	LEO (49.4)	CIV	USA			
	19958	LEO (28.5)	CIV	USA	Sep 03	EAFB Rwy 23	7 days 02:17:42
PAM-D	1250	GTO	CML	Australia			
PAM-D	1271	GTO	CML	USA			
PKM	3230	GTO	CML	USA			
		GTO	CML	USA			
		LEO (28.5)	MIL	USA	Oct 07	EAFB Rwy 23	4 days 01:44:38
IABS	1043		MIL	USA			
IABS	10443		MIL	USA			
	14456	LEO (57)	CIV	USA	Nov 06	EAFB Rwy 17	7 days 00:44:51
	68	LEO (57)	CIV	USA			
	21797	LEO (28.5)	CIV	USA	Dec 03	EAFB Rwy 22	6 days 21:04:49
PAM-D	1140	GTO	CML	Mexico			
PAM-D	1259	GTO	CML	Australia			
PAM-D2	1926	GTO	CML	USA			
	16	LEO (28.5)	CIV	USA			
	14729	LEO (28.5)	CIV	USA	Jan 18	EAFB Rwy 22	6 days 02:03:51
PAM-D2	1923	GTO	CML	USA			

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Flight History

		Date (UTC)	Launch Interval (days)	Model	Flight Designation	Pad	Crew Up/Dn	Payload Designation		P/L up	P/L down	Payload Name
F	25	Jan 28	16	Challenger	51-L	LC 39B	7/	1986 F01A				Orbiter
								1986 F01B				TDRS B
								1986 F01C				Spartan 203
	26	1988 Sep 29	975	Discovery	STS 26	LC 39B	5/5	1988 091A		⇑	⇓	Orbiter
								1988 091B		⇑		TDRS 3 (TDRS C)
	27	Dec 02	64	Atlantis	STS 27	LC 39B	5/5	1988 106A		⇑	⇓	Orbiter
								1988 106B		⇑		USA 34
	28	1989 Mar 13	101	Discovery	STS 29	LC 39B	5/5	1989 021A		⇑	⇓	Orbiter
								1989 021B		⇑		TDRS 4 (TDRS D)
	29	May 04	52	Atlantis	STS 30	LC 39B	5/5	1989 033A		⇑	⇓	Orbiter
								1989 033B		⇑		Magellan
	30	Aug 08	96	Columbia	STS 28	LC 39B	5/5	1989 061A		⇑	⇓	Orbiter
								1989 061B	C	⇑		USA 40
								1989 061C	C	⇑		USA 41
	31	Oct 18	71	Atlantis	STS 34	LC 39B	5/5	1989 084A		⇑	⇓	Orbiter
								1989 084B		⇑		Galileo
	32	Nov 23	36	Discovery	STS 33	LC 39B	5/5	1989 090A		⇑	⇓	Orbiter
								1989 090B		⇑		USA 48
	33	1990 Jan 09	47	Columbia	STS 32	LC 39A	5/5	1990 002A		⇑	⇓	Orbiter
								1990 002B		⇑		Leasat 5 (Syncom 405)
								1984 034B	R		⇓	LDEF
	34	Feb 28	50	Atlantis	STS 36	LC 39A	5/5	1990 019A		⇑	⇓	Orbiter
								1990 019B		⇑		USA 53
	35	Apr 24	55	Discovery	STS 31	LC 39B	5/5	1990 037A		⇑	⇓	Orbiter
	34							1990 037B		⇑		Hubble Space Telescope
	36	Oct 06	165	Discovery	STS 41	LC 39B	5/5	1990 090A		⇑	⇓	Orbiter
	34							1990 090B		⇑		Ulysses
	37	Nov 15	40	Atlantis	STS 38	LC 39A	5/5	1990 097A		⇑	⇓	Orbiter
								1990 097B		⇑		USA 67
	38	Dec 02	17	Columbia	STS 35	LC 39B	7/7	1990 106A		⇑	⇓	Orbiter
	39	1991 Apr 05	124	Atlantis	STS 37	LC 39B	5/5	1991 027A		⇑	⇓	Orbiter
								1991 027B		⇑		Compton Gamma Ray Observatory
	40	Apr 28	23	Discovery	STS 39	LC 39A	7/7	1991 031A		⇑	⇓	Orbiter
								1991 031B		⇑	⇓	IBSS/SPAS 2
								1991 031C		⇑		USA 70
								1991 031D-F		⇑		CRO A/B/C
	41	Jun 05	38	Columbia	STS 40	LC 39B	7/7	1991 040A		⇑	⇓	Orbiter/Spacelab LS 1
	42	Aug 02	58	Atlantis	STS 43	LC 39A	5/5	1991 054A		⇑	⇓	Orbiter
								1991 054B		⇑		TDRS 5 (TDRS E)
	43	Sep 12	41	Discovery	STS 48	LC 39A	5/5	1991 063A		⇑	⇓	Orbiter
								1991 063B		⇑		UARS
	44	Nov 24	73	Atlantis	STS 44	LC 39A	6/6	1991 080A		⇑	⇓	Orbiter
								1991 080B		⇑		DSP 16
	45	1992 Jan 22	59	Discovery	STS 42	LC 39A	7/7	1992 002A		⇑	⇓	Orbiter/Spacelab IML 1
	46	Mar 24	62	Atlantis	STS 45	LC 39A	7/7	1992 015A		⇑	⇓	Orbiter
	47	May 07	44	Endeavour	STS 49	LC 39B	7/7	1992 026A		⇑	⇓	Orbiter
								1990 021A	R			Intelsat 603 [Repair]
	48	Jun 25	49	Columbia	STS 50	LC 39A	7/7	1992 034A		⇑	⇓	Orbiter/Spacelab
	49	Jul 31	36	Atlantis	STS 46	LC 39B	7/7	1992 049A		⇑	⇓	Orbiter/TSS-1
								1992 049B		⇑		Eureca 1
	50	Sep 12	43	Endeavour	STS 47	LC 39B	7/7	1992 061A		⇑	⇓	Orbiter/Spacelab J
	51	Oct 22	40	Columbia	STS 52	LC 39B	6/6	1992 070A		⇑	⇓	Orbiter
								1992 070B		⇑		Lageos 2
								1992 070C		⇑		CTA
	52	Dec 02	41	Discovery	STS 53	LC 39A	5/5	1992 086A		⇑	⇓	Orbiter
								1992 086B		⇑		DOD 1
	53	1993 Jan 13	42	Endeavour	STS 54	LC 39B	5/5	1993 003A		⇑	⇓	Orbiter
								1993 003B		⇑		TDRS 6 (TDRS F)
	54	Apr 08	85	Discovery	STS 56	LC 39B	5/5	1993 023A		⇑	⇓	Orbiter
								1993 023B		⇑	⇓	Spartan 201
	55	Apr 26	18	Columbia	STS 55	LC 39A	7/7	1993 027A		⇑	⇓	Orbiter/Spacelab D2
	56	Jun 21	56	Endeavour	STS 57	LC 39B	6/6	1993 037A		⇑	⇓	Orbiter/Spacehab 1
								1992 049B	R		⇓	Eureca 1
	57	Sep 12	83	Discovery	STS 51	LC 39B	5/5	1993 058A		⇑	⇓	Orbiter
								1993 058B		⇑		ACTS
								1993 058C		⇑	⇓	Orefeus-SPAS

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FLIGHT HISTORY

Upper Stage	Mass up (kg)	Orbit (Incl)	Market	Country	Date down	Landing	Duration
	23733	LEO (28.5)	CIV	USA			
IUS			CIV	USA			
			CIV	USA			
	21088	LEO (28.5)	CIV	USA	Oct 03	EAFB Rwy 17	4 days 01:00:11
IUS	2200	GEO	CIV	USA			
		LEO (57)	MIL	USA	Dec 06	EAFB Rwy 17	4 days 09:05:35
			MIL	USA			
	17285	LEO (28.5)	CIV	USA	Mar 18	EAFB Rwy 22	4 days 23:38:52
IUS	2120	GEO	CIV	USA			
	20839	LEO (28.9)	CIV	USA	May 08	EAFB Rwy 22	4 days 00:57:31
IUS	3444	Venus	CIV	USA			
		LEO (57)	MIL	USA	Aug 13	EAFB Rwy 17	5 days 01:00:09
			MIL	USA			
			MIL	USA			
	22070	LEO (34.3)	CIV	USA	Oct 23	EAFB Rwy 23	4 days 23:39:24
IUS	3881	Jupiter	CIV	USA			
		LEO (28.5)	MIL	USA	Nov 27	EAFB Rwy 04	5 days 00:06:49
IUS			MIL	USA			
	12018	LEO (28.5)	CIV	USA	Jan 20	EAFB Rwy 22	10 days 21:37:00
PKM	12018	GTO	MIL	USA			
		Retrieve	CIV	USA			
		LEO (62)	MIL	USA	Mar 04	EAFB Rwy 23	4 days 10:18:23
			MIL	USA			
	13010	LEO (28.5)	CIV	USA	Apr 29	EAFB Rwy 22	5 days 01:16:05
	10863	LEO (28.5)	CIV	USA			
	22147	LEO (28.5)	CIV	USA	Oct 10	EAFB Rwy 22	4 days 02:10:00
IUS	367	Solar	CIV	Europe			
		LEO (28.5)	MIL	USA	Nov 20	KSC Rwy 33	4 days 21:55:22
			MIL	USA			
	11946	LEO (28.5)	CIV	USA	Dec 10	EAFB Rwy 22	8 days 23:05:08
	16616	LEO (28.5)	CIV	USA	Apr 11	EAFB Rwy 33	5 days 23:32:44
	15620	LEO (28.5)	CIV	USA			
	9716	LEO (56.9)	MIL	USA	May 06	KSC Rwy 15	8 days 07:22:22
	1901	LEO (56.9)	MIL	USA			
			MIL	USA			
	3@197	LEO (56.9)	MIL	USA			
	11770	LEO (39)	CIV	USA	Jun 14	EAFB Rwy 22	9 days 02:14:20
	21271	LEO (28.5)	CIV	USA	Aug 11	KSC Rwy 15	8 days 21:21:25
IUS	2200	GEO	CIV	USA			
	7857	LEO (57)	CIV	USA	Sep 18	EAFB Rwy 22	5 days 08:27:34
	6795	LEO (57)	CIV	USA			
	20249	LEO (28.5)	CIV	USA	Dec 01	EAFB Rwy 05	6 days 22:50:42
IUS	2355		MIL	USA			
	13005	LEO (57)	CIV	USA	Jan 30	EAFB Rwy 22	8 days 01:14:45
	8023	LEO (57)	CIV	USA	Apr 02	KSC Rwy 33	8 days 22:09:25
	14790	LEO (28.4)	CIV	USA	May 16	EAFB Rwy 22	8 days 21:17:39
		GTO	CML	International			
	11157	LEO (28.5)	CIV	USA	Jul 09	KSC Rwy 33	13 days 19:30:04
	12970	LEO (28.5)	CIV	USA	Aug 08	KSC Rwy 33	7 days 23:16:07
	4491	LEO (28.5)	CIV	Germany			
	12776	LEO (57)	CIV	USA	Sep 20	KSC Rwy 33	7 days 22:31:11
	9109	LEO (28.5)	CIV	USA	Nov 01	KSC Rwy 33	9 days 20:56:13
IRIS	405	LEO (28.5)	CIV	Italy			
	82	LEO (28.5)	CIV	Canada			
	11872	LEO (57)	MIL	USA	Dec 09	EAFB Rwy 22	7 days 07:19:17
			MIL	USA			
	21163	LEO (28.5)	CIV	USA	Jan 19	KSC Rwy 33	5 days 23:38:17
IUS	2530	GEO	CIV	USA			
	7444	LEO (57)	CIV	USA	Apr 17	KSC Rwy 33	9 days 06:09:21
	1289	LEO (57)	CIV	USA			
	12189	LEO (28.5)	CIV	USA	May 06	EAFB Rwy 22	9 days 23:39:59
	8934	LEO (28.5)	CIV	USA	Jul 01	KSC Rwy 33	9 days 23:45:00
		Retrieve	CIV	Germany			
	19366	LEO (28.5)	CIV	USA	Sep 22	KSC Rwy 15	9 days 20:11:00
TOS	2767	GTO	CIV	USA			
	3202	LEO (28.5)	CIV	USA			

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Flight History

		Date (UTC)	Launch Interval (days)	Model	Flight Designation	Pad	Crew Up/Dn	Payload Designation	P/L up	P/L down	Payload Name
	58	Oct 18	36	Columbia	STS 58	LC 39B	7/7	1993 065A	⇑	⇓	Orbiter/Spacelab LS 2
	59	Dec 02	45	Endeavour	STS 61	LC 39B	7/7	1993 075A	⇑	⇓	Orbiter
								1990 037B R			Hubble Space Telescope [Service]
	60	1994 Feb 03	63	Discovery	STS 60	LC 39A	6/6	1994 006A	⇑	⇓	Orbiter/SpaceHab 2/Wake Shield Facility 1
								1994 006H	⇑		BREMSAT
								1994 006B-G M	⇑		Oderacs A/B/C/D/E/F
	61	Mar 04	29	Columbia	STS 62	LC 39B	5/5	1994 015A	⇑	⇓	Orbiter
	62	Apr 09	36	Endeavour	STS 59	LC 39A	6/6	1994 020A	⇑	⇓	Orbiter
	63	Jul 08	90	Columbia	STS 65	LC 39A	7/7	1994 039A	⇑	⇓	Orbiter/Spacelab IML 2
	64	Sep 09	63	Discovery	STS 64	LC 39B	6/6	1994 059A	⇑	⇓	Orbiter
								1994 059B	⇑	⇓	Spartan 201
	65	Sep 30	21	Endeavour	STS 68	LC 39A	6/6	1994 062A	⇑	⇓	Orbiter
	66	Nov 03	34	Atlantis	STS 66	LC 39B	6/6	1994 073A	⇑	⇓	Orbiter
								1994 073B	⇑	⇓	CRISTA-SPAS
	67	1995 Feb 03	92	Discovery	STS 63	LC 39B	6/6	1995 004A R	⇑	⇓	Orbiter/Spacehab 3
								1995 004B	⇑	⇓	Spartan 204
								1995 004C M	⇑		ODERCS 2A/B/C/D/E/F
	68	Mar 02	27	Endeavour	STS 67	LC 39A	7/7	1995 007A	⇑	⇓	Orbiter
	69	Jun 27	117	Atlantis	STS 71	LC 39A	7/8	1995 030A D	⇑	⇓	Orbiter/Spacelab (S/MM-1)
	70	Jul 13	16	Discovery	STS 70	LC 39B	5/5	1995 035A	⇑	⇓	Orbiter
								1995 035B	⇑		TDRS 7 (TDRS G)
	71	Sep 07	56	Endeavour	STS 69	LC 39A	5/5	1995 048A	⇑	⇓	Orbiter
								1995 048B	⇑	⇓	Spartan 201-03
								1995 048C	⇑	⇓	Wake Shield Facility 2
	72	Oct 20	43	Columbia	STS 73	LC 39B	7/7	1995 056A	⇑	⇓	Orbiter/Spacelab USML 2
	73	Nov 12	23	Atlantis	STS 74	LC 39A	5/5	1995 061A D	⇑	⇓	Orbiter/Spacehab Double Module
	74	1996 Jan 11	60	Endeavour	STS 72	LC 39B	6/6	1996 001A	⇑	⇓	Orbiter
								1996 001B	⇑	⇓	OAST-Flyer (Spartan 206)
								1995 011A R		⇓	SFU
	75	Feb 22	42	Columbia	STS 75	LC 39B	7/7	1996 012A	⇑	⇓	Orbiter
S								1996 012B	⇑		TSS-1R
	76	Mar 22	29	Atlantis	STS 76	LC 39B	6/5	1996 018A D	⇑	⇓	Orbiter/Spacehab-SM (S/MM-3)
	77	May 19	58	Endeavour	STS 77	LC 39B	6/6	1996 032A	⇑	⇓	Orbiter/Spacehab 4
								1996 032B	⇑	⇓	Spartan 207
								1996 032C	⇑		IAE
								1996 032D	⇑		PAMS-STU
	78	Jun 20	32	Columbia	STS 78	LC 39B	7/7	1996 036A	⇑	⇓	Orbiter/Spacelab LMS
	79	Sep 16	88	Atlantis	STS 79	LC 39A	7/7	1996 057A D	⇑	⇓	Orbiter/Spacehab Double Module
	80	Nov 19	64	Columbia	STS 80	LC 39B	5/5	1996 065A	⇑	⇓	Orbiter
								1996 065B	⇑	⇓	Orfeus-SPAS
	80							1996 065C	⇑	⇓	Wake Shield Facility 3
	81	1997 Jan 12	54	Atlantis	STS 81	LC 39B	6/6	1997 001A D	⇑	⇓	Orbiter/Spacehab Double Module
	82	Feb 11	30	Discovery	STS 82	LC 39A	7/7	1997 004A	⇑	⇓	Orbiter
								1990 037B R			Hubble Space Telescope [Service]
P	**83**	**Apr 04**	**52**	**Columbia**	**STS 83**	**LC 39A**	**7/7**	1997 013A	⇑	⇓	Orbiter/Spacelab MSL-1
	84	May 15	41	Atlantis	STS 84	LC 39A	8/8	1997 023A D	⇑	⇓	Orbiter/Spacehab Double Module
	85	Jul 01	47	Columbia	STS 94	LC 39A	7/7	1997 032A	⇑	⇓	Orbiter/Spacelab MSL-1R [Reflight]
	86	Aug 07	37	Discovery	STS 85	LC 39A	6/6	1997 039A	⇑	⇓	Orbiter
								1997 039B	⇑	⇓	Crista-SPAS 02
	87	Sep 26	50	Atlantis	STS 86	LC 39A	7/7	1997 055A D	⇑	⇓	Orbiter/Spacehab Double Module
	88	Nov 19	54	Columbia	STS 87	LC 39B	6/6	1997 073A	⇑	⇓	Orbiter
								1997 073B	⇑	⇓	Spartan 201-04
	89	1998 Jan 23	65	Endeavour	STS 89	LC 39A	7/7	1998 003A D	⇑	⇓	Orbiter/Spacehab Double Module
	90	Apr 17	84	Columbia	STS 90	LC 39B	7/7	1998 022A	⇑	⇓	Orbiter/Neurolab
	91	Jun 02	46	Discovery	STS 91	LC 39A	6/7	1998 034A D	⇑	⇓	Orbiter/Spacehab [Mir]
	92	Oct 29	149	Discovery	STS 95	LC 39B	7/7	1998 064A	⇑	⇓	Orbiter/Spacehab
								1998 064B	⇑	⇓	Spartan 201-05
								1998 064C	⇑		PANSAT
	93	Dec 04	36	Endeavour	STS 88	LC 39A	6/6	1998 069A D	⇑	⇓	Orbiter (ISS-01-2A)
								1998 069F	⇑		Unity (Node 1)
		Dec 14						1998 069B	⇑		SAC A
		Dec 15						1998 069C	⇑		MightySat 1
	94	1999 May 28	175	Discovery	STS 96	LC 39B	7/7	1999 030A D	⇑	⇓	Orbiter/Spacehab Double Module/ Integrated Cargo Carrier
		Jun 05						1999 030B	⇑		Starshine

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FLIGHT HISTORY

Upper Stage	Mass up (kg)	Orbit (Incl)	Market	Country	Date down	Landing	Duration
	10521	LEO (39)	CIV	USA	Nov 01	EAFB Rwy 22	14 days 00:12:32
	8014	LEO (28.5)	CIV	USA	Dec 13	KSC Rwy 33	10 days 19:58:33
		LEO (28.5)	CIV	USA			
	13010	LEO (57)	CIV	USA	Feb 11	KSC Rwy 33	8 days 07:10:13
	63	LEO (57)	CIV	Germany			
		LEO (57)	CIV	USA			
	8873	LEO (39)	CIV	USA	Mar 18	KSC Rwy 33	13 days 23:17:28
	12494	LEO (57)	CIV	USA	Apr 20	EAFB Rwy 22	11 days 05:49:30
	10815	LEO (28.5)	CIV	USA	Jul 23	KSC Rwy 33	14 days 17:56:00
	9264	LEO (57)	CIV	USA	Sep 20	EAFB Rwy 04	10 days 22:49:57
	1288	LEO (57)	CIV	USA			
	12515	LEO (57)	CIV	USA	Oct 11	EAFB Rwy 22	11 days 05:47:08
	10548	LEO (57)	CIV	USA	Nov 14	EAFB Rwy 22	10 days 22:34:51
	3260	LEO (57)	CIV	USA			
	8644	STA (Mir)	CIV	USA	Feb 11	KSC Rwy 15	8 days 06:29:35
	1167	LEO (51.6)	CIV	USA			
		LEO (51.6)	CIV	USA			
	13120	LEO (28.5)	CIV	USA	Mar 18	EAFB Rwy 22	16 days 15:09:46
	12195	STA (Mir)	CIV	USA	Jul 07	KSC Rwy 15	9 days 19:23:08
	20166	LEO (28.5)	CIV	USA	Jul 22	KSC Rwy 33	8 days 22:20:05
IUS	2221	GEO	CIV	USA			
	11503	LEO (28.5)	CIV	USA	Sep 18	KSC Rwy 33	10 days 20:29:52
	1287	LEO (28.5)	CIV	USA			
	1977	LEO (28.5)	CIV	USA			
	15255	LEO (39)	CIV	USA	Nov 05	KSC Rwy 33	15 days 21:53:16
	6137	STA (Mir)	CIV	USA	Nov 20	KSC Rwy 33	8 days 04:31:42
	6512	LEO (28.5)	CIV	USA	Jan 20	KSC Rwy 33	8 days 22:01:47
	1197	LEO (28.5)	CIV	USA			
		Retrieve	CIV	Japan			
	10596	LEO (28.5)	CIV	USA	Mar 09	KSC Rwy 33	15 days 17:40:21
	518	LEO (28.5)	CIV	Italy			
	6755	STA (Mir)	CIV	USA	Mar 31	EAFB Rwy 22	9 days 05:16:48
	12237	LEO (39)	CIV	USA	May 29	KSC Rwy 33	10 days 00:40:10
	1294	LEO (39)	CIV	USA			
	60	LEO (39)	CIV	USA			
	52	LEO (39)	CIV	USA			
	10679	LEO (39)	CIV	USA	Jul 07	KSC Rwy 33	16 days 21:48:30
	8855	STA (Mir)	CIV	USA	Sep 26	KSC Rwy 15	10 days 03:19:28
	9750	LEO (28.5)	CIV	USA	Dec 07	KSC Rwy 33	17 days 15:54:28
	3567	LEO (28.5)	CIV	Germany			
	2106	LEO (28.5)	CIV	USA			
	8766	STA (Mir)	CIV	USA	Jan 22	KSC Rwy 33	10 days 04:56:31
	7593	LEO (28.5)	CIV	USA	Feb 21	KSC Rwy 15	9 days 23:38:09
		LEO (28.5)	CIV	USA			
	11583	LEO (28.5)	CIV	USA	Apr 04	KSC Rwy 33	3 days 23:13:38
	8974	STA (Mir)	CIV	USA	May 24	KSC Rwy 33	9 days 05:20:47
	11601	LEO (28.5)	CIV	USA	Jul 17	KSC Rwy 33	15 days 16:45:29
	11335	LEO (57)	CIV	USA	Aug 19	KSC Rwy 33	11 days 20:28:07
		LEO (57)	CIV	USA			
	9315	STA (Mir)	CIV	USA	Oct 06	KSC Rwy 15	10 days 19:22:12
	10041	LEO (28.5)	CIV	USA	Dec 05	KSC Rwy 33	15 days 16:35:01
	1290	LEO (28.5)	CIV	USA			
	9955	STA (Mir)	CIV	USA	Jan 31	KSC Rwy 15	8 days 19:48:04
	11865	LEO (39)	CIV	USA	May 03	KSC Rwy 33	15 days 21:50:58
	11761	STA (Mir)	CIV	USA	Jun 12	KSC Rwy 15	9 days 19:55:01
	12940	LEO (28.5)	CIV	USA	Nov 07	KSC Rwy 33	8 days 21:44:56
	1136	LEO (28.5)	CIV	USA			
	68	LEO (28.5)	NGO	USA			
		STA (Mir)	CIV	USA	Dec 15	KSC Rwy 15	11 days 19:18:47
	11615	STA (Mir)	CIV	USA			
	268	LEO (57.6)	CIV	Argentina			
	320	LEO (51.6)	MIL	USA			
	1618	STA (ISS)	CIV	USA	May 27	KSC Rwy 15	9 days 19:13:57

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Flight History

	Date (UTC)		Launch Interval (days)	Model	Flight Designation	Pad	Crew Up/Dn	Payload Designation		P/L up	P/L down	Payload Name
95		Jul 23	56	Columbia	STS 93	LC 39B	5/5	1999 040A		⇑	⇓	Orbiter
								1999 040B		⇑		Chandra (AXAF)
96		Dec 20	150	Discovery	STS 103	LC 39B	7/7	1999 069A		⇑	⇓	Orbiter
								1990 037B	R			Hubble Space Telescope [Service]
97	2000	Feb 11	53	Endeavour	STS 99	LC 39A	6/6	2000 010A		⇑	⇓	Orbiter [Shuttle Radar Topography Mission]
98		May 19	98	Atlantis	STS 101	LC 39A	7/7	2000 027A	D	⇑	⇓	Orbiter/Spacehab Double Module/Integrated Cargo Carrier (ISS-2A.2a)
99		Sep 08	112	Atlantis	STS 106	LC 39B	7/7	2000 053A	D	⇑	⇓	Orbiter/Spacehab Double Module/Integrated Cargo Carrier (ISS-2A.2b)
100		Oct 11	33	Discovery	STS 92	LC 39A	7/7	2000 062A	D	⇑	⇓	Orbiter (ISS-3A)
										⇑		Z1 Truss
										⇑		PMA 3
101		Dec 01	51	Endeavour	STS 97	LC 39B	5/5	2000 078A	D	⇑	⇓	Orbiter (ISS-4A)
										⇑		P6 Truss
102	2001	Feb 07	68	Atlantis	STS 98	LC 39A	5/5	2001 006A	D	⇑	⇓	Orbiter (ISS-5A)
								2001 006B		⇑		Destiny (US Laboratory Module)
103		Mar 08	29	Discovery	STS 102	LC 39B	7/7	2001 010A	D	⇑	⇓	Orbiter/Integrated Cargo Carrier (ISS-5A.1)
									B	⇑	⇓	MPLM Leonardo F-1
104		Apr 19	42	Endeavour	STS 100	LC 39A	7/7	2001 016A	D	⇑	⇓	Orbiter (ISS-6A)
									B	⇑	⇓	MPLM Raffaelo F-1
										⇑		Canadarm 2
105		Jul 12	84	Atlantis	STS 104	LC 39B	5/5	2001 028A	D	⇑	⇓	Orbiter (ISS-7A)
										⇑		Quest (Airlock)
106		Aug 10	29	Discovery	STS 105	LC 39A	7/7	2001 035A	D	⇑	⇓	Orbiter/Integrated Cargo Carrier (ISS-7A.1)
									B	⇑	⇓	MPLM Leonardo F-2
								2001 035B		⇑		SimpleSat
107		Dec 05	117	Endeavour	STS 108	LC 39B	7/7	2001 054A	D	⇑	⇓	Orbiter (ISS-UF-1)
									B	⇑	⇓	MPLM Raffaelo F-2
								2001 054B		⇑		Starshine 2
108	2002	Mar 01	86	Columbia	STS 109	LC 39A	7/7	2002 010A		⇑	⇓	Orbiter
								1990 037B	R			Hubble Space Telescope [Service]
109		Apr 08	38	Atlantis	STS 110	LC 39B	7/7	2002 018A	D	⇑	⇓	Orbiter (ISS-8A)
										⇑		S0 Truss
110		Jun 05	58	Endeavour	STS 111	LC 39A	7/7	2002 028A	D	⇑	⇓	Orbiter (ISS-UF-2)
										⇑		MSS
									B	⇑	⇓	MPLM Leonardo F-3
111		Oct 07	124	Atlantis	STS 112	LC 39B	6/6	2002 047A	D	⇑	⇓	Orbiter (ISS-9A)
										⇑		S1 Truss
112		Nov 24	48	Endeavour	STS 113	LC 39A	7/7	2002 052A	D	⇑	⇓	Orbiter (ISS-11A)
										⇑		P1 Truss
								2002 052B		⇑		MEPSI
F 113	**2003**	**Jan 16**	**53**	**Columbia**	**STS 107**	**LC 39A**	7/	2003 003A		⇑		Orbiter/Spacehab Research Double Module

Launch complexes LC-39A and B are at the Kennedy Space Center.

Some Space Shuttle missions were military in nature and full details have not been disclosed.

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

R = Rendezvous with spacecraft already in orbit; D = Docking with space station [station identified in brackets under payload column]

Flight History

Upper Stage	Mass up (kg)	Orbit (Incl)	Market	Country	Date down	Landing	Duration
		LEO (28.5)	CIV	USA	Jul 28	KSC Rwy 33	4 days 22:49:35
IUS	5867	EEO (28.5)	CIV	USA			
		LEO (28.5)	CIV	USA	Dec 20	KSC Rwy 33	7 days 23:10:45
		LEO (28.5)	CIV	USA			
		LEO (57)	CIV	USA	Feb 22	KSC Rwy 33	11 days 05:38:41
	2700	STA (ISS)	CIV	USA	May 29	KSC Rwy 15	9 days 20:09:10
	2993	STA (ISS)	CIV	USA	Sep 20	KSC Rwy 15	11 days 19:11:00
		STA (ISS)	CIV	USA	Oct 24	EAFB Rwy 22	12 days 21:42:37
	8300	STA (ISS)	CIV	USA			
	1156	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Dec 11	KSC Rwy 15	10 days 19:57:22
	7700	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Feb 20	EAFB Rwy 22	12 days 21:20:01
	14060	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Mar 08	KSC Rwy 15	12 days 19:49:35
		STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	May 01	EAFB Rwy 22	11 days 23:30:03
		STA (ISS)	CIV	USA			
	1800	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Jul 25	KSC Rwy 15	12 days 18:35:00
	6064	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Aug 22	KSC Rwy 15	11 days 21:12:40
	3073	STA (ISS)	CIV	USA			
	52	LEO (51.6)	CIV	USA			
		STA (ISS)	CIV	USA	Dec 17	KSC Rwy 15	11 days 19:35:46
		STA (ISS)	CIV	USA			
	38	LEO (51.6)	CIV	USA			
		LEO (28.5)	CIV	USA	Mar 12	KSC Rwy 33	10 days 22:09:50
		LEO (28.5)	CIV	USA			
		STA (ISS)	CIV	USA	Apr 19	KSC Rwy 33	10 days 19:42:40
	12250	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Jun 19	EAFB Rwy 22	13 days 22:34:54
	1500	STA (ISS)	CIV	USA			
	2541	STA (ISS)	CIV	USA			
		STA (ISS)	CIV	USA	Oct 18	KSC Rwy 33	10 days 19:57:48
	13971	STA (ISS)	CIV	USA			
	1969	STA (ISS)	CIV	USA	Dec 07	KSC Rwy 33	13 days 18:47:23
	12477	STA (ISS)	CIV	USA			
	2	LEO (51.6)	MIL	USA			
	?	LEO (39)	CIV	USA	Feb 01		15 days 22:20:21

Launch complexes LC-39A and B are at the Kennedy Space Center.

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Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

R = Rendezvous with spacecraft already in orbit; D = Docking with space station [station identified in brackets under payload column]

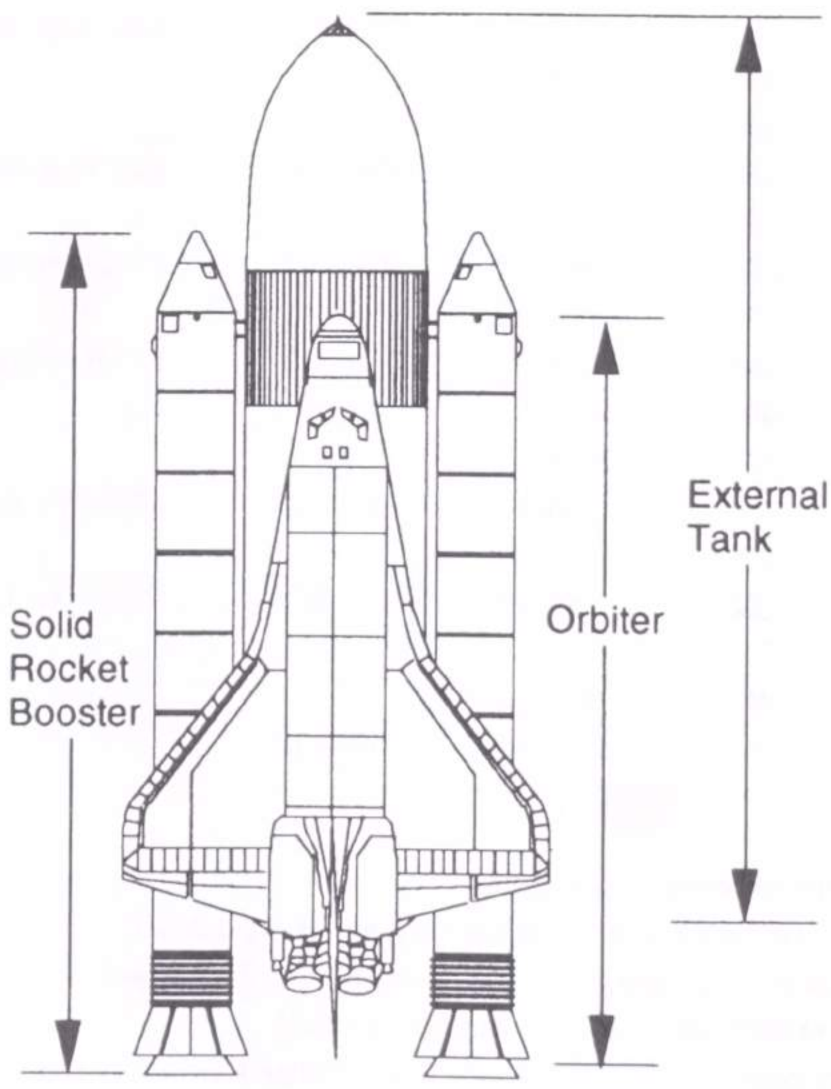
Flight History

Failure Descriptions:

S	1983 Apr 04	TDRS 1 (TDRS A)	1983 026B	Toward the end of the IUS second-stage burn, the Techroll gimbal joint on the nozzle of the SRM-2 motor failed. The spacecraft was able to separate and complete the GEO insertion maneuver with its own thrusters.
S	1984 Feb 03	Westar 6	1984 011B	PAM-D upper stage failed, satellite intended for GTO, but stranded in LEO.
S	1984 Feb 03	Palapa B2	1984 011D	PAM-D upper stage failed, satellite intended for GTO, but stranded in LEO.
S	1985 Apr 12	Leasat 3 (Syncom 403)	1985 028C	Perigee kick motor failed, satellite intended for GTO, but stranded in LEO.
P	1985 Jul 29	51-F	1985 063	Faulty temperature sensor incorrectly indicated that fuel turbine discharge temperature exceeded the limit. Therefore, one main engine was shut down at T+345 s, resulting in a much lower orbit than planned. All mission objectives achieved.
F	1986 Jan 28	51-L	1986 F01	At T+70 s, a burn-through of a SRB O-ring resulted in the rupturing of the external tank and the subsequent breakup of the orbiter; seven astronauts died.
S	1992 Jul 31	TSS-1	1992 049A	Tether reel jammed after deploying only 262 m instead of the planned 20.1 km, after numerous attempts to free the tether, operations were terminated, and the satellite was stowed for return to Earth.
S	1996 Feb 22	TSS-1R	1996 012B	Tether broke during deployment, satellite lost.
P	1997 Apr 04	STS 83	1997 013	Mission, planned for two-week duration, was limited to 4 days because of a problem with fuel cell No. 2, which displayed evidence of internal voltage degradation after launch. Mission was reflown as STS-94.
F	2003 Jan 16	STS 107	2003 003A	A piece of foam falling from the External Tank damaged the thermal protection system on the leading edge of *Columbia*'s left wing. During reentry hot gas penetrated the wing, causing destruction of the orbiter over Texas. Seven astronauts died.

Vehicle Design

Overall Vehicle



	Space Shuttle
Height	38.1 m (184.2 ft)
Gross Liftoff Mass	2040 t (4500 lbm)
Thrust at Liftoff	28.2 MN (6.4 Mlbf)

Stages

Solid Rocket Boosters

Two solid rocket boosters (SRBs) provide 80% of the Space Shuttle's total liftoff thrust. A number of reliability improvements have been made to the SRBs since the *Challenger* accident. As a result, the improved SRBs are also referred to as redesigned solid rocket motors (RSRMs). The expended motors are parachuted back to Earth, retrieved from the Atlantic Ocean approximately 200 km (110 nmi) from the launch site, towed back to Port Canaveral, disassembled, and returned on railcars to ATK Thiokol in Utah. There the booster segments are refurbished, reloaded with propellants, and returned to KSC where they are reassembled for another launch. The steel case components of the RSRMs can be used 20 times.

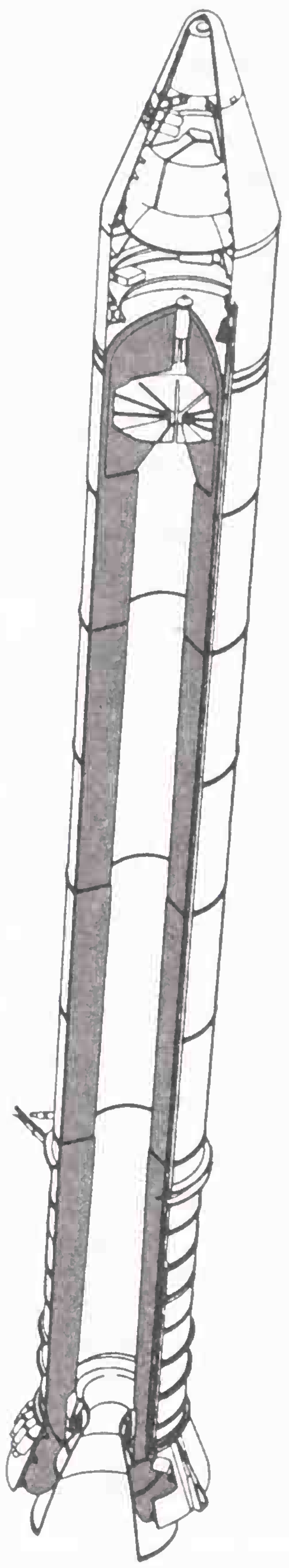
External Tank

The external tank (ET) contains the propellants for the three Space Shuttle main engines (SSMEs) and forms the structural backbone of the system in the launch configuration. At liftoff, the ET absorbs the total thrust loads of the three SSMEs and the two SRBs. When the SRBs separate at an altitude of approximately 46.3 km (25 nmi), the orbiter, with the main engines still burning, carries the ET piggyback to near orbital velocity, approximately 106 km (57 nmi) above the Earth. There, 8.5 min into the mission, the now empty tank separates and falls in a preplanned trajectory into the Pacific Ocean. The ET is the only major element of the Space Shuttle that is expended rather than recovered.

The three main components of the ET are an oxygen tank, located in the forward position; an aft-positioned hydrogen tank; and a collar-like intertank structure, which connects the two propellant tanks, houses instrumentation and processing equipment, and provides the attachment and thrust bearing structure for the forward end of the SRBs. The hydrogen tank is 2.5 times larger than the oxygen tank. The skin of the ET is covered with a thermal protection system that is nominally 25 mm (1 in.) thick consisting of spray-on polyisocyanurate foam. The thermal protection system maintains the propellants at an acceptable temperature during loading, protects the skin surface from aerodynamic heating, and minimizes ice formation on the external surface.

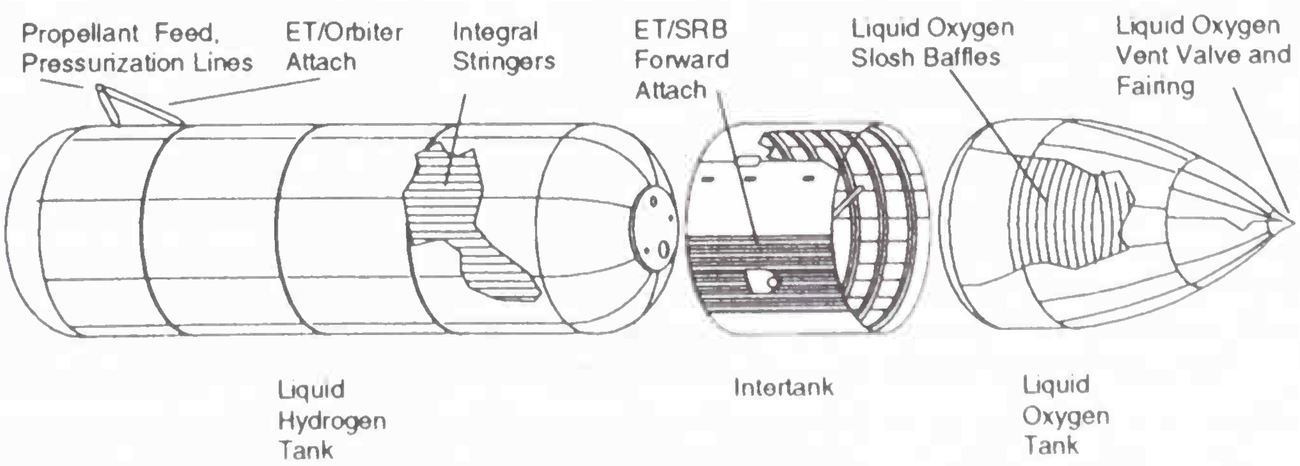
The ET includes a propellant feed system to duct the propellants to the orbiter engines, a pressurization and ventilation system to regulate the tank pressure, an environmental conditioning system to regulate the temperature and render the atmosphere in the intertank area inert, and an electrical system to distribute power and instrumentation signals and provide lightning protection. Most of the fluid control components (except for the ventilation valves) are located in the orbiter to minimize throwaway costs.

The ET has been upgraded from the standard lightweight ET to the new super lightweight external tank (SLWT) configuration. The SLWT is made primarily of 2195 aluminum–lithium alloy, in place of the former 2219 aluminum alloy. The 1% lithium content in the new alloy makes the metal stronger and about 5% lighter. The improved material makes the SLWT about 3500 kg (7700 lbm) lighter than the old tank design. The SLWT was first used on STS-91 in June 1998.



Courtesy Dennis Jenkins.

Redesigned Solid Rocket Motors

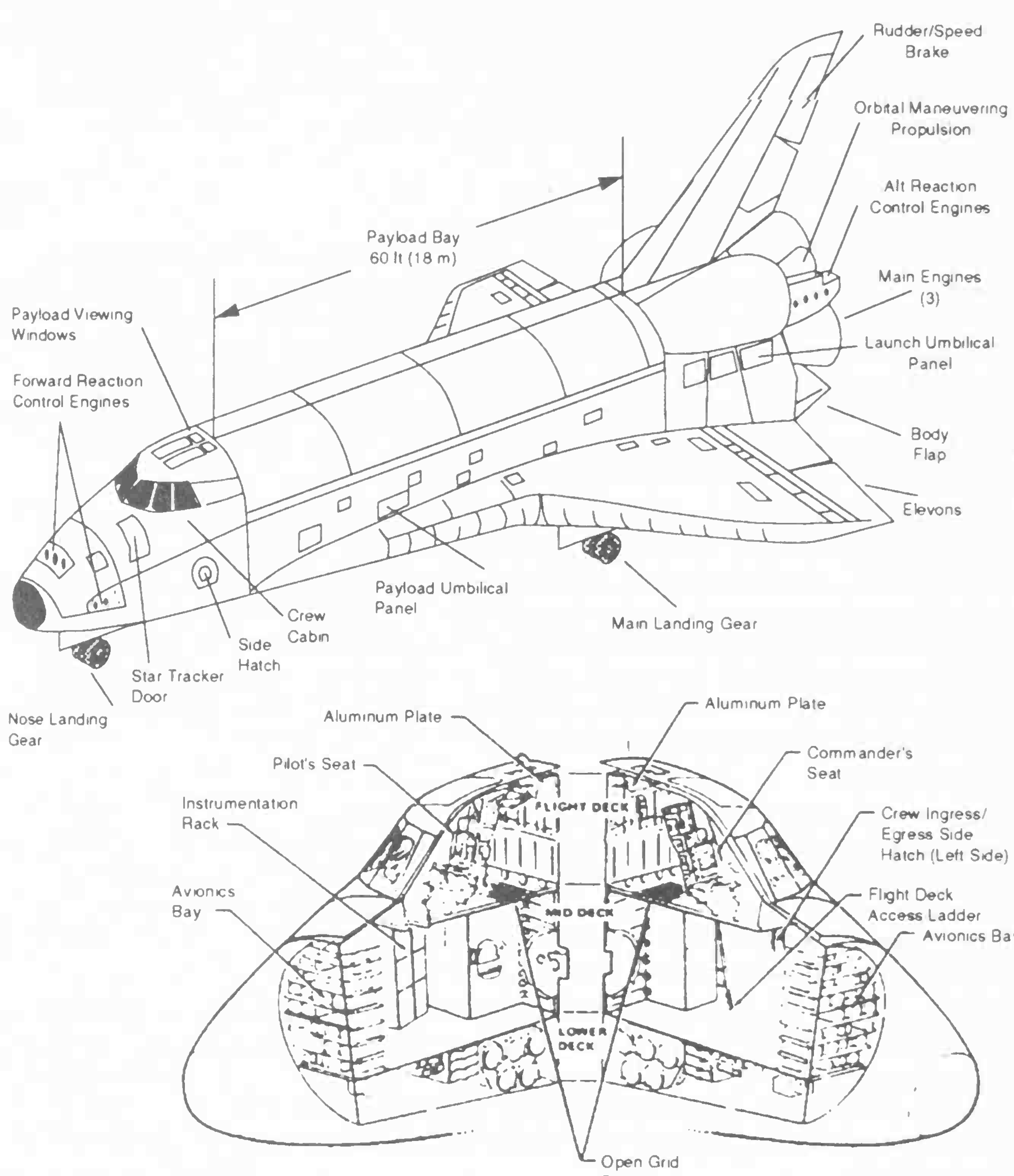


External Tank

Vehicle Design

Orbiter

The Space Shuttle orbiter is designed as a space transport vehicle that can be reused for 100 missions. The crew compartment of the spacecraft accommodates up to seven crew members and can handle 10 people during emergency operations. The orbiter's 18.3×4.5m (60×15 ft) cargo bay can ferry payloads to and from LEO up to 600 km (330 nmi) altitude. It is similar in size and weight to modern transport aircraft. The three SSMEs located in the aft fuselage comprise the main propulsion system. Fuel for the orbiter's main engines is carried in the ET. Both the SRBs and the ET are jettisoned before orbital insertion. The Space Shuttle circularizes its orbit, and maneuvers between different orbits, using the orbital maneuvering system (OMS) contained in two pods on the aft fuselage. The reaction control system (RCS), contained in the two OMS pods and in a module in the nose section of the forward fuselage, provides attitude control in space and during reentry and is used during rendezvous and docking maneuvers. After completing on-orbit operations, the orbiter reenters the Earth's atmosphere and glides to a runway landing. Nominal landing velocity is approximately 330 km/h (180 knots). The orbiter is constructed primarily of aluminum and is protected from reentry heat by the thermal protection system. The principal substructures of the orbiter are the crew module and forward fuselage; mid fuselage and payload bay doors, aft fuselage and engine thrust structures, wings and vertical tail.



Space Shuttle Orbiter

The LH_2/LOX-fueled SSME is a reusable high-performance rocket engine capable of operating at various thrust levels. Ignited on the ground before launch, three SSMEs operate in parallel with the SRBs during the initial ascent. After the boosters separate, the main engines become the sole propulsion element for the remainder of the ascent to orbit. The SSMEs develop thrust by using high-energy propellants in a staged combustion cycle. The propellants are combusted partially in dual prebumers to produce high-pressure hot gas to drive the turbopumps. Combustion is completed in the main combustion chamber. The cycle ensures maximum performance by reducing parasitic losses. The SSME can be throttled over a thrust range of 67–109%, which provides for a high thrust level during liftoff and the initial ascent phase but allows thrust to be reduced to limit dynamic pressure in the Mach 1 region and acceleration to 3 *g* during the final ascent phase. The engines are gimbaled to provide pitch, yaw, and roll control during the orbiter boost phase. The SSMEs have been undergoing a multistep upgrade program. The Block IIA SSME design was first used on all three engines during the STS-89 flight in 1998, after single upgraded engines had been tested on previous flights. The Block IIA SSMEs have a number of design modifications that result in larger margins for improved safety and simplified manufacturing and maintenance. The Block IIA nozzle throat is 11% larger, resulting in a 18.6 bar (270 psi) decrease in chamber pressure, which also reduces temperatures and turbopump shaft speeds. This modification improves the engine reliability and lifetime, but resulted in a slight reduction in specific impulse. To make up for this, the Block IIA operates at a 104.5% power level, a slight increase from the 104% level used previously. The Block IIA engine also includes modified low-pressure turbopumps, improved main injectors, and upgrades for systems like sensors, controllers, and valves. Use of precision castings in manufacturing and various design modifications reduces the number of parts and welds required to assemble the engine. The Block II engines are similar to the Block IIA with the addition of a Pratt & Whitney high pressure fuel turbopump (HPFT) that enables the SSME to reliably operate at 109% power level. The goal for operability is to perform 10 flights without major inspections or overhaul. The first single Block II engine flew on STS 104 in July 2001, and the first flight with three Block II engines was STS 110 in April 2002. The last flight of the Block IIA engines was STS-109 in March 2002.

The orbiter cabin is designed as a combination working and living area. The pressurized crew compartment contains three levels. The flight deck contains the displays and controls used to pilot, monitor, and control the orbiter and the mission payloads. Seating for as many as four crew members can be provided on the flight deck. The mid deck contains passenger seating for three crew members, the living area, an airlock, the galley, sleeping compartments, the toilet, and avionics equipment compartments. An aft hatch in the airlock provides access to the payload bay. The lower deck contains the environmental control equipment and is readily accessible from above through removable floor panels. Located outside the crew module in the payload bay are provisions for a docking module and a transfer tunnel with an adapter to allow crew and equipment transfer during docking and extravehicular operations and for access to Spacelab or Spacehab pressurized modules in the payload bay. The environmental control and life support system (ECLSS) provides a comfortable shirtsleeve habitable environment 18–32° C (61–90° F) for the crew and a conditioned thermal environment (heat-controlled) for the electronic components. The ECLSS bay, which includes air-handling equipment, lithium hydroxide canisters, water circulation pumps, and supply and waste water, is located in the mid deck of the orbiter and contains the pressurization system, the air revitalization system, the active thermal control system, and the water and waste management system.

Two orbital maneuvering engines, located in external pods on each side of the aft fuselage, provide thrust for orbit insertion, orbit transfer, rendezvous, and deorbit. Up to 10,800 kg (24,000 lbm) of usable propellant can be loaded in the two OMS pods. Each pod contains a high-pressure helium storage bottle, the tank pressurization regulators and controls, a fuel tank, an oxidizer tank, and a pressure-fed regeneratively cooled rocket engine. Each engine develops a vacuum thrust of 26.7 kN (6 klbf) using monomethylhydrazine (MMH) and nitrogen tetroxide (N_2O_4). They are burned at a nominal oxidizer-to-fuel ratio of 1.66 and a chamber pressure of 860K Pa (125 psia). The engine is designed for 100 missions with a service life of 10 years and is capable of sustaining 1000 starts and 15 h of cumulative firing time. Each engine is 1.96 m (77 in.) long and weighs 118 kg (260 lbm). The engine is gimbaled by pitch and yaw electromechanical actuators attached to the vehicle structure at the forward end of the combustion chamber. The controller for the actuators is mounted in the pod structure.

VEHICLE DESIGN

	RSRM	External Tank	Orbiter	
Dimensions				
Length	45.46 m (149.16 ft)	47.0 m (154.2 ft)	37.24 m (122.17 ft)	
Wingspan	—	—	23.79 m (78.06 ft)	
Diameter	3.77 m (12.38 ft)	8.4 m (27.6 ft)	17.25 m (56.58 ft) (This is the orbiter height on the runway.)	
Mass				
Propellant Mass	502 t (1106 klbm) each	721 t (1589 klbm)	OMS: 10.9 t (24 klbm) RCS: 3300 kg (7260 lbm)	
Inert Mass	88 t (194 klbm) each	27 t (59.5 klbm	*Discovery:* 78.7 t (173.5 klbm) *Atlantis:* 78.4 t (172.8 klbm) *Endeavour:* 78.8 t (173.7 klbm)	
Gross Mass	590 t (1300 klbm) each	748 t (1648 klbm)	Approximately 94 t (207 klbm)	
Propellant Mass Fraction	0.85	0.96	0.15	
Structure				
Type	Monocoque	Skin Stringer	Semimonocoque	
Material	Steel	Aluminum and aluminum–lithium	Aluminum	
Propulsion				
Engine Designation	RSRM	No propulsion (see orbiter main engines)	**SSME**	**OMS**
Number of Engines	2 (4 segments each)	—	3	2
Propellant	PBAN	—	LOX/LH_2 (from external tank)	MMH/N_2O_4
Average Thrust (each)	Sea level: 11.79 MN (2.65 Mlbf)	—	At 109% power Sea level: 1.862 MN (418,660 klbf) Vacuum: 2.218 MN (512,950 klbf)	26.7 kN (6 klbf)
Isp	Vacuum: 267.3 s	—	Vacuum: 452 s	Vacuum: 316 s
Chamber Pressure	63.3 bar (918 psi)	—	207 bar (3008 psi)	8.6 bar (125 psi)
Nozzle Expansion Ratio	7.5:1	—	69:1	55:1
Propellant Feed System	—	—	Staged combustion turbopump	Pressure fed
Mixture Ratio (O/F)	—	—	6.03:1	1.65:1
Throttling Capability	—	—	67–109%	100% only
Restart Capability	—	—	No	1000 start lifetime over multiple missions
Tank Pressurization	—		Hot-gas recirculation	Helium
Attitude Control				
Pitch, Yaw	Pumped hydraulic nozzle gimbal ± 8 deg with integral flexible bearing	—	Hydraulic nozzle gimbal	Electromechanical nozzle gimbal
Roll	Pumped hydraulic nozzle gimbal ± 8 deg with integral flexible bearing	—	Hydraulic nozzle gimbal ± 10.5 deg, ± 8 deg yaw	RCS ± 7 deg
Staging				
Nominal Burn Time	123 s	—	522 s (includes 6 s preliftoff)	
Shutdown Process	Burn to depletion	—	Commanded shutdown	
Stage Separation	4 separation rockets fore and aft	Pyrotechnic	RCS thrusters	
Return				
Final deceleration	Parachutes	Not recovered	Wings	
Landing	Vertical water landing	—	Horizontal runway landing	
Cross range	None	—	2034 km (1100 nmi)	

Vehicle Design

Attitude Control System

During launch, control is established through gimbaling of the SRB and SSME nozzles. Both systems use hydraulic actuators. Once in orbit, the orbiter RCS provides the thrust for velocity changes along the axis of the orbiter and attitude control (pitch, yaw, and roll) during the orbit insertion, on-orbit, and reentry phases of flight. It has 38 bipropellant primary thrusters and 8 vernier thrusters. The primary thrusters are used for normal translations and attitude control. The vernier thrusters are used for the fine attitude control and payload pointing where contamination or plume impingement are important considerations. Each primary thruster provides 3870 N (870 lbf) thrust. The vernier thrusters, which have no redundancy, are oriented to vector plumes away from the payload bay. Each vernier thruster provides 110 N (25 lbf) thrust.

The RCS is grouped in three modules, one in the orbiter nose and one in each aft fuselage pod. Each module is independent and contains its own pressurization system and propellant tanks. The forward module contains 14 primary thrusters and 2 vernier thrusters. The multiple primary thrusters pointing in each direction provide redundancy for mission safety.

RCS propellants are MMH and N_2O_4. Total RSC propellant weight is 3300 kg (7250 lbm). The design mixture ratio of 1.65 (oxidizer weight to fuel weight) permits the use of identical propellant tanks for both fuel and oxidizer. A system of heaters is used to maintain the temperatures of the engines, propellant lines, and other components within operational limits.

An interconnection between the OMS and RCS in the aft pods permits the use of OMS propellant by the RCS for orbital maneuvers. In addition, the interconnection can be used for crossfeeding OMS and RCS propellants between the right and left OMS. In the event of an abort during ascent, OMS propellant can be dumped via the RCS thrusters as well as the OMS engine to meet orbiter landing weight constraints.

Avionics

The Space Shuttle avionics system control functions include guidance, navigation, control, and electrical power distribution for the orbiter, the ET, and the SRBs. In addition, the avionics control the communications equipment and can control payloads. Orbiter avionics automatically determine vehicle status and operation readiness and provide sequencing and control for the ET and the SRBs during launch and ascent. Automatic vehicle flight control can be used for every mission phase except docking. Manual control is also available at all times as a crew option.

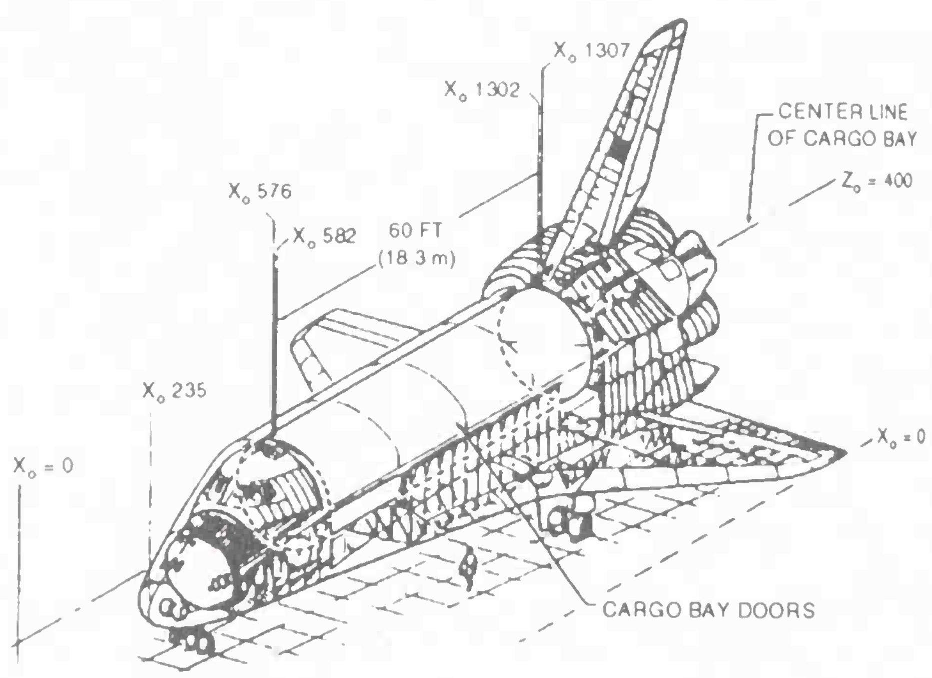
The avionics are designed with redundant hardware and software to withstand multiple failures. The Space Shuttle avionics system consists of more then 200 electronic "black boxes" connected to a set of five computers through common data buses. The electronic black boxes offer dual or triple redundancy for every function.

The avionics system is closely interrelated with three other systems of the orbiter—the guidance, navigation, and control system; the controls and displays system; and the communications and data systems.

The orbiter has an electrical power and distribution system and a hydraulic power system. Electrical power is generated by three fuel cells that use cryogenically stored hydrogen and oxygen reactants. Hydraulic power is derived from three independent hydraulic pumps, each driven by its own hydrazine-fueled auxiliary power unit and cooled by its own ammonia spray boiler.

Payload Bay

The payload bay is enclosed by two doors that open to expose the entire length and width of the cargo bay. The usable volume of the cargo bay measures 4.6×18.3 m (15.0×60.0 ft). This volume is the maximum allowable payload dynamic envelope, including payload deflections. In addition, a nominal 76 mm (3 in.) clearance between the payload envelope and the orbiter structure is required to prevent orbiter deflection interference between the orbiter and the payload envelope. Payloads can be mounted in the bay in several ways. Fixed payloads are generally mounted using attachment points at the bottom and sides of the payload bay. Deployable payloads are generally mounted on mission-specific deployment structures.



Length	18.6 m (60.8 ft)
Primary Diameter	4.7 m (15.5 ft)
Sections	Two nonseparating payload bay doors
Structure	Skin-stringer mid fuselage
Material	Aluminum

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	4570 mm (180.0 in.)
Maximum Cylinder Length	18,300 mm (720.0 in.)
Maximum Cone Length	—
Payload Adapter-Interface Diameter	Mission Unique
Payload Integration	
Nominal Mission Schedule Begins	T−36 to 48 months for primary payloads
Launch Window	
Latest Countdown Hold Not Requiring Recycling	T−31 s
On-Pad Storage Capability	8 h fueled
Latest Access to Payload	T−48 to 17 h
Environment	
Maximum Load Factors	+ 3.2 *g* axial, ±2.5 *g* lateral, 4.2 *g* landing
Minimum Lateral/Longitudinal Payload Frequency	15 Hz/35Hz
Maximum Overall Acoustic Level	140 dB (one third octave)
Maximum Flight Shock	5500 *g* at 4000 Hz
Maximum Dynamic Pressure on Vehicle	39.2 kPa (819 psf)
Maximum Pressure Change in Fairing	3.45 kPa/s (0.5 psi/s)
Cleanliness Level in Fairing (before launch)	Class 10,000 +
Payload Delivery	
Standard Orbit and Accuracy (3 sigma)	Circular orbit: ±10 nmi (18 km), ±0.5 deg inclination
Altitude Accuracy (3 sigma)	Primary RCS thrusters: ±0.1 deg, ±0.2 deg/s all axes Vernier RCS thrusters: ±0.1deg, ± 0.01 deg/s all axes
Nominal Payload Separation Rate	0.3 m/s (1 ft/s)
Deployment Rotation Rate Available	0 rpm (spin tables or special carrier required)
Loiter Duration in Orbit	7 days or 16 days with EDO
Maneuvers (Thermal/Collision/Telemetry)	Yes
Multiple or Comanifest	Space Shuttle missions routinely carry a large number of discrete payloads both inside the payload bay and in the crew cabin mid deck. A variety of standardized cargo carriers and interfaces are available. Contact JSC for more information.
Auxiliary Payloads	Small payloads can be carried inside the crew cabin in mid deck lockers. Each locker can hold 25 kg (54 lbm) in a 0.056 m3 (2 ft3) volume measuring 516 x 439 x 252 mm (20.32 x 17.31 x 9.95 in). Auxiliary payloads can be carried in the payload bay using several standard interfaces. One option is the Get Away Special (GAS) program. GAS canisters are 0.14 m3 (5 ft3) cylindrical structures that can hold up to 90 kg (200 lbm).

Production and Launch Operations

Launch Operations—Kennedy Space Center

Space Shuttle launch operations require the use of a wide variety of both advanced and routine technologies and interrelated subsystems. At Kennedy Space Center (KSC), Space Shuttle processing is carried out by contractors monitored by NASA employees. United Space Alliance (USA), a joint-venture company owned by Boeing and Lockheed Martin, currently holds the prime contract for this work. USA subcontracts with a variety of other companies, including Boeing as the prime contractor for the orbiter. The NASA Johnson Space Center (JSC) has responsibility for on-orbit mission operations and some launch operations. The NASA Marshall Space Flight Center (MSFC) contributes engineering expertise for Space Shuttle modifications.

Payload Processing—Payloads for the Space Shuttle can be installed either horizontally or vertically. Horizontal payloads such as Spacelab are installed in the orbiter when it is still in the orbiter processing facility (OPF), before being mated with the ET and SRBs. Vertical payloads are installed in the orbiter's payload bay after the fully assembled Space Shuttle arrives at the launch pad.

Payload owners have five options for processing payloads ranging from minimum KSC involvement, essentially "ship and shoot," up to maximum KSC involvement. Payloads ready for launch upon arrival at KSC can be installed in the orbiter 2–50 days before launch with no servicing. Payloads requiring KSC participation for assembly must have all flight experiments and component hardware delivered to KSC up to a year before the expected launch date to allow for assembly, integration, and testing.

Prelaunch Processing—The basic processing flow is organized according to the integrate–transfer–launch (ITL) concept, which separates the major processing elements and allows certain functions to proceed in parallel until the vehicle is assembled in the Vehicle Assembly Building (VAB) and transported to the launch pad.

Orbiter Processing—Orbiter processing constitutes the critical limit to the achievable flight rate. Refurbishing the orbiter *Columbia* for the second Shuttle flight consumed nearly 200 working days at three shifts per day. By the time of the ill-fated flight of *Challenger* in January 1986, this turnaround time had been reduced to 56 days. Numerous modifications to the orbiter refurbishment process, resulting from the *Challenger* accident, have resulted in a nominal turn around time of 75 days.

In the OPF, NASA contractors check and refurbish every major system in the orbiter after each flight. They remove the main engines and complete refurbishment of the SSME in the offline engine shop. Any of the 31,000 ceramic tiles of the thermal protection system that are lost or damaged during the flight are also replaced. Modifications of the orbiters are made during refurbishment. Finally, any horizontal payloads, such as Spacelab, are installed in the payload bay. The OPF resembles a modern aircraft maintenance hangar. The OPF is located west of the VAB. It can handle two orbiters at a time.

The OPF consists of two identical high bays connected by a low bay. Each high bay is 60 m (197 ft) long, 45 m (150 ft) wide, and 29 m (95 ft) high. Each bay has two 27 t (30 U.S. ton) bridge cranes and contains a complex series of platforms that surround the orbiter and permit work access. The high bays also have underfloor trench systems that contain electrical, electronic, and communications instrumentation as well as outlets for gaseous nitrogen, oxygen, and helium. In addition, the high bay areas have emergency exhaust systems that are used in the event of a fuel spill in the area. Fire protection systems are located throughout the facility. The low bay is 71 m (233 ft) long, 29 m (95 ft) wide, and 7.6 m (25 ft) high. In addition to an office annex, it also contains electronic, mechanical, and electrical support systems.

SRB Processing—Thiokol, the contractor, ships new and refurbished SRM segments and associated hardware, including the forward and aft closures, nozzle assemblies, and nozzle extensions by rail. When the segments arrive at KSC, they are moved into the Rotational Processing and Surge Facility where they are inspected and stored until needed. The SRBs are stacked in the VAB on top of the mobile launch platform (MLP). Stacking operations take approximately 19 days.

ET Processing—The ET, which is manufactured by Lockheed Martin at its Michoud Assembly Facility outside New Orleans, Louisiana, is transported to KSC by sea barge. In the VAB, contractors inspect the external insulation and interfaces for ground support equipment connection. The electrical systems are checked, and the fluid systems are tested. A crane hoists the tank to a vertical position and transfers it to the MLP, where it is mated with the twin SRBs.

Vehicle Assembly and Integration—After orbiter processing is complete, the orbiter is towed to the VAB high bay, erected, and mated to the ET and SRBs. After mating all the sections of the Space Shuttle and connecting all umbilicals, engineers test each connection electrically and mechanically. The computer-controlled launch processing system, which is operating from the firing rooms of the launch control center, semiautomatically controls and checks out much of the vehicle, both in the VAB and at Launch Complex 39. If any subsystem is found to be unsatisfactory, the computer will provide data that will help isolate the fault.

Transfer and Launch—When the Space Shuttle is fully assembled on the MLP, a crawler-transporter is positioned under the MLP and slowly [(1.6 km/h (1 mi/h)] moves the Space Shuttle to Launch Complex 39A or B. Once at the pad, workers gain access to the vehicle through the fixed service structure. The rotating service structure (RSS) gives access to service fuel cells, to load and remove payloads, and to load hypergolic fuels for the orbital maneuvering system and the RCS. Those payloads to be installed vertically are transported to the rotating service structure in a protective payload canister.

After the Space Shuttle arrives at the pad, most checkout operations are controlled from the launch control center. After checkout operations are completed, power is applied to the orbiter and ground support equipment. Launch-readiness tests are performed, and the tanks are prepared to receive their fuels. The Space Shuttle is now ready for the cryogenic propellants to be loaded and the flight crew to board.

During the final 6 or 7 h of the countdown and the LH_2 and LOX are loaded into the ET. Finally, the flight crew and operations personnel complete all preparations and the Shuttle lifts off.

Launch Complex Facilities—KSC Launch Complex 39 (LC-39), has two identical launch pads that, like many Space Shuttle facilities, were originally designed and built for the Apollo program. The pads, built in the 1960s, were used for all of the Apollo/Saturn V missions and the Skylab space station program. Between 1967 and 1975, 12 Saturn V/Apollo vehicles, one Saturn V/Skylab workshop, three Saturn 1B/Apollo vehicles for Skylab crews, and one Apollo/Saturn 1B for the joint U.S.–USSR Apollo–Soyuz Test Project, were launched from these pads. Each of the dual launch pads, designated Pads A and B, covers an area of about one-quarter of a square mile. Located not far from the Atlantic Ocean, Pad A is 14 m (48 ft) above sea level, while Pad B is 17 m (55 ft) above sea level. The pad areas are octagonal in shape.

To accommodate the Space Shuttle vehicle, major modifications to the pads were necessary. Initially, Pad A modifications were completed in mid 1978, while Pad B was finished in 1985 and first used for the ill-fated STS-51L, *Challenger*, mission in January 1986. Major pad modifications included construction of new hypergolic fuel and oxidizer support areas at the southwest and southeast corners of the pads; construction of new Fixed Service Structures (FSSs); addition of a RSS; addition of 1100 m^3 (300,000 gal) water towers and associated plumbing; and, finally, replacement of the original flame deflectors with Space Shuttle-compatible deflectors.

Production and Launch Operations

Following the flight schedule delays resulting from the *Challenger* accident, an additional 105 pad modifications were made. Among them were installation of a sophisticated laser parking system on the MPL to facilitate mounting the Space Shuttle on the pad and emergency escape system modifications to provide emergency egress for up to 21 people. The emergency shelter bunker also was modified to allow easier access from the slidewire baskets.

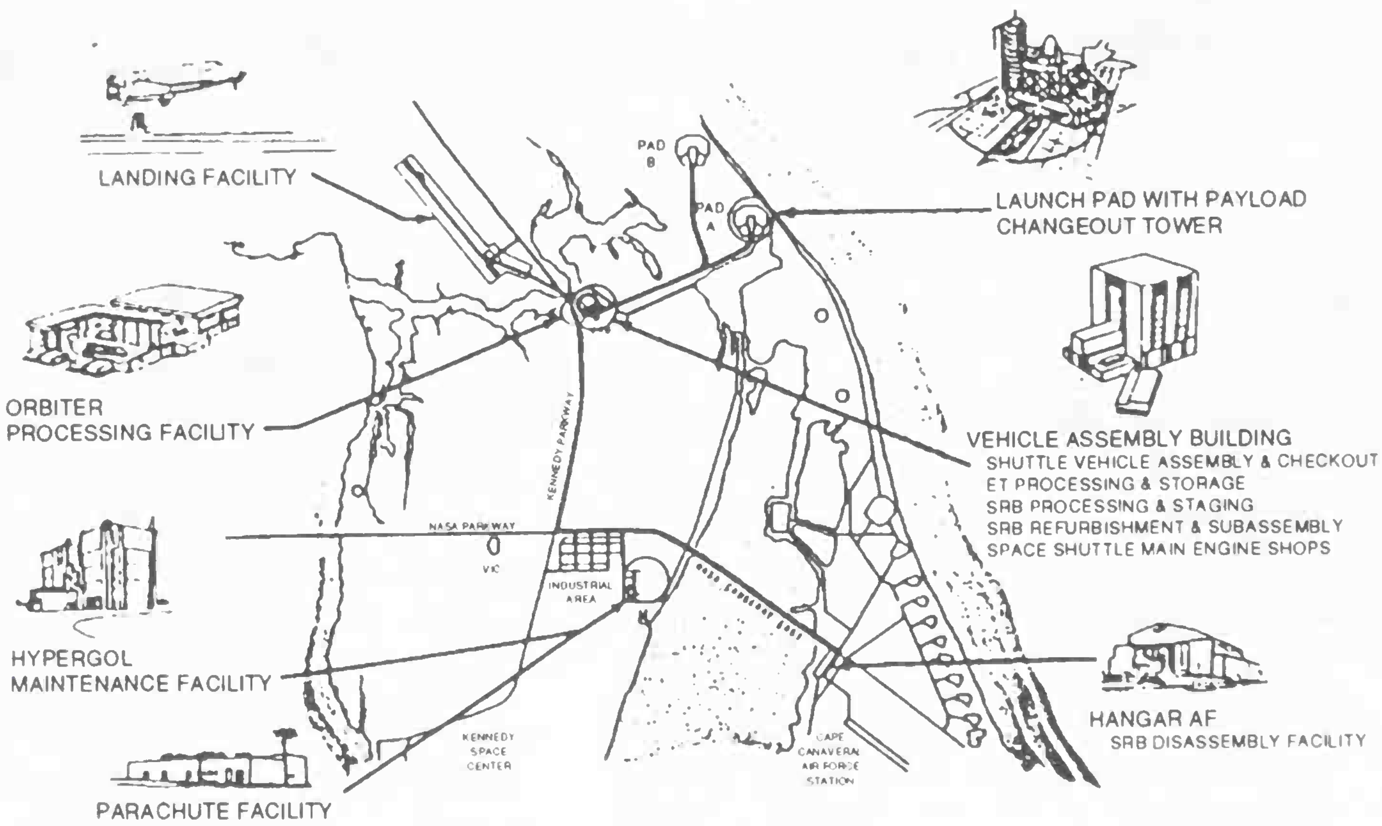
Mission Operations—Mission operations comprise all activities associated with planning and executing a mission. The primary focus is on gathering data, performing analyses, and developing the software required to meet the mission's objectives. Mission operations begin the day a payload is conceived, continue through the day of the launch, and end only after the mission is satisfactorily completed and the postmission data are analyzed. From beginning to end, mission operations for a flight may take two years or more. JSC is the central control point for Space Shuttle missions.

As might be expected, crew training and planning for a particular Space Shuttle mission are intertwined closely. The key elements of mission planning outline specific crew activities and essential flight support functions. The effort, like astronaut training, is directed by the Mission Operations Directorate (MOD) at JSC. The degree of thoroughness of this planning probably can best be described as mind boggling. Because crew activity planning is the analysis and development of when and what activities are to be performed on a specific mission, the end result is a minute-by-minute timeline of each crewmember's activities.

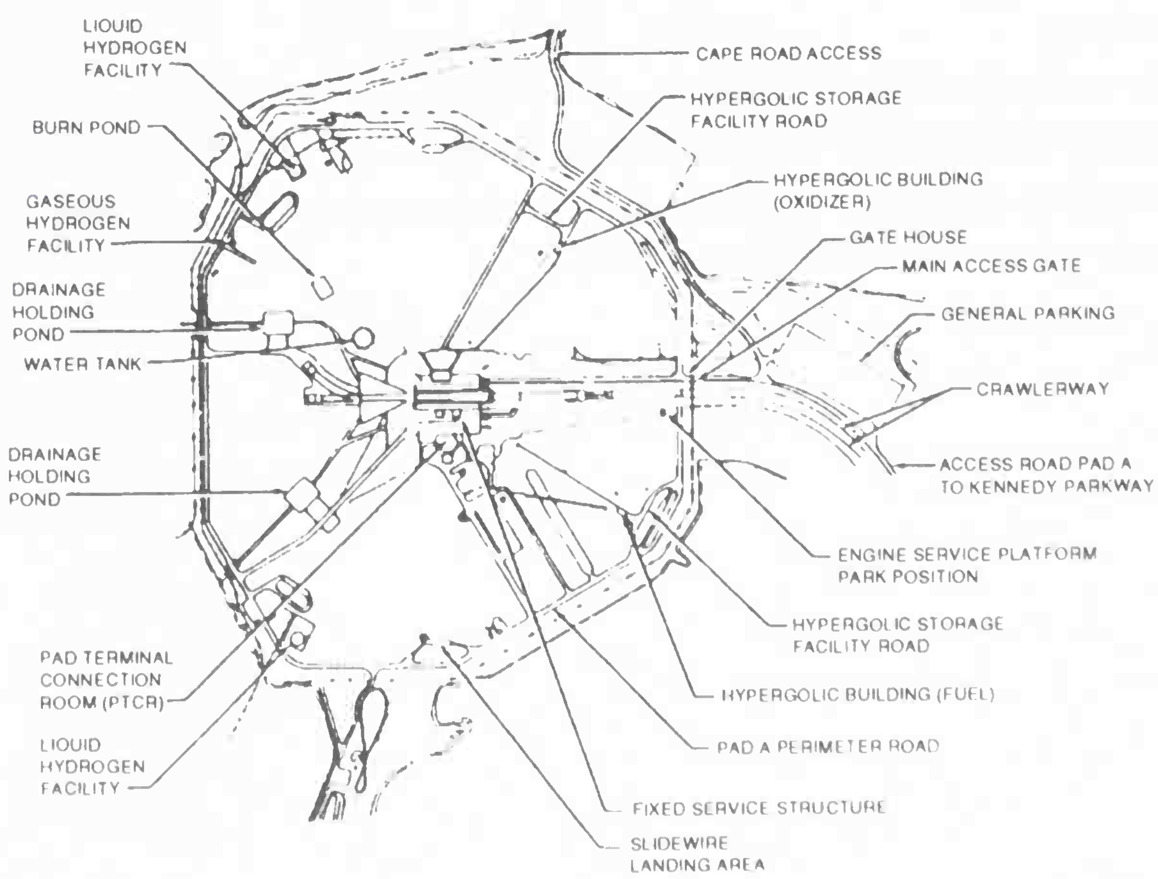
Postlaunch Processing—Two minutes into the Space Shuttle's flight the two SRBs are jettisoned and parachute into the Atlantic Ocean downrange from KSC. Two specially designed retrieval vessels recover the boosters and their components. The smaller components are hauled onboard the ships and the boosters are towed back to the KSC SRB Disassembly Facility. At this facility, the boosters and other components are washed, disassembled, cleaned, and stripped before they are shipped by rail to Thiokol for refurbishment.

Nominal completion of a Space Shuttle mission calls for a landing at the KSC Shuttle Landing Facility. When required, the orbiter can land at backup sites at Edwards AFB, California, or White Sands, New Mexico. These landing facilities, along with others located in Zaragoza, Spain; Rota, Spain; Morocco; and Guam also serve as emergency landing facilities for aborted launches. After landing, the orbiter must be drained of hazardous fuels and inspected for any exterior damage. Payload technicians remove any payloads brought back to Earth. If it lands anywhere other than KSC, the orbiter must be lifted onto the back of a specially equipped Boeing 747 and ferried back to KSC.

Launch Facilities



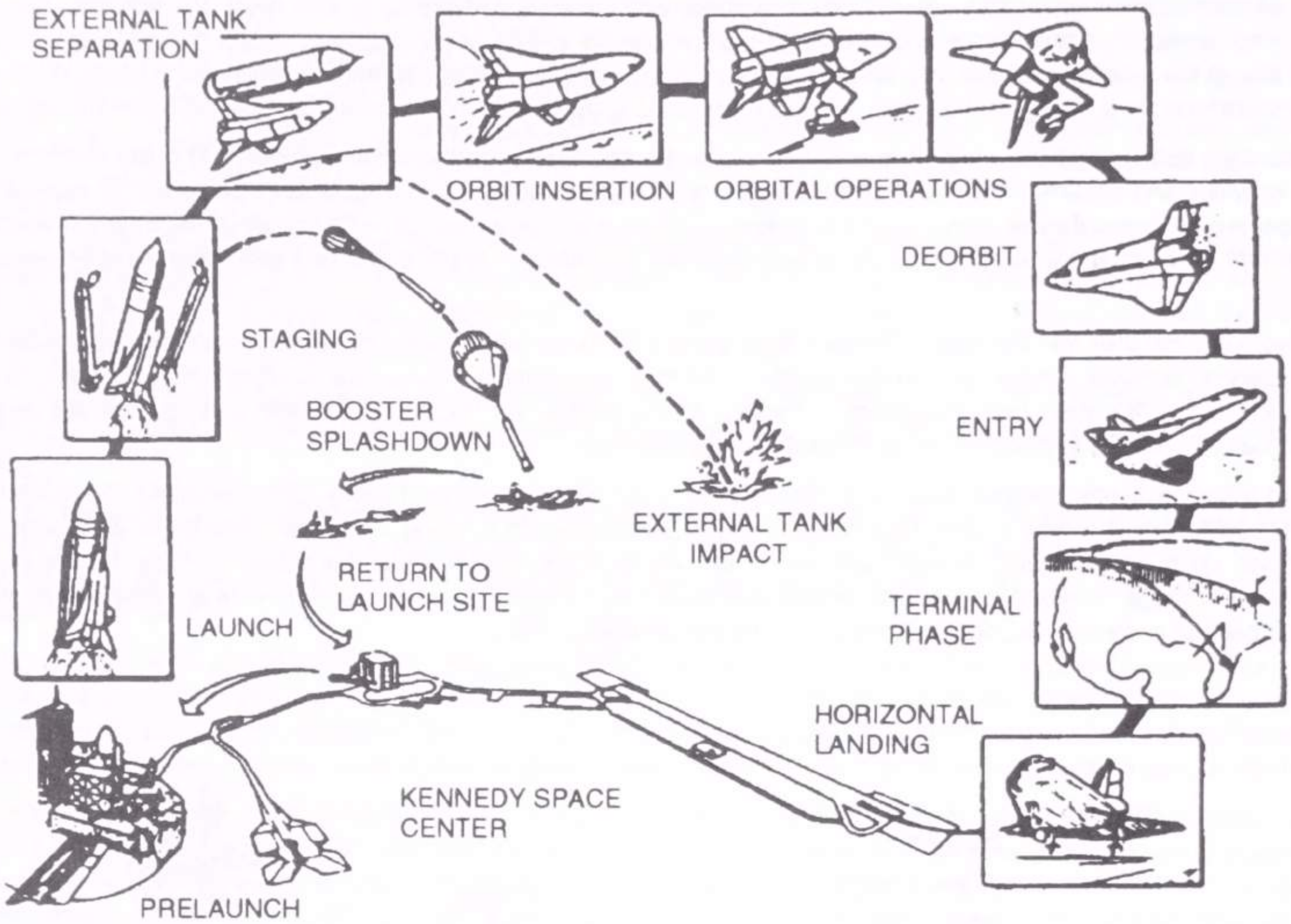
Kennedy Space Center



LC-39A Layout

Production and Launch Operations

Flight Sequence



Event	Time, s	Geodetic Altitude km (mi)[a]		Inertial Velocity km/h	(mph)	Inertial Velocity km/h	(mph)
SSME ignition	−6.0	56[b]	(184)	1471[c]	(914)	0	(0)
SRB ignition	0.0	56[b]	(184)	1471[c]	(914)	0	(0)
Begin pitchover	7.0	166[b]	(545)	1476	(917)	0	(0)
Maximum dynamic pressure	60.0	13.4	(8.3)	2662	(1654)	6.4	(4)
SRB separation	124.0	47.3	(29.4)	5533	(3438)	38.1	(23.7)
Main-engine cutoff	512.0	117.5	(73)	28,163	(17,500)	1335	(829.3)
External Tank separation	530.0	118.3	(73.5)	28,160	(17,498)	1427	(886.6)
OMS-2 ignition	2638.0	279.4	(173.6)	27,682	(17,201)	15,731	(9775)
OMS-2 cutoff	2734.0	280.3	(174.2)	27,875	(17,321)	16,526	(10,269)

[a]Altitude reference point is the orbiter center-of-gravity above the geodetic representation of the Earth's surface.

[b]in meters (feet)

[c]Rotational velocity of Earth at KSC's latitude (28.5° N).

Vehicle Upgrade Plans

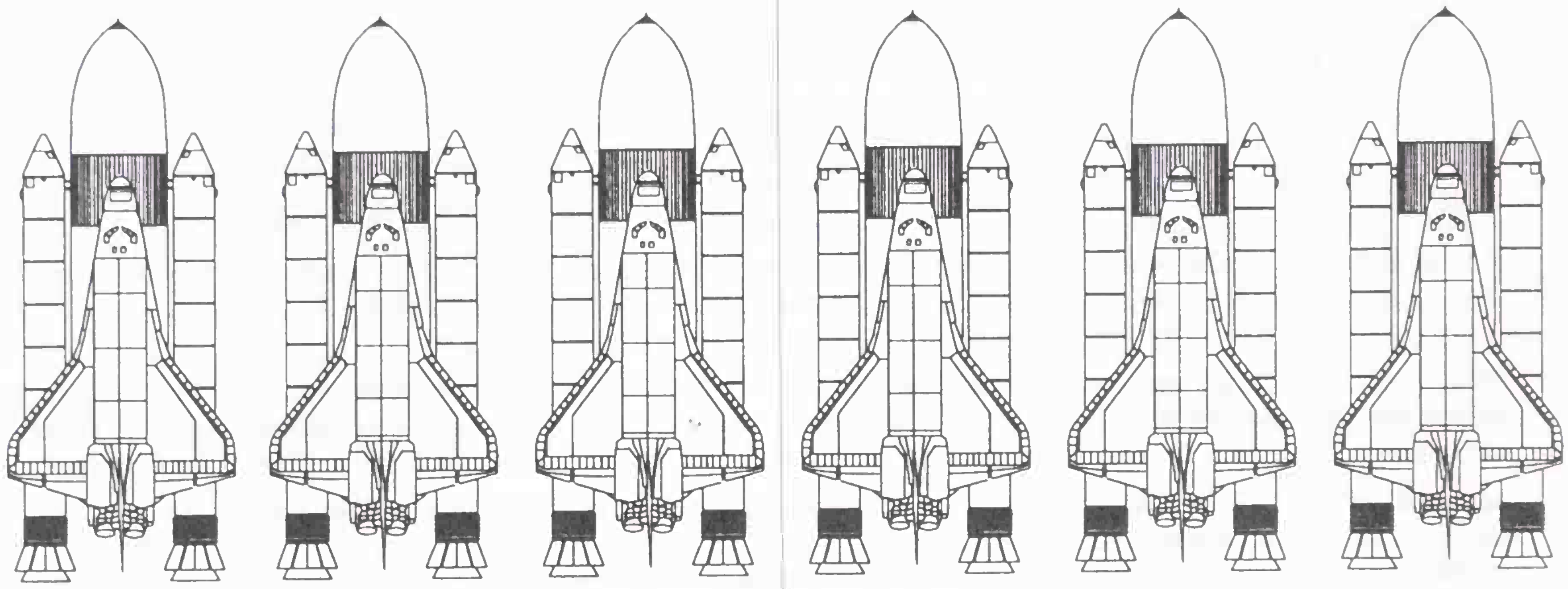
One upgrade being implemented in the fleet is the multifunction electronic display subsystem (MEDS). This upgrade puts a modern "glass cockpit," consisting of 11 multifunction flat panel color displays, in the orbiter cockpit, replacing old gauges with systems similar to modern commercial aircraft. The MEDS upgrade has been performed on *Atlantis*, and will be phased into the rest of the fleet during standard maintenance overhauls through 2002.

NASA has considered a number of other vehicle upgrades for the Space Shuttle program. Near-term upgrades will focus on minor incremental modifications of existing systems to improve safety and reduce costs. More complex and expensive upgrades could include complete replacement of some subsystems, without changing the orbiter mold line. For example, the OMS engines could be replaced by a propulsion system that uses nontoxic propellants such as LOX/ethanol. Some hydraulic actuators could be replaced by electromechanical systems. Fuel cells and auxiliary power units might be replaced with improved systems. The most expensive potential upgrades include those that affect the interfaces and mold lines of the Space Shuttle stack. For example, NASA has proposed replacing the SRBs with new reusable liquid boosters that would fly back to the launch site after separation from the orbiter using jet engines and wings.

Vehicle History

Vehicle Evolution

Out of Service | In Service



Vehicle	***Enterprise***	***Challenger***	***Columbia***	***Discovery***	***Atlantis***	***Endeavour***
Period of Service	*(none)*	*1983–1986*	*1981–2003*	*1984–Present*	*1985–Present*	*1992–Present*

Vehicle Description

•***Enterprise***	OV-101	Test orbiter used only for atmospheric and landing tests and fit verification and practice tests at launch sites.
•***Challenger***	OV-099	Originally built as structural test article, it became second operational orbiter. It was destroyed in flight in 1986.
•***Columbia***	OV-102	The first operational orbiter, heavier than later orbiters, but had extended duration orbiter (EDO) capability for 16-day missions. It was destroyed on descent in 2003.
•***Discovery***	OV-103	The third operational orbiter.
•***Atlantis***	OV-104	The fourth operational orbiter.
•***Endeavour***	OV-105	The replacement for *Challenger*. It also has EDO capability for 16-day missions.

VEHICLE HISTORY

Historical Summary

In the early 1960s, virtually all U.S. aerospace companies conducted studies of recoverable space boosters. The experiments, studies, and developmental efforts (such as Dynasoar and the X-series of rocket-powered aircraft) conducted during this time explored the technological possibilities for an aerospace vehicle that would combine the performance of the expendable vehicles with reusability. These efforts supplied the background for studies in the late 1960s on reusable space transportation vehicle configurations.

In January 1969 NASA initiated four six-month Phase A feasibility study contracts on reusable vehicle concepts, called at that time integrated launch and reentry vehicles (ILRVs). General Dynamics (now part of Lockheed Martin), Lockheed (now part of Lockheed Martin), McDonnell Douglas (now part of Boeing), and North American Rockwell (now part of Boeing) each received one of the $600,000 contracts. Though NASA had originally favored a stage-and-a-half design with external propellant tanks, the Manned Spacecraft Center (now Johnson Space Center) began a study of two-stage fully reusable concepts.

NASA established a Space Shuttle Task Group, which reported to NASA in May 1969. This report stated that the Space Shuttle system development should not be considered in terms of probability, but rather as essential for future space operations. The preferred concept was a fully or near-fully reusable system capable of controlled runway landings.

In March 1970, President Richard Nixon redefined U.S. space goals. The president made it clear that while the space program should not be allowed to stagnate, it had to be balanced with the critical problems facing the United States here on Earth. Proposed post-Apollo lunar missions and manned expeditions to Mars could not be mounted at that time. The administration recommended a NASA budget for fiscal year 1971 that was less than the budget for fiscal year 1970.

NASA revised its near-term goals. The post-Apollo civilian space program would now include only the space station and the Space Shuttle. NASA planned for tandem development of both programs. The dual program was to cost about $5 billion each for the space station and the Space Shuttle. When it became clear that concurrent development would not be funded, NASA was forced to choose between the two programs. Because it was unreasonable to build the space station without a low-cost supply system, NASA's only logical choice was to develop the Space Shuttle as the major program for the 1970s and to postpone the space station until after Space Shuttle expenditures peaked.

The Space Shuttle, as originally envisioned, was to be a fully reusable two-stage vehicle. The booster stage was to be the size of a Boeing 747 and the orbital stage about the size of a Boeing 707. Both stages were to be rocket-powered burning hydrogen and oxygen carried in internal fuel tanks. The two stages would be attached in parallel for vertical takeoff. After launch, the booster would fly back to the launch site for horizontal landing and would be refurbished for the next flight. The orbital stage would proceed to orbit and, upon completing its mission, return to Earth and land horizontally.

In July 1970, NASA awarded Phase B detailed design contracts for the Space Shuttle to North American Rockwell and McDonnell Douglas. The Space Shuttle was to be capable of placing a payload of 11,340 kg (25,000 lbm) in a 445-km (240-nmi) 55-deg inclination orbit. Cross-range capability was to be 370–2780 km (200–1500 nmi). Both straight-wing and delta-wing designs were studied.

In a parallel effort, NASA conducted an in-house review of the Space Shuttle program as well as contracting for three extended Phase A feasibility studies on alternate concepts. These contracts went to Chrysler, Lockheed, and a Grumman–Boeing partnership. Although these studies pursued concepts that did not prove viable, they did impact the ultimate design of the orbiter by influencing design concepts and philosophies.

In January 1971, NASA changed the requirements for the Space Shuttle. NASA now required the capability to put 29,500 kg (65,000 lbm) of payload into a 185 km (100 nmi) due-east orbit, 18,150 kg (40,000 lbm) into a 55-deg orbit, and 11,340 kg (25,000 lbm) into a 513-km (277-nmi) polar orbit. The projected development cost of this configuration was estimated to be approximately $9.9 billion.

Economic factors were important in reaching the decision on the final configuration. As the Space Shuttle studies progressed, it became clear that the development would be more expensive than originally projected. The U.S. Office of Management and Budget (OMB) asked NASA to do an in-house cost–benefit analysis of the Space Shuttle and also to contract out a more detailed independent study.

NASA's internal cost–benefit study done in response to the OMB request showed a distinct advantage for the Space Shuttle over existing, expendable rocket systems, possible future low-cost expendable launchers, and a hybrid system of a reusable orbiter and expendable booster. But the study also estimated development costs ranging from $6.4 to $9.8 billion (FY 71 dollars).

Mathematica, Inc., of Princeton, New Jersey, estimated the development costs at $12.8 billion for the two-stage reusable Space Shuttle. In its final report, in which Mathematica considered a number of different possibilities for a space transportation system as well as different configurations, Mathematica concluded "that the development of a thrust-assisted-orbiter Shuttle system (TAOS) was justified, within a level of space activities between 300 and 360 flights in the 1979–1990 period...."

Political consideration also affected the development of the Space Shuttle. Development of the fully reusable Space Shuttle would have been technically difficult, which meant that it would have been expensive to build, and it would have been opposed by many members of Congress, and it would be questioned by the administration because of the uncertainty of the required technology for the crewed booster.

Opposition to the Space Shuttle in the Senate was reinforced by the opposition to the project by some of the scientific community. Some scientists believed that the anticipated mission model and payload savings projections would not materialize and that launch costs would be significantly higher than anticipated. Others believed that there would be extensive cost overruns and that a civilian space program dominated by crewed missions would not be cost effective. Some critics attacked NASA's changing justification for the Space Shuttle from a vehicle necessary to provide logistical support for large-scale crewed space efforts to a vehicle that would revolutionize space activity as a utilitarian cost-effective transportation system for civilian space science and applications and for national defense missions. Still others fought the Space Shuttle on the basis that the DoD, as the projected largest single user, should help finance the development.

The opposition to the Space Shuttle peaked in 1970 when an amendment to delete funds for the project was defeated in the Senate by only four votes. Despite intense debate and continuing opposition in the Senate, the Congress voted repeatedly for funding for the Space Shuttle program, although perhaps not at the level viewed as necessary by some program proponents.

The administration established a December 1971 deadline for final endorsement or cancellation of the Space Shuttle program. The administration, NASA, and congressional proponents of the Space Shuttle felt compelling pressure to make a decision by the publication of the fiscal 1973 budget in January 1972. They feared that opponents of the program would be able to defeat it in Congress, if it were deferred any longer. Also, further deferment of the Space Shuttle might have made it impossible to hold together the industrial teams that had been brought together.

Vehicle History

Alternate booster concepts were studied to provide a less-expensive design for the Space Shuttle. NASA officially decided to scrap the fully reusable design that it had worked on for 18 months and to find a new more cost-effective concept. NASA ultimately chose to pursue a TAOS. The new configuration involved less technological risk. The TAOS consisted of a crewed orbiter, an expendable external propellant tank, and two recoverable solid-fueled rockets. The only nonreusable part would be the ET.

On 5 January 1972, President Nixon endorsed the Space Shuttle program based on the TAOS concept and requested the development of the space transportation system to begin at once. The Space Shuttle, he noted, would enable the United States to achieve a working presence in space by making space transportation routinely available and by reducing costs and preparation time.

Having received the go-ahead, NASA was ready to begin acquisition and development of the Space Shuttle elements. Acquisition for the Space Shuttle was competitive. Separate contractors were selected for the design and manufacture of the orbiter, its main engines, the ET, and the SRBs. Rockwell International was selected as prime integrating contractor. A protest of the main engine contract award resulted in a one-year delay to the engine development effort.

By the end of 1972 estimated development costs for the Space Shuttle were $5.15 billion (FY 71 dollars) and the estimated per-flight cost was $10.5 million (FY 71 dollars). Initial launch was anticipated for 1978 and the flight rate was projected to be up to 50 flights per year. By 1980, estimates of the development costs had increased 20% to $6.2 billion (FY 71 dollars), the per-launch cost increased to $15.2 million (FY 71 dollars), the initial launch had slipped to 1981, and the projected flight rate had been reduced to 24 flights per year. These changes resulted from fiscal constraints, early in the development program, and to schedule slips in the engine and thermal protection system development. Despite all these challenges, the first Space Shuttle launch occurred on 12 April 1981 and was spectacularly successful.

During the next five years the Space Shuttle system had many notable successes, despite problems with systems reliability. The Space Shuttle performed the first satellite retrieval, repair, and redeployment. Microgravity experiments for materials sciences and biology were performed. Initial work on space manufacturing of pharmaceutical materials and semiconductor crystal growth experiments was also done. Invaluable experience was gained during long and complicated extravehicular activities.

On 28 January 1986 the 25th mission and the 10th for the orbiter *Challenger* ended in disaster 70 s into the flight when a burn-through of a SRB O-ring resulted in the rupture of the ET and the subsequent break up of the orbiter.

In addition to replacing the *Challenger* with the new orbiter *Endeavour,* NASA implemented a comprehensive return-to-flight strategy. During the 32-month recovery period, 10 major reviews and analyses were undertaken for the program, including Space Shuttle hardware, operation, and organization. Numerous improvements were made to the orbiters, including a new crew escape system, enhanced landing and deceleration systems, safety modifications to the gaseous oxygen flow control valves, and the redesign of the orbiters' 432-mm (17-in.) quick disconnect valves. Also, the SSME underwent an extensive ground-test program to recertify the engine and to demonstrate its reliability through testing at and beyond its original design limits. Upgrades were also made to the flight computers and the orbiter braking systems. Finally, the SRBs were significantly redesigned and thoroughly tested.

As a part of the long-term systematic improvement approach, a number of key improvements were undertaken, including orbiter modifications to enable extended duration missions and refinements to extend the service life of the SSME. The nominal flight rate is 8 flights per year with an annual maximum of 12 flights with a fleet of four orbiters. Currently only those payloads that require a crew present or the unique capabilities of the Space Shuttle are allowed to be manifested on the Space Shuttle.

As a result of the design changes and safety modifications made during the return-to-flight effort, the payload capability of the Space Shuttle decreased from a maximum of 27,850 kg (61,400 lbm) due east to approximately 24,400 kg (53,700 lbm). The West Coast launch and landing facility for Space Shuttle operations, developed by DoD at VAFB for launches into polar orbit, was placed in mothball status, and planned launches out of VAFB were eliminated. A NASA program to develop and use a modified Centaur upper stage to lift 4540 kg (10,000 lbm) to GSO was canceled because of safety concerns. NASA had proposed an advanced SRM (ASRM) to increase payload capability approximately 3630 kg (8000 lbm); this program was canceled in the early 1990s.

The Space Shuttle program performed a wide variety of missions in the 1990s, including spacecraft deployments, the rescue of the Intelsat 603 satellite, microgravity research missions using Spacelab and Spacehab modules, Hubble servicing missions, and 10 visits to the Russian space station Mir as practice for jointly operating the International Space Station. However, since 1998 the Shuttle program has focused almost exclusively on assembling and servicing ISS. Most missions are now dedicated to launching new station elements or rotating logistics modules and crews. One rare exception was a 16-day research flight of *Columbia*, launched in January 2003. During reentry, damage to the carbon–carbon leading edge of the left wing allowed hot gas to penetrate the structure of the wing, causing a catastrophic breakup of the vehicle and loss of the crew. As a result of the tragedy, the Shuttle program is currently on hold while the causes of the accident are determined and fixed.

START

Courtesy United Start Coporation.

The Start-1 is a derivative of the solid-propellant Topol (SS-25) mobile ICBM. It is capable of carrying small payloads to LEO from a variety of launch sites with minimal infrastructure. The Start-1 launch of Earlybird-1 from Svobodny in 1997 is shown.

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START

Start-1

GENERAL DESCRIPTION

Summary

The Start-1 is a converted RT-2PM Topol (SS-25) ICBM. It is one of the smallest space launch vehicles used by Russia and the only one to use all-solid propulsion. Start-1 is marketed internationally for commercial launches by ZAO Puskovie Uslugi in Russia, and United Start Corporation in the United States. The first commercial launch of a Start-1 in 1997 carried the Earlybird earth-imaging spacecraft for the U.S. company Earthwatch. Since then, Start-1 has found a niche, launching small foreign satellites about once a year. It is not used for Russian government payloads. Because Start is based on an actively deployed ICBM, only limited technical information is publicly available.

Status

Operational. First launch in 1993.

Origin

Russia

Key Organizations

Marketing Organization	United Start (International) Puskovie Uslugi (Russia)
Launch Service Provider	STC Complex, Moscow Institute of Heat Technology
Prime Contractor	STC Complex, Moscow Institute of Heat Technology

Primary Missions

Small payloads to LEO

Estimated Launch Price

$9 million (United Start, 1999)

Spaceports

Launch Site	Svobodny LC 5
Location	51.8° N, 128.4° E
Available Inclinations	Specific inclinations from 52 deg to SSO (see Performance section for additional details)
Launch Site	Plesetsk LC 158
Location	62.7° N, 40.3° E
Available Inclinations	76 deg demonstrated, others may be possible

Performance Summary

200 km (108 nmi), 52 deg	632 kg (1393 lbm)
200 km (108 nmi), 90 deg	489 kg (1078 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	Roughly 500 kg (1100 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	167 kg (368 lbm)
GTO	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	5
Launch Vehicle Successes	5
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–1 per year

Nomenclature

The basic Start-1 launch vehicle is based on the Topol ICBM designated RT-2PM or RS-12M in Russia. The NATO designation for the missile is the SS-25 Sickle. The RT-2PM was in turn derived from the RSD-10M Pioneer IRBM, designated SS-20 Saber by NATO. The name Start comes from the Russian word for "launch." The second, larger member of this launch vehicle family was not called the Start-2 as might be expected, but simply "Start." The name Start can therefore refer either to the family of launch vehicles or specifically to the larger of the two systems.

Cost

In 1999 United Start reported that the typical commercial launch price for the Start-1 was about $9 million. This suggests that prices have increased somewhat from the $6–7 million price range reported in the early 1990s when commercialization of the Start began. The Start-1 conversion project was reportedly funded by loans of 3 billion rubles (1 billion rubles in late 1991, and 2 billion in 1993) by a Moscow-based commercial bank. One billion rubles corresponded to roughly $100 million in 1991 at unofficial exchange rates, and 2 billion rubles was equivalent to $1.7 million in 1993 at official exchange rates. Development of the five-stage Start vehicle was funded by a 1 billion ruble investment in mid 1992 by a privately held Russian firm called IVK. The project was also indirectly subsidized by the Moscow Institute of Heat Technology (designers of the RT-2PM missile), which provided engineers for the project, and by the Russian Ministry of Defense, which provided excess missile stages at or below cost.

Availability

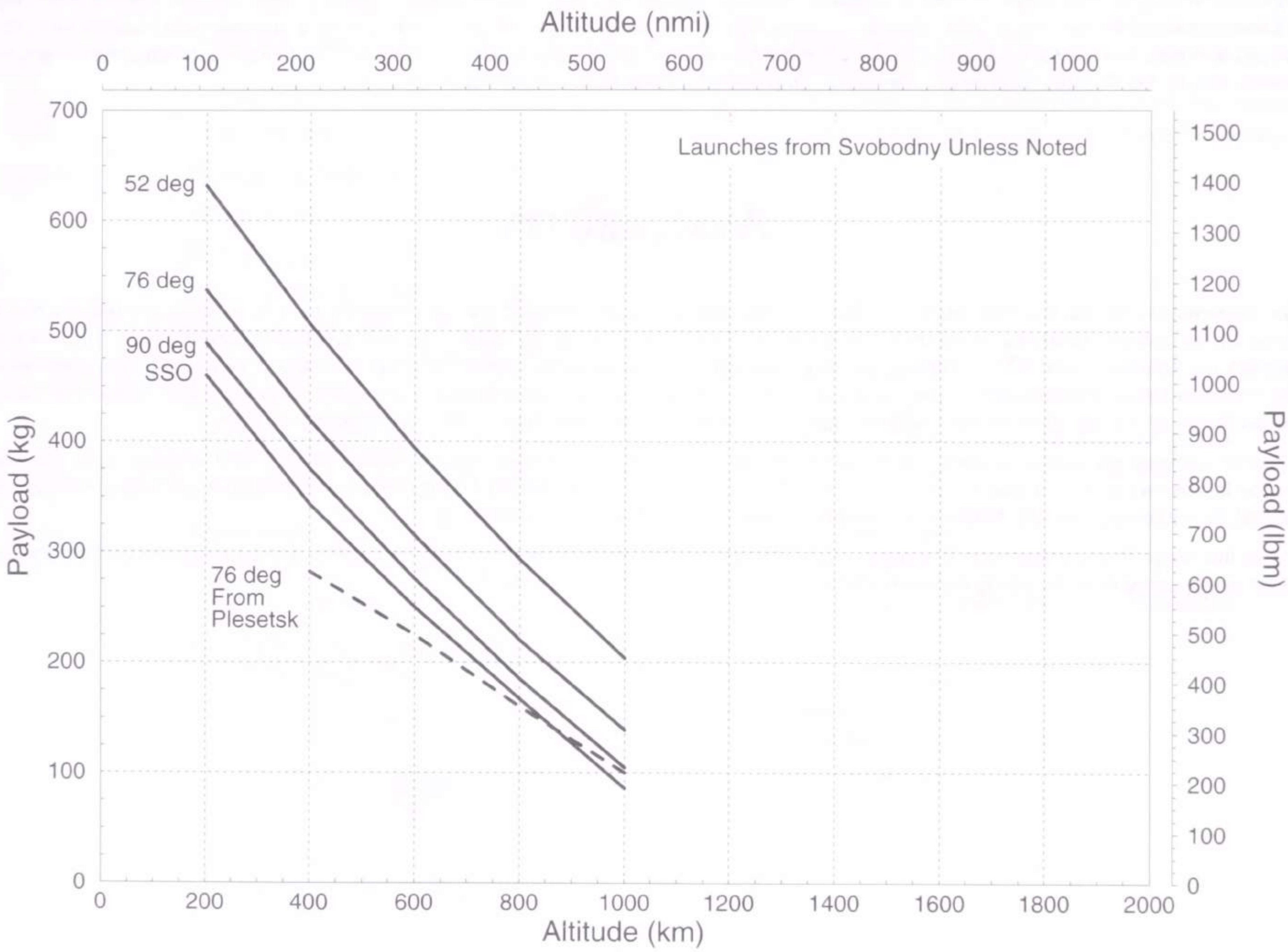
In 1998 responsibility for the Kosmos, Start, and Start-1 small Russian launch vehicles was combined under a new Russian company named ZAO Puskovie Uslugi (which translates to Launch Services). Puskovie Uslugi is jointly owned by the Russian Space Agency (RKA—Rosiyskoye Kosmicheskoye Agentstvo) and STC Complex, which is a subsidiary of the Moscow Institute of Heat Technology. To market these vehicles internationally, Puskovie Uslugi and Assured Space Access, Inc., of the United States formed the United Start joint venture in 1998. United Start is based in the United States and is the international marketing agent for Kosmos, Start-1, and Tsiklon commercial launch services.

Start launch vehicles are produced from retired ICBMs rather than newly built stages. Approximately 450 RT-2PM missiles were manufactured. Production is believed to have ended in 1994. In January 2002 Russia declared in START I treaty documentation that it had 360 operational RT-2PM ICBMs and 48 undeployed ICBMs. Missiles for conversion presumably fall in the latter category.

The larger five-stage Start configuration flew once unsuccessfully in 1995 and has not flown since. It continued to be marketed until around 2000, but has since disappeared from marketing materials and is, therefore, assumed to be retired.

Performance

From the Svobodny Cosmodrome, Start can, in principle, reach inclinations as low as 52 deg and as high as sun-synchronous orbit. However, lower stages will impact on land within Russia. Therefore, missions must be analyzed on a case-by-case basis to determine whether a specific desired inclination can be reached without causing jettisoned stages to fall near populated areas. Because Svobodny and Start are both new, the available launch azimuths have not yet been as fully defined as those for older cosmodromes and launch systems. It is likely that some launch azimuths will not be available, however the Start launch vehicles are some of the most modern in Russia and have yaw steering capability to perform dogleg maneuvers to reach inclinations that might otherwise be restricted. The Start-1 has performed launches to sun-synchronous orbit from Svobodny. While the Russian province of Sakha-Yakutia, north of Svobodny, has formally objected to spent stages falling on its territory, there apparently is enough support within the Russian government to overcome this concern. Launches from Plesetsk have delivered payloads to a 76-deg inclination. It is likely that additional inclinations are feasible from Plesetsk. However, because Svobodny is the primary Start launch site, other Plesetsk capabilities have not yet been documented.

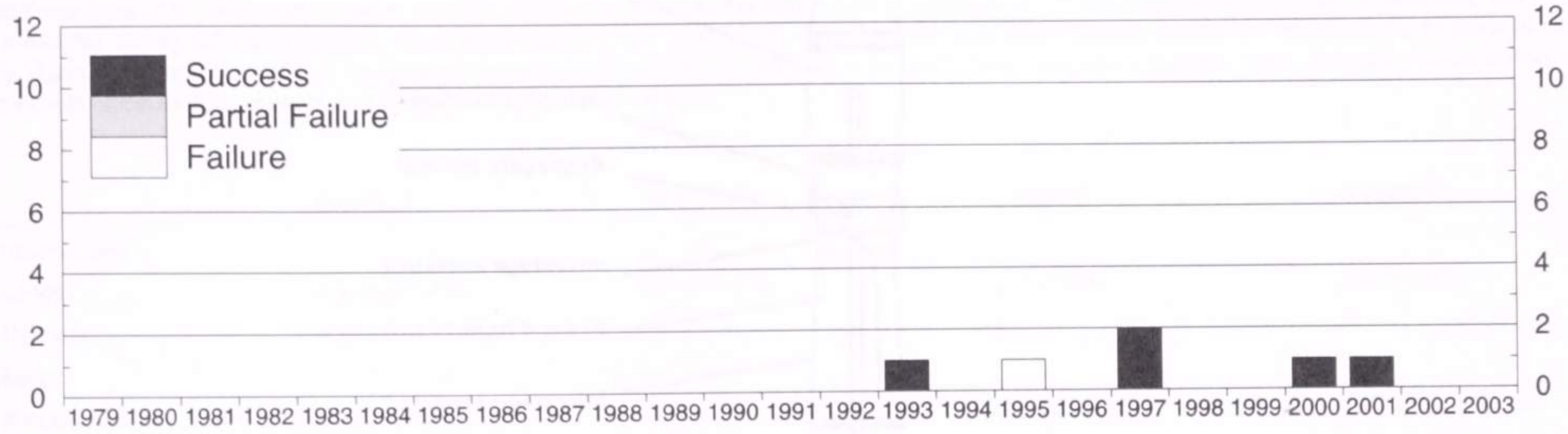


Start-1: Performance to LEO

Flight History

In addition to the orbital launches shown below, the RT-2PM and the older RSD-10M missiles from which the Start-1 and Start are derived have performed over 400 ballistic flights. More than 60 of these have been successful flights of the RT-2PM.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Start-1	Combined Start Family
Total Orbital Flights	5	6
Launch Vehicle Successes	5	5
Launch Vehicle Partial Failures	0	0
Launch Vehicle Failures	0	1

		Date (UTC)	Launch Interval (days)	Model	Launch Site	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1	1993 Mar 25	—	Start-1	Plesetsk	LC 158	1993 014A		EKA 1 (Start 1)	260	LEO (75.8)	CML	Russia
T,F	**2**	**1995 Mar 28**	**733**	**Start**	**Plesetsk**	**LC 158**	1995 F02A	A	Gurwin 1 (Techsat 1)	55	LEO	NGO	Israel
							1995 F02B	A	Unamsat 1	12	LEO	NGO	Mexico
							1995 F02C		EKA 2	200	LEO	CML	Russia
	3	1997 Mar 04	707	Start-1	Svobodny	LC 5	1997 010A		Zeya	87	SSO	NGO	Russia
S	4	Dec 24	295	Start-1	Svobodny	LC 5	1997 085A		EarlyBird 1	284	SSO (97.3)	CML	USA
	5	2000 Dec 05	1077	Start-1	Svobodny	LC 5	2000 079A		EROS A1	240	SSO	CML	Sweden
	6	2001 Feb 20	77	Start-1	Svobodny	LC 5	2001 007A		Odin 1	242	SSO	CIV	Israel

The launch pad for Plesetsk is LC 158; the launch pad for Svobodny is LC 5

T = Test Flight, **F** = Launch Vehicle Failure, **P** = Launch Vehicle Partial Failure, **S** = Spacecraft or Upper-Stage Anomaly

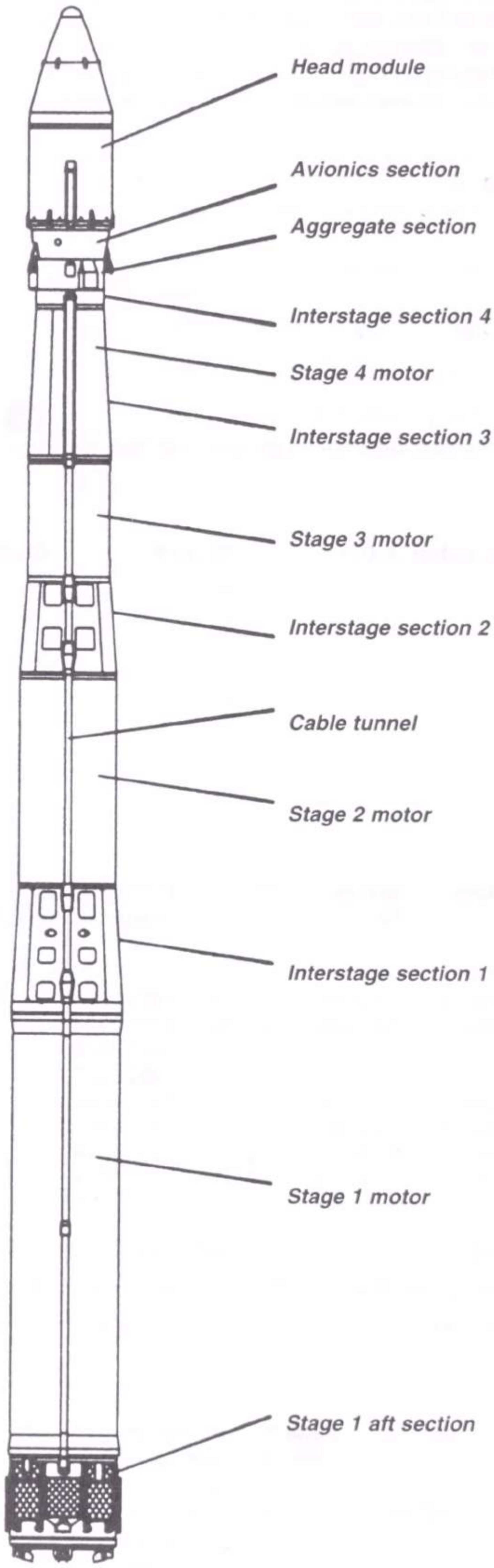
Payload Types: C = Comanifest, M = Multiple Manifest, A = Auxiliary Payload

Failure Descriptions:

F	1995 Mar 28	Start	1995 F02	The flight termination system shut down the fourth stage motor about 12 s before the nominal end of its burn because of arcing in the interface between the FTS and guidance and control system.
S	1997 Dec 24	EarlyBird 1	1997 085A	Satellite was launched successfully, but failed on orbit four days later due to an anomalous undervoltage condition with the onboard power system.

Vehicle Design

Overall Vehicle



	Start-1
Height	22.7 m (74.5 ft)
Gross Liftoff Mass	47 t (104 klbm)
Thrust at Liftoff	890 kN (200 klbf)

Vehicle Design

Stages

The Start-1 uses components from the RT-2PM Topol ICBM. The RT-2PM is a three-stage vehicle, whereas the Start-1 is a four-stage vehicle. Clearly at least one stage must come from a source other than the RS-12M. The fourth stage is believed to come from the RS-14 (SS-X-16) missile, which was tested but never fully deployed; however, this cannot be confirmed. Some systems also are likely taken from the RSD-10M Pioneer (SS-20) IRBM.Because the RT-2PM is still actively deployed, relatively little detailed technical information is publicly available on Start family stages. Therefore, many of the values provided should be considered approximate.

The Start-1 consists of four solid-propellant stages, plus a small post boost propulsion system (PBPS). The PBPS has a solid-propellant gas generator and three pairs of thruster assemblies to provide precise injection into the target orbit. If precision is not required, the system can be removed, resulting in a roughly 100 kg (220 lbm) increase in payload capacity. The Start family comprises the only space launch systems in Russia or the Ukraine to use any significant solid propulsion, a technology largely ignored by the Soviet space program. Solid-propulsion systems have otherwise been limited to small units such as retro-rockets and ullage motors on other vehicles.

	Stage 1	Stage 2	Stage 3	Stage 4
Dimensions				
Length	8.5 m (27.9 ft)	6 m (19.5 ft)	3 m (10 ft)	2.5 m (8 ft)
Diameter	1.6 m (5.3 ft)	?	?	?
Mass				
Propellant Mass (each)	23 t (51 klbm)	11.5 t (25.4 klbm)	5 t (11 klbm)	700 kg (1550 lbm)
Inert Mass (each)	3 t (6.6 klbm)	1.5 t (3.3 klbm)	1 t (2 klbm)	300 kg (650 lbm)
Gross Mass (each)	26 t (57 klbm)	13 t (28.7 klbm)	6 t (13 klbm)	1 t (2200 lbm)
Propellant Mass Fraction	0.88	0.88	0.83	0.70
Structure				
Type	Monocoque?	Filament-wound monocoque	Filament-wound monocoque	?
Material	Glass fiber reinforcement	Composite	Composite	?
Propulsion				
Motor Designation	?	?	?	?
Number of Motors	1	1	1	1
Propellant	Solid	Solid	Solid	Solid
Number of Segments	1	1	1	1
Average Thrust (each)	Sea level: 890 kN (200 klbf) Vacuum: 980 kN (220 klbf)	Vacuum: 490 kN (110 klbf)	Vacuum: 245 kN (55 klbf)	Vacuum: 100 kN (23 klbf)
Isp	Sea level: 238 s Vacuum: 263 s	Vacuum: 280 s	Vacuum: 280 s	Vacuum: 295 s
Chamber Pressure	?	?	?	?
Nozzle Expansion Ratio	?	?	?	?
Attitude Control				
Pitch, Yaw	Jet vanes, aerodynamic panels	Gas-injection thrust-vector control	Gas-injection thrust-vector control	Hydraulically gimbaled nozzle
Roll	Jet vanes, aerodynamic panels	Cold-gas roll control	Cold-gas roll control	Cold-gas roll control
Staging				
Nominal Burn Time	63 s	60 s	63 s	53 s
Shutdown Process	Burn to depletion?	Burn to depletion?	Command thrust termination with 8 thrust termination ports	Burn to depletion?
Stage Separation	?	?	?	?

Vehicle Design

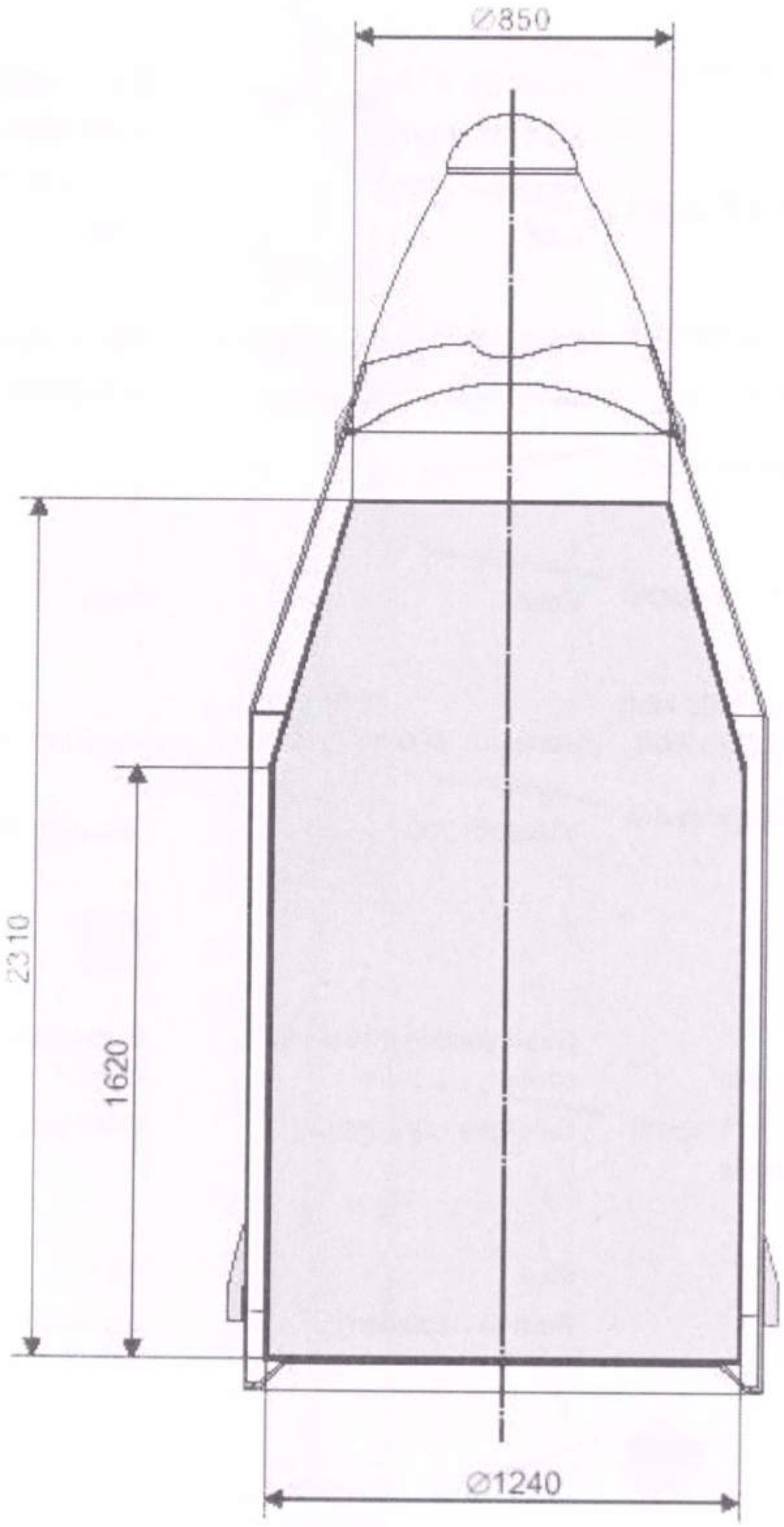
Attitude Control System

The first stage uses jet vanes in the motor exhaust plume to provide initial attitude control. An array of eight rectangular lattice panels, similar to air brakes found on some fighter aircraft, are deployed outward around the base of the first stage after the vehicle clears the launch canister. Four of these are fixed to provide aerodynamic stability, while the other four paddles can be differentially adjusted to provide the primary steering forces. The second and third stages have fixed nozzles with gas-injection thrust-vector control. The injected gas deflects the exhaust in the same way that more conventional liquid-injection thrust-vector control systems work. The fourth solid stage has a hydraulically actuated nozzle gimbal for pitch and yaw control, and a nitrogen cold gas thruster system called GRACS to provide roll control during the boost and three axis control during coast periods.

Avionics

Avionics are contained in a sealed instrumentation compartment. Navigation data is provided by an autonomous gyrostabilized platform. The vehicle receives GLONASS and GPS location data, although this is probably for tracking only. The compartment includes an optical window, perhaps for a star-tracker. An autonomous flight termination system is designed to end the mission if the error in attitude exceeds a predetermined value.

Payload Fairing



	Start-1 Fairing
Length	Payload envelope: 2.31 m (7.6 ft)
Primary Diameter	Payload envelope: 1.24 m (4.07 ft)
Mass	?
Sections	1
Structure	Skin stringer?
Material	Composite

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	1240 mm (48.8 in.)
Maximum Cylinder Length	1620 mm (63.8 in.)
Maximum Cone Length	700 mm (27.6 in.)
Payload Adapter Interface Diameter	1158 mm (45.6 in.)
Payload Integration	
Nominal Mission Schedule Begins	Unknown. Approximately 1 year demonstrated for Earthwatch Earlybird 1 mission.
Launch Window	
Last Countdown Hold Not Requiring Recycling	?
On-Pad Storage Capability	Long-term storage may be possible inside launch canister.
Last Access to Payload	?
Environment	
Maximum Axial Load	8–12 *g*
Maximum Lateral Load	1.25 *g*
Minimum Lateral/Longitudinal Payload Frequency	15 Hz/50 Hz
Maximum Acoustic Level	129 dB
Overall Sound Pressure Level	138 dB
Maximum Flight Shock	2000 *g* at 1000 Hz
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	?
Maximum Pressure Change in Fairing	20 kPa/s (3 psi/s)
Cleanliness Level in Fairing	?
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	Altitude: 5 km (2.7 nmi) Inclination: 0.05 deg Period: 2.5 s
Attitude Accuracy (3 sigma)	± 5 deg
Nominal Payload Separation Rate	0.15–0.7 m/s (0.5–2.3 ft/s)
Deployment Rotation Rate Available	?
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	?
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Availability unknown. Contact United Start.
Auxiliary Payloads	Availability unknown. Contact United Start.

Production and Launch Operations

Production

The Start-1 is built using stages from refurbished RT-2PM ICBMs and other related missiles. The RT-2PM and the Start family of vehicles were designed by the Moscow Institute of Heat Technology (MIHT). MIHT was founded as the design bureau of A. D. Nadiradze in 1948 and became the Institute of Thermal Technology around 1960. It is the only organization in Russia with the capacity to design and build large solid-fueled motors for strategic missiles and space launch applications. (PO Yuzhmash in Ukraine also has solid-propellant casting facilities.) A commercial branch of MIHT, the Scientific and Technical Center Complex or STC Complex (also transliterated NTTs Komplex) is currently responsible for the Start vehicles. Manufacturing of new RT-2PM missiles and refurbishing of missiles into Start launch vehicles is performed by the Votinsky Zavod Association plant, located in the Udmurtia region west of the Ural mountains. Under the INF (Intermediate Nuclear Forces) Treaty, U.S. observers are present at the plant to monitor missile production activities. The Votkinsky factory is one of the oldest in the area, having been created in 1759 to produce ship anchors. Its facilities evolved to produce the first Russian metal steamships and later railway locomotives, bridges, and construction equipment. During World War II the plant began producing artillery pieces and short-range missiles, and from this experience advanced to intermediate and strategic nuclear missiles, including the RSD-10M Pioneer (SS-20 Saber) and RT-2PM Topol (SS-25 Sickle) mobile missiles from which the Start vehicles are derived. Missile production has been dramatically reduced, and the plant now primarily produces civilian products, including oil and gas drilling equipment, heating and cooling systems, machine tools, and washing machines.

Launch Facilities

Courtesy United Start Corporation.

Start-1 Mobile Launcher

Production and Launch Operations

Launch Operations—Svobodny

The primary launch facility for Start-1 launches is the relatively new Svobodny Cosmodrome in the Russian Far East, near the Manchurian border of China. Early test launches were also conducted from the Plesetsk Cosmodrome. Svobodny is the first new Russian Cosmodrome since the 1960s. Following the breakup of the Soviet Union, Russia is now dependent on Kazakhstan for use of the Baikonur Cosmodrome. In 1992, the Russian Military Space Forces proposed that a new facility be established within Russia so that military space launches currently performed at Baikonur would not need to be conducted on foreign soil. Some programs will likely shift to the existing Russian cosmodrome at Plesetsk, but a launch site was desired at a lower latitude, comparable to Baikonur. In February 1993, Colonel General V. Ivanov directed that a study be conducted to determine the best location for the new cosmodrome.

Sites in European Russia near Kapustin Yar were considered. However, launches to high inclinations would overfly Russia's major cities, and launches to low inclinations would overfly foreign countries such as Kazakhstan. Sites in southern Siberia in the area of Lake Baikal were rejected because the terrain is mountainous and launch vehicles would have to avoid overflight of China. The area near Vladivostok was attractive because it is farther south than Baikonur, but northerly launches would overfly China, and low inclination launches would overfly Japan. Finally, the options were narrowed to two sites. Sovetskaya Gavan, on the east coast of Russia opposite Sakhalin Island at a latitude of about 48°, was attractive because all inclinations would be available, with stages falling in the Sea of Okhotsk or in Russian territory. Alternatively, the existing ICBM base at Svobodny, while not coastal, would also allow launches to all inclinations without overflight of foreign territory. In late 1993, Svobodny was selected because it is on the Trans-Siberian Railway, and because the existing infrastructure of the missile base would dramatically reduce development costs. The decision to proceed with the cosmodrome was made by presidential decree on 1 March 1996.

Svobobody is a 250 km (150 mi) drive from the nearest significant airport, at Blagoveschensk The Svobodny base is located about 20 km (12 mi) from the nearest community, and the Start launch area in turn is located only about 5 km (3 mi) from the residential center of the base. The launch area includes a spacecraft assembly and test building and launch vehicle preparation building. The spacecraft preparation facility has bridge cranes, tent-type clean room areas, and hydrazine loading equipment. The payload is integrated horizontally onto a launch plate, then encapsulated into the payload fairing. This encapsulated "space head assembly" is then integrated onto the launch vehicle.

Start-1 launch vehicles are transported and launched from inside a launch canister mounted on a seven-axle mobile transporter/erector truck built for the RT-2PM ICBM by the Minsk Automobile Plant and Shumerlya Specialized Automobiles Manufacturing Plant. While this system is generally referred to as road mobile, to distinguish it from railroad-based mobile missiles, the mobile transporter can easily travel off-road as well. The transporter vehicle is driven to a flat concrete pad and large hydraulic foot pads are lowered to stabilize and level the system. While the location of the concrete pad is presurveyed, the RT-2PM has a system that can rapidly and accurately determine its position on the ground before launch, and the Start family of vehicles may also have this capability to launch from nonsurveyed sites. Specialized service vans are positioned in close proximity to the mobile launcher to generate power, provide heated air to maintain the temperature of the rocket and spacecraft, and to house the customer's electrical ground support equipment for the payload. The launch pad is only about 300 m (1000 ft) away from the launch control bunker and 600 m (2000 ft) from the spacecraft and launch vehicle test buildings.

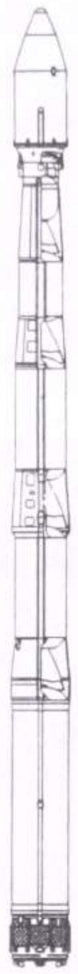
At the appropriate launch time, several things happen in a matter of seconds. The forward protective cover of the launch container is jettisoned, the canister is lifted to the vertical position using large hydraulic pistons, and then the rocket is ejected from the canister by a gas generator. Once the vehicle clears the canister, the sabots fall free, and the aerodynamic paddles deploy outward to provide stability and control. The first-stage motor then ignites and the rocket climbs rapidly away. The canister, although scorched by the launch, can be cleaned and reused. Following first stage burnout, there is a coast period of about 20 s, during which the first stage remains attached. This allows the vehicle to climb to a higher altitude and reduce dynamic pressure for the staging event.

Launch Operations—Other Sites

The Start launch infrastructure is road mobile, and the launch vehicle, transporter/erector, and support equipment can be transported by heavy aircraft. Therefore, Start launches could be performed from any launch facility in the world without requiring a new vehicle-specific infrastructure. Only standard payload processing and tracking facilities and a suitable concrete pad would be required. Because of this flexibility, Start launches have been considered from a number of different sites, including existing U.S. spaceports at Vandenberg, Cape Canaveral, and Wallops Island. New facilities, such as the Kodiak Launch Complex in Alaska, the now-defunct Spaceport Canada planned for Churchill, Manitoba, and Australian launch sites near Darwin or Woomera have also been the focus of plans by a variety of U.S., Canadian, and Australian companies. Currently there are no active plans to launch Start outside of Russia.

Vehicle History

Vehicle Evolution

	Retired	Operational
		
Vehicle	**Start**	**Start-1**
Period of Service	1995	1993–present
Polar LEO payload	645 kg (1420 lbm)	489 kg (1078 lbm)

Historical Summary

The deployment of large, road-mobile, solid-propellant missiles in the Soviet Union began in the 1970s with the development of the RSD-10M Pioneer (SS-20 Saber) IRBM. The first launch was conducted in 1974, active deployment followed in 1977, and up to 850 were produced. Subsequently, the larger RT-2PM Topol (SS-25 Sickle) with intercontinental range was tested in 1983 and deployed beginning in 1985. More than 450 missiles were produced. In the late 1980s the RSD-10M was retired, and under the INF Treaty, the missiles were destroyed. The RT-2PM is still actively deployed, and may still be in low-rate production.

The first proposals for the Start launch vehicle family appeared in 1989, but little activity occurred until 1991 or 1992, when MIHT set up the joint stock company STC Complex to market the Start, with investment from a privately held computer company called IVK. Development of the Start family has been a quasi-commercial project, with private investment and bank loans used for funding the project, but direct support from the Ministry of Defense and MIHT has also been crucial to the success of the project. In 1993 the first Start-1 launch occurred from Plesetsk, carrying test instrumentation. The first test launch of the larger Start configuration was conducted in 1995, but failed when the fourth-stage thrust termination system fired prematurely. The first commercial contract for Start-1 launch services of a primary payload was signed in early 1997, for the launch of the Earlybird-1 imaging satellite for the U.S. company Earthwatch. Before the launch of Earlybird, STC Complex conducted the first space launch from the Svobodny Cosmodrome in March 1997 by launching the Zeya spacecraft, built by a Russian university. The spacecraft is presumably named after the Zeya River that flows past the cosmodrome. The first commercial launch took place on 24 December 1997 when the Start-1 successfully placed the Earlybird-1 spacecraft into a 475 km sun-synchronous orbit. The launch campaign was conducted by STC Complex/MIHT and by Assured Space Access, Inc., of the United States, which had helped market Start launch services to international customers. Following the successful flight, STC Complex and the Russian Space Agency formed the joint stock company ZAO Puskovie Uslugi (Launch Services) to market and operate the Start family, as well as the Kosmos small launch vehicle. Puskovie Uslugi in turn partnered with Assured Space Access to form the U.S. company United Start, which is now responsible for international marketing of both Start and Kosmos.

STRELA

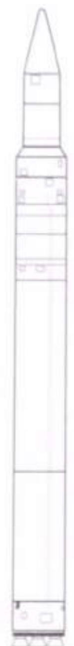
Strela is a small space launch vehicle based on Russia's RS-18 (SS-19) ICBM.

Contact Information

Strela Business Development:
NPO Mashinostroyenia
33 Gagarin St.
Reutov, Moscow Region
143951
Russia
Phone: +095 307-04-96
Fax: +095 302-20-01

General Description

Strela



Strela

Summary

The Strela launch system uses decommissioned Russian RS-18 (SS-19) ICBMs. Unlike Rockot, which is also based on the RS-18, Strela uses only existing RS-18 hardware without additional stages. Strela is marketed by NPO Mashinostroyenia and is to be launched from the new Russian spaceport at Svobodny.

Status

Operational. First launch in 2003

Origin

Russia

Key Organizations

Marketing Organization	NPO Mashinostroyenia
Launch Service Provider	NPO Mashinostroyenia
Prime Contractors	NPO Mashinostroyenia

Primary Missions

Small satellites to high inclination LEO

Estimated Launch Price

$10.5 million (NPO Mashinostroyenia, 1999)

Spaceport

Launch Site	Svobodny Cosmodrome, Russia
Location	51.8° N, 128.2° E
Available Inclinations	51.8–61 deg and 90–99 deg
Launch Site	Baikonur Cosmodrome, Kazakhstan
Location	45.6° N, 63.4° E
Available Inclinations	63 deg

Performance Summary

250 km (130 nmi), 51.8 deg	1560 kg (3440 lbm)
250 km (130 nmi), 90 deg	1170 kg (2580 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	1450 kg (3200 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	700 kg (1540 lbm)
GTO: 200×35,786 km (108×19,323 nmi), 0 deg	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	1
Launch Vehicle Successes	1
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

Unknown

Nomenclature

The Strela launch vehicle is derived from the RS-18 ICBM, designated the SS-19 Stiletto by NATO. The missile was designated the UR-100N by its manufacturer, and is part of the same "Universal Rocket" series as the Proton (UR-500). Strela is produced by the Scientific Production Machine Building Association, known in Russia as NPO Mashinostroyenia. The name is also abbreviated to NPO Mash or NPOM.

Cost

NPO Mashinostroyenia reported in 1999 that it will charge about $10.5 million for Strela launches. The cost of converting launch facilities at Svobodny, including one missile silo, is estimated at $20 million. Although the Strela program is endorsed by the Russian government, it receives no direct funding from the national budget. Funding comes from nonbudgetary sources.

Availability

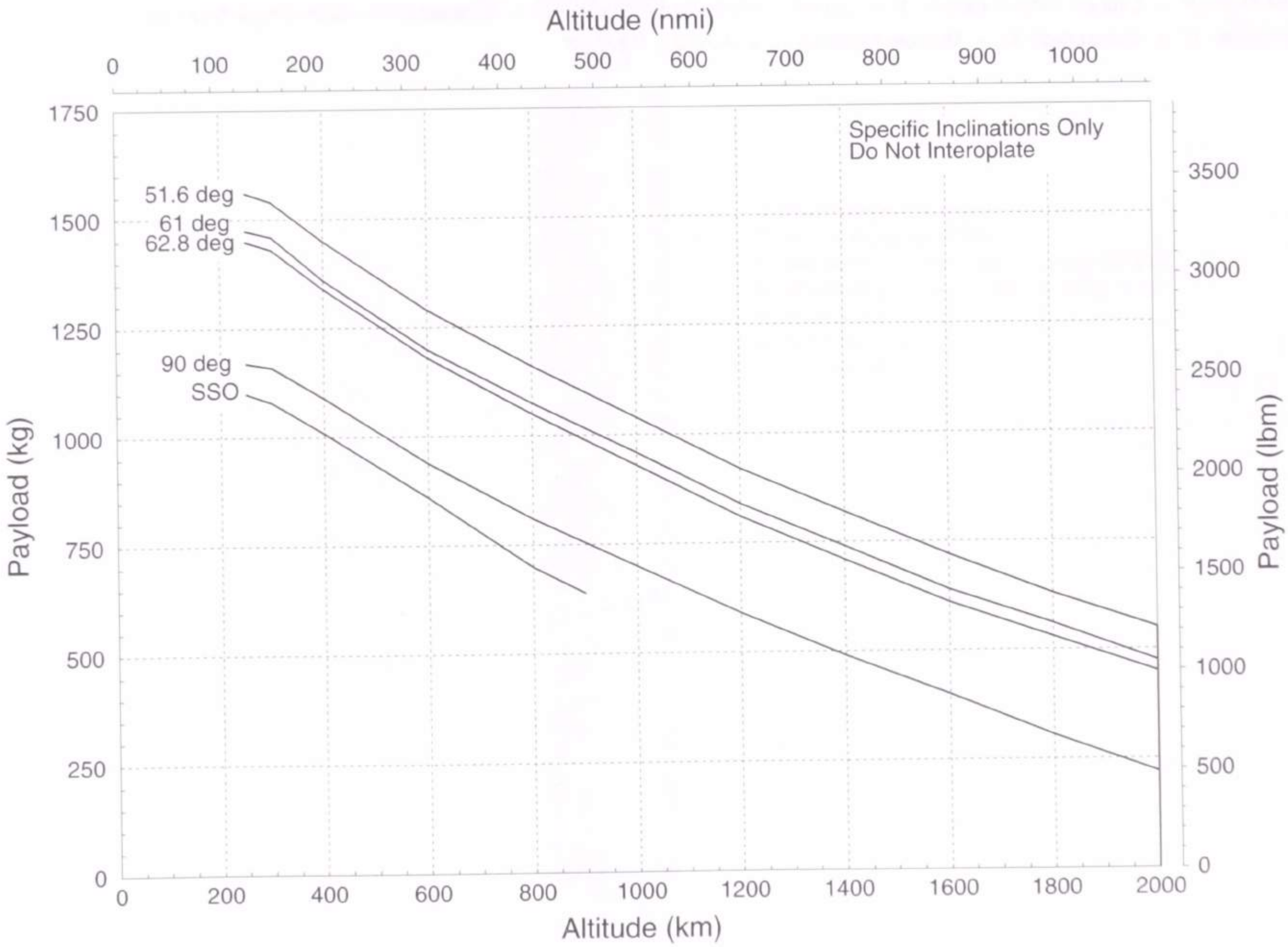
Strela launch vehicles are decommissioned RS-18 ICBMs. The missiles are no longer produced, and Strela launches will have to come from the existing stock of missiles being surplused as a result of disarmament agreements. About 360 RS-18 missiles were produced between 1977 and 1991. As of January 2002, Russia had 137 deployed missiles. Forty-five additional missiles have been assigned to Khrunichev for the Rockot program. The number of missiles available to NPO Mashinostroyenia is unclear.

Performance

Strela is launched from the new Svobodny Cosmodrome in the Russian Far East. Like Baikonur and Plesetsk, Svobodny has limited available inclinations because jettisoned stages fall on land. In this case, stages typically fall in the neighboring territory of Sakha-Yakutia to the north of Svobodny.

Strela has a postboost stage called the Mechanisms and Instruments Section (MIS) that can perform circularization burns to reach orbits at high LEO altitudes. Performance shown reflects the use of the SHS-2 payload fairing. If the smaller SHS-1 fairing is used the capability is 105 kg (230 lbm) greater.

Although Strela cannot launch satellites into GTO or GEO by itself, NPO Mashinostroyenia has proposed to launch spacecraft that could reach GEO using additional upper stages and onboard electric propulsion systems. The company has a contract to build Ruslan-MM satellites, which would be launched on Strela using this approach with mass about 520 kg (145 lbm) in GEO.



Flight History

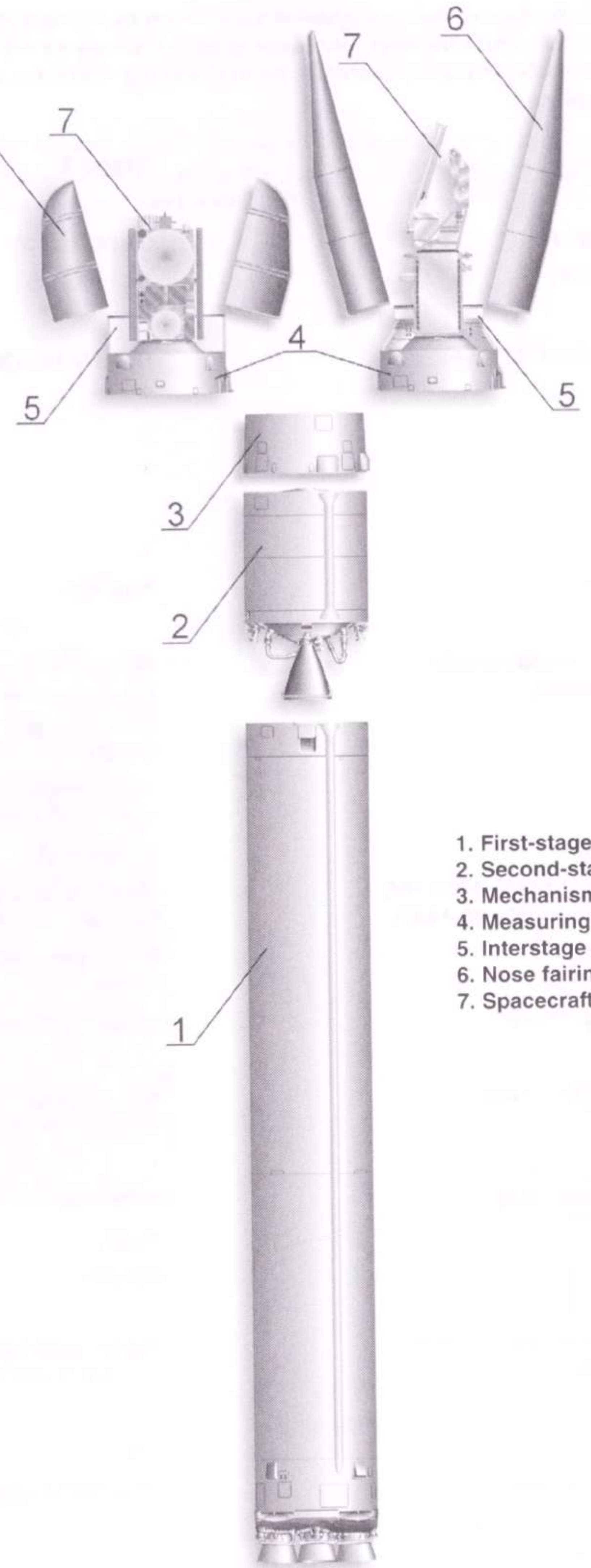
Orbital Flights Per Year

		Date (UTC)	Launch Interval (days)	Model	Launch Site	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	1	2003	Dec 05	Strela 1	Baikonur	LC 132	2003 055A	Kondor E	GVM800	LEO (67)	CML	Russia

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly
Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Vehicle Design

Overall Vehicle



1. First-stage booster
2. Second-stage booster
3. Mechanisms and instruments section
4. Measuring instruments compartment
5. Interstage
6. Nose fairing
7. Spacecraft

	Strela
Height	26 or 29.2 m (85.3 or 95.8 ft)
Gross Liftoff Mass	104 t (229 klbm)
Thrust at Liftoff	1870 kN (420 klbf)

Vehicle Design

Stages

The first and second stages of Strela are decommissioned RS-18 ICBM stages, and therefore relatively little information is available about them. Each stage contains tanks for N_2O_4 and UDMH, separated by a common bulkhead.

In addition to two primary stages, Strela uses the postboost stage and guidance section from the RS-18 ICBM. The mechanisms and instruments section (MIS) contains the control system, power supply, and a low-thrust propulsion system. This may be the RD-0237 engine, which generates 4.9 kN (1100 lbf). Above the MIS is the measuring instruments compartment, which is 0.8 m (2.6 ft) high and 2.4 m (7.9 ft) in diameter. It contains tracking and telemetry systems as well as flight termination systems.

	Stage 1	Stage 2
Dimensions		
Length	17.2 m (56.4 ft)	3.9 m (12.8 ft)
Diameter	2.5 m (8.2 ft)	2.5 m (8.2 ft)
Mass		
Propellant Mass	Roughly 80 t (175 klbm)	Roughly 14 t (30 klbm)
Inert Mass	?	?
Gross Mass	?	?
Propellant Mass Fraction	?	?
Structure		
Type	?	?
Material	Aluminum	Aluminum
Propulsion		
Engine Designation	RD-0233 (Khimavtomatiki Design Bureau)	RD-0235 engine with RD-0236 verniers (Khimavtomatiki Design Bureau)
Number of Engines	4	1 main engine + 1 vernier engine with 4 nozzles
Propellant	N_2O_4/UDMH	N_2O_4/UDMH
Average Thrust (Total)	Sea level: 1870 kN (420.4 klbf) Vacuum: 2070 kN (465.4 klbf)	Main Engine: 240 kN (54 klbf) Verniers: 15.76 kN (3540 lbf)
Isp	Sea level: 285 s Vacuum: 310 s	Main engine: 320 s Verniers: 293 s
Chamber Pressure	205 bar (2975 psi)	75 bar (1090 psi)
Nozzle Expansion Ratio	?	?
Propellant Feed System	Closed-cycle turbopump	Main engine: closed-cycle turbopump Verniers: single shared turbopump
Mixture Ratio (O/F)	?	?
Throttling Capability	Yes, unknown range	100% only
Restart Capability	None	None
Tank Pressurization	Hot gas	Hot gas
Attitude Control		
Pitch, Yaw, Roll	Single axis nozzle gimbal on each engine	Vernier system with single axis gimbal on each nozzle
Staging		
Nominal Burn Time	121 s	183 s
Shutdown Process	Command shutdown	Command shutdown
Stage Separation	Four retro-rockets. Stage 2 vernier thrusters	?

VEHICLE DESIGN

Attitude Control System

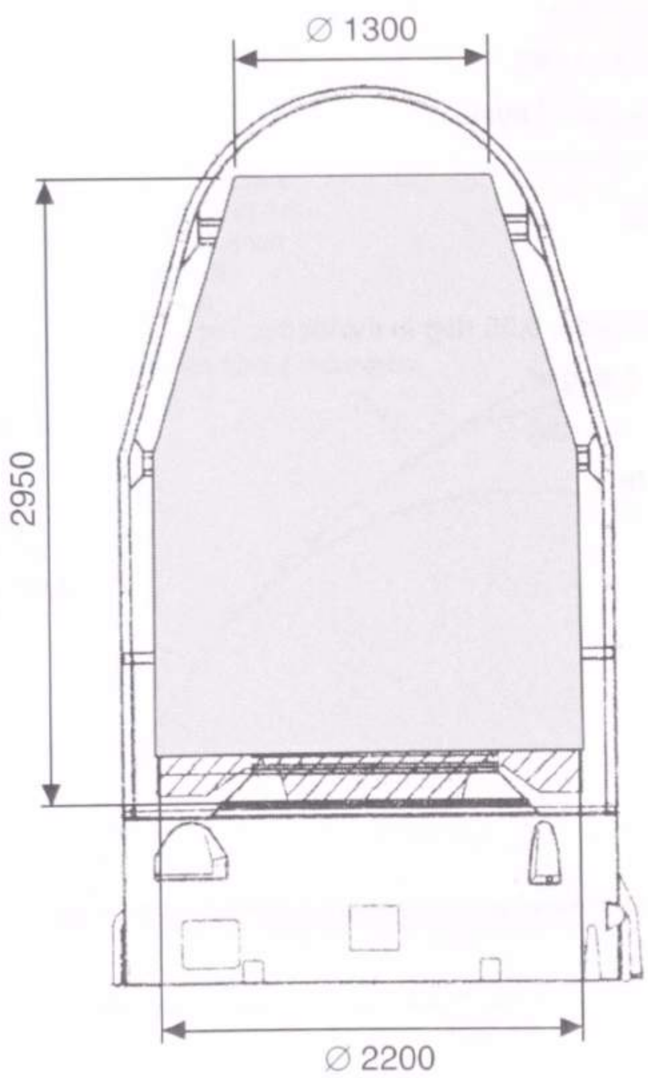
The first-stage attitude is controlled by a single-axis gimbal on each of the four engines. The second-stage attitude is controlled with a single vernier engine that has four nozzles.

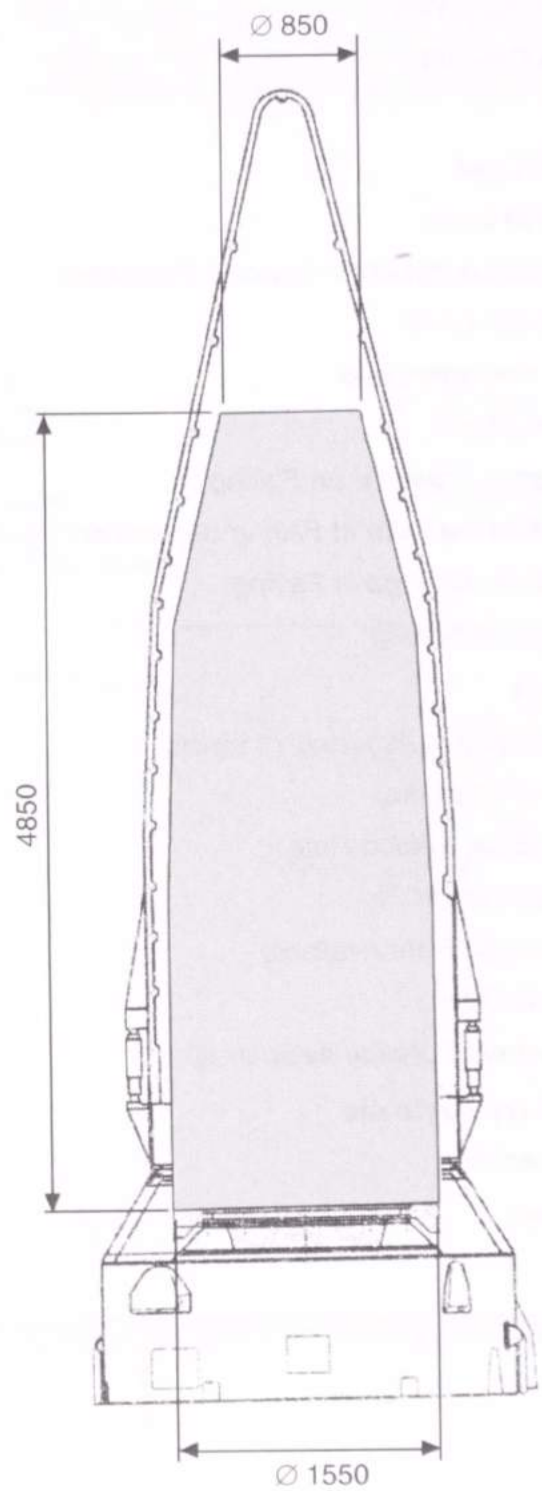
Avionics

No information available.

Payload Fairing

Two variations of the payload encapsulation system, dubbed the space head section (SHS) are available. The SHS consists of the measuring instruments compartment, a payload adapter, a payload fairing, and a structure mounted between the faring and the measuring instruments compartment. The SHS-1 is based on the fairing used in operational missiles and is the shorter and wider of the two. The SHS-2 is taller and narrower and was used for flight tests of the missile.





Dimensions in mm

	SHS 1	SHS 2
Length	?	?
Primary Diameter	2.5 m (8.2 ft)	?
Mass	?	?
Sections	2	2
Structure	Skin Stringer?	Skin Stringer?
Material	?	?

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	SHS-1: 2200 mm (86.6 in) SHS-2: 1550 mm (61.0 in)
Maximum Total Length (cylinder + cone)	SHS-1: 2950 mm (116.1 in) SHS-2: 4850 mm (190.9 in)
Payload Adapter Interface Diameter	Various, including 1194 mm (47.0 in)
Payload Integration	
Nominal Mission Schedule Begins	T–18 months
Launch Window	
Last Countdown Hold Not Requiring Recylcing	?
On Pad Storage Capability	?
Last Access to Payload	T-18 months
Environment	
Maximum Axial Load	8.7 *g*
Maximum Lateral Load	2 *g*
Maximum Lateral/Lontigtudinal Payload Frequency	15 Hz/40 Hz
Maximum Acoustic Level	138 dB at 50–1000 Hz in silo, 129 at 250–1000 Hz in flight
Overall Sound Pressure Level	149 dB in silo, 140 dB in flight
Maximum Flight Shock	5000 *g* at 2–10 kHz
Maximum Dynamic Pressure on Fairing	65.8 kPa (1375 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	?
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	1% error in altitude, 0.05 deg in inclination
Attitude Accuracy (3 sigma)	±1.5 deg, ±2.5 deg/s°
Nominal Payload Separation Rate	0.1–0.9 m/s (1–3 ft/s)
Nominal Payload Tip Off Rate	> 0.6 m/s (2 ft/s)
Deployment Rotation Rate Available	?
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	?
Multiple/Auxiliary Payloads	
Multiple or Comanifest	?
Auxiliary Payloads	?

Production and Launch Operations

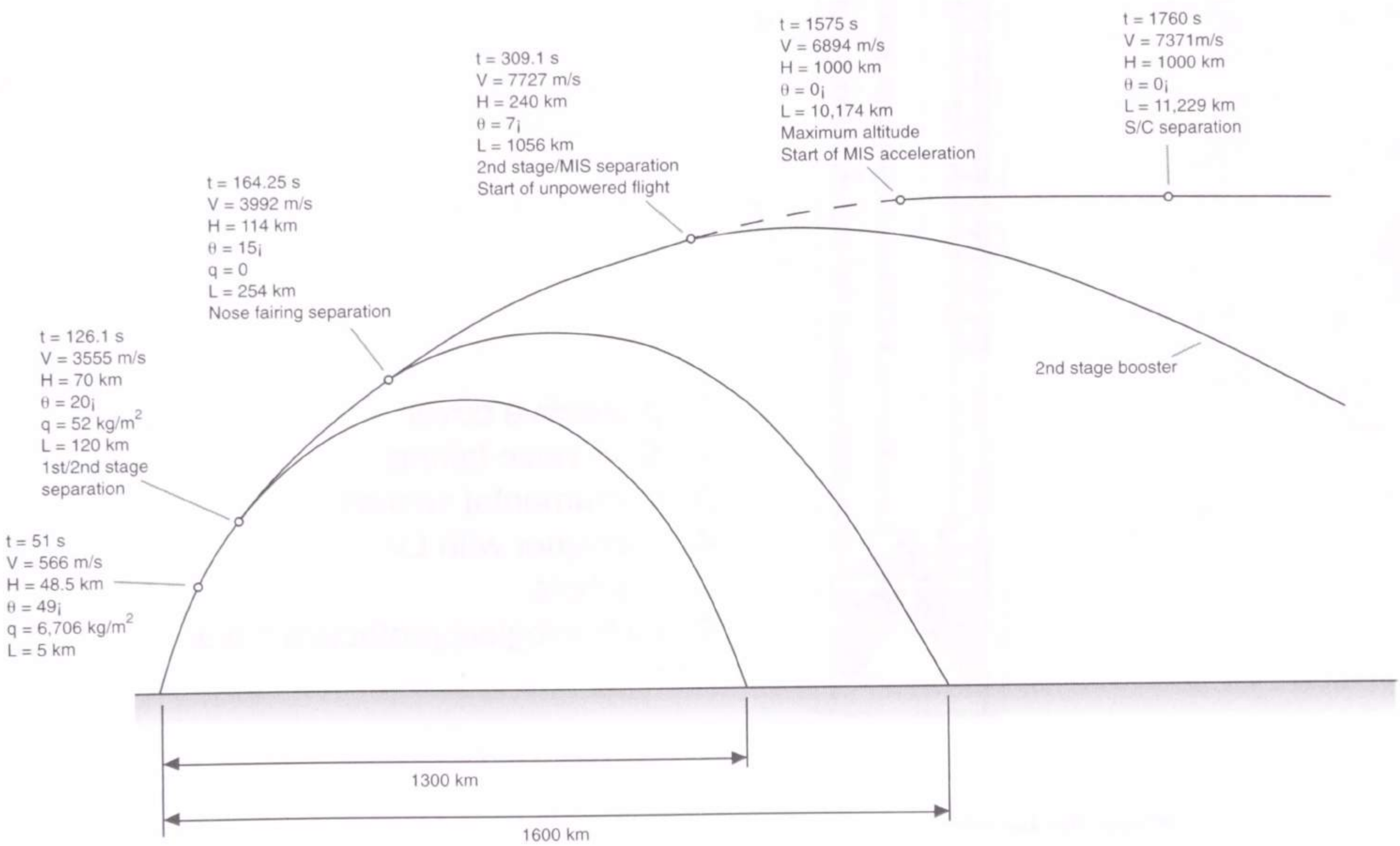
Production

The first and second stages of Strela are retired stages from RS-18 (SS-19) ICBMs. The missiles are drained of propellant, removed from their silos, and maintained in climate-controlled storage inside their transport containers in Russian military facilities. Reliability is verified with periodic tests, including engine hot fire tests. The first- and second-stage engines were produced by the Khimavtomatiki Design Bureau. Modifications to convert the ICBM to a Strela configuration primarily involve reprogramming the flight software with little change to the hardware.

Launch Operations

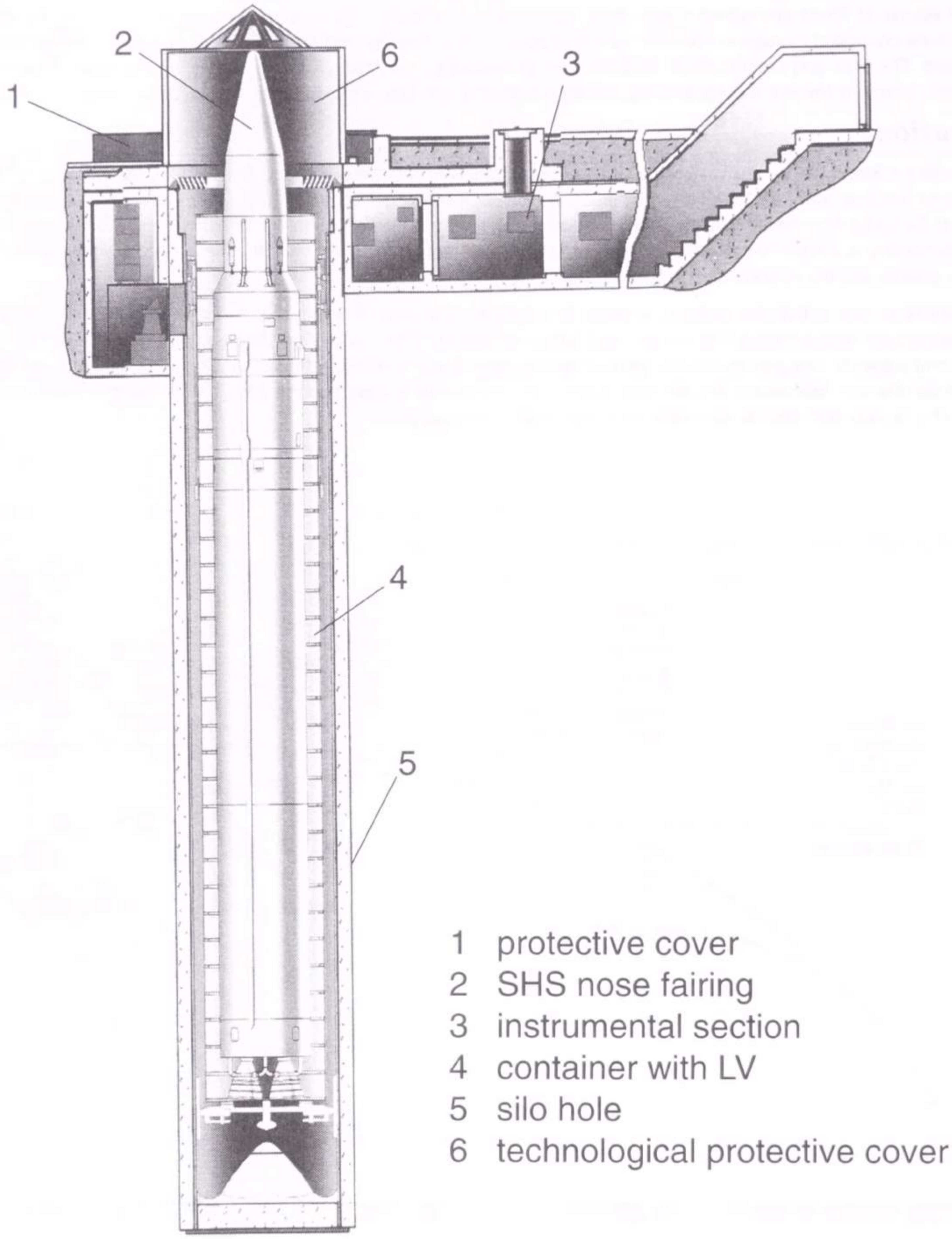
The primary launch site for Strela will be the Svobodny Cosmodrome, officially called the Second State Test Cosmodrome of the Ministry of Defence. Launches from Baikonur are also possible. Svobodny was designated a cosmodrome in the mid 1990s after the collapse of the Soviet Union left Russia's primary launch site at Baikonur in a newly foreign country. Svobodny provides a lower latitude alternative to the other major Russian launch site at Plesetsk. However, Svobodny is a much smaller facility. It has a launch area for mobile Start launch vehicles and a single silo for Strela. For more information on Svobodny, please see the chapter on the Start launch system.

The Strela launch vehicle arrives at the cosmodrome at least 19 days before launch. It is mounted in the launch silo for electrical checks. The spacecraft payload is processed and encapsulated into the payload fairing separately. The payload building has a 12×30 m (40×100 ft) test hall with a bridge crane of 5 t (11,000 lbm) capacity. Integration into the payload fairing takes place in a second building. The encapsulated payload, fairing, and avionics section of the launch vehicle are delivered to the silo and attached to the missile stages about 8 days before launch. The launch vehicle is fueled 2–4 days before launch. The launch operation is controlled from the base command post.



Strela Flight Profile

PRODUCTION AND LAUNCH OPERATIONS



Universal Silo Launcher

Vehicle History

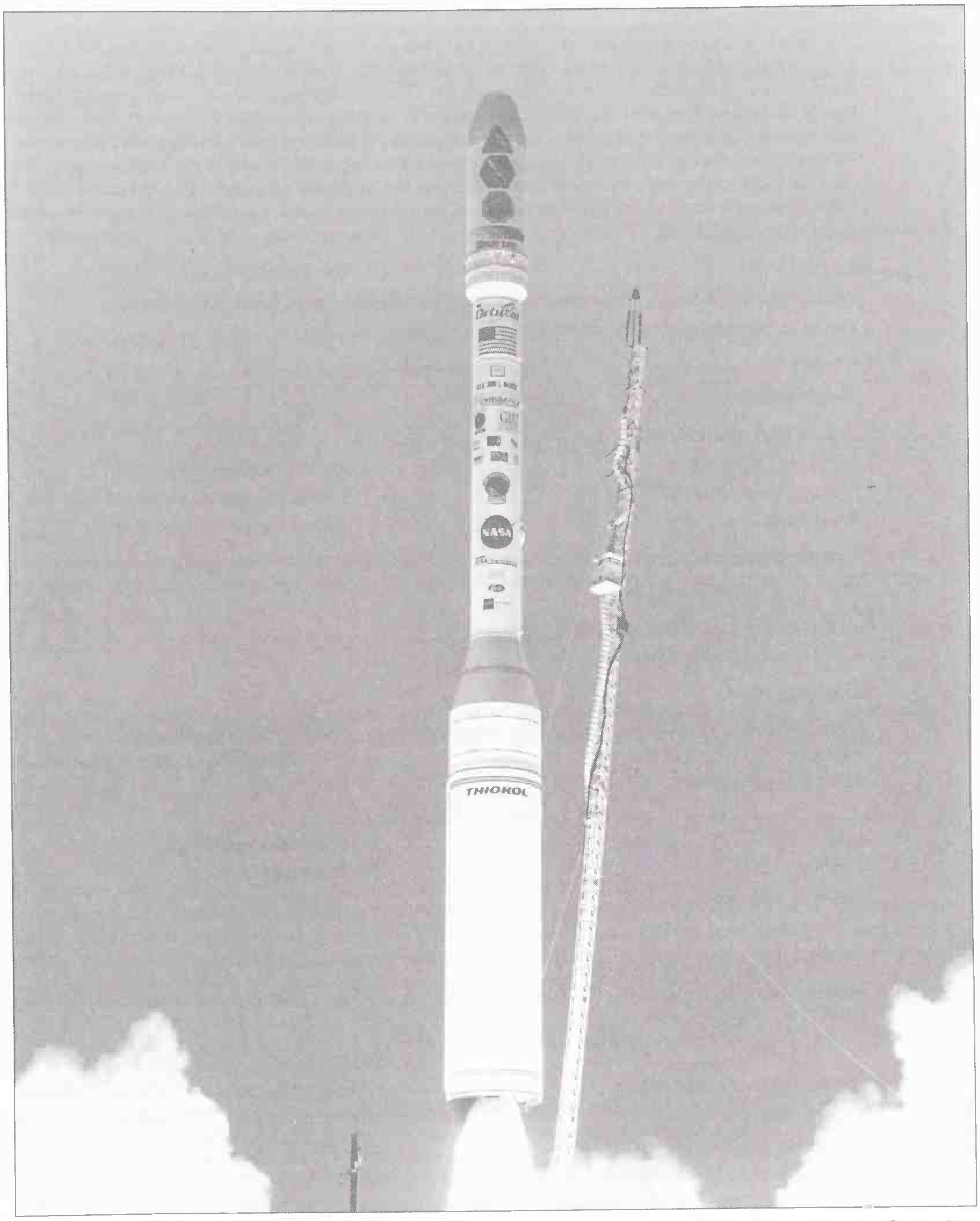
The Rockot and Strela launch vehicles are based on the RS-18 ICBM, also known as UR-100NU in Russia and SS-19 Mod 2 in the West. The RS-18 was developed between 1975 and 1977 as a replacement for the earlier UR-100 (SS-11) ICBM and was commissioned into service in 1979. It was designed by NPO Mashinostroyenia and its KB Salyut design bureau. The Salyut design bureau later became part of the Khrunichev State Research and Production Space Center. About 360 of the missiles were built through 1991.

In the mid 1980s the Soviet Ministry of Defense funded the development of a space launch vehicle derived from the RS-18, which was given the name Rockot. The Breeze upper stage was added to the two missile stages to increase the capability of the vehicle for space launch missions. Test flights of Rockot were conducted before government funding ran out. Rockot would eventually reemerge as a commercial effort under the auspices of Khrunichev.

Although Strela is derived from the same hardware as Rockot, it has a different programmatic history. In October 1992 the Russian federal government issued a decree "On Rational Use in the National Economy of Missile Complexes Which Are to be Liquidated Pursuant to Reduction and Limitation of Strategic Offensive Weapons" that enabled retired missiles to be turned into space launch systems. In May 1994 another decree formalized the federal program for industrial use of these missiles. In the 1994 decree NPO Mashinostroyenia was designated as the lead contractor for the conversion of RS-18s into Strela launch vehicles. In March 1996, the government created the Svobodny Cosmodrome at the site of a former Strategic Rocket Forces ICBM base. The site was a good fit for Strela, because it had been home to RS-18 missiles and therefore was already equipped with silos and support equipment compatible with Strela.

Although Strela is part of the Russian space program, it is not funded by the government. Funding must come from "non-budgetary sources." NPO Mashinostroyenia also does not have a Western marketing partner. These factors have contributed to the fact that Strela has not yet performed a space launch several years after the initiation of the program. There have been some successes though. In 2000, the U.S. company Transorbital announced that it had signed a contract to launch its commercial lunar probe on a Strela. However, the mission was shifted to a Dnepr later. In 2001, a contract with Intersputnik for construction and launch of two small GEO communications satellites was announced.

TAURUS



Courtesy Orbital Sciences Corporation.

Taurus is a higher performance, ground-launched derivative of Orbital Sciences Corporation's Pegasus vehicle. Flight 6, using the Commercial Taurus configuration with the 63-in. payload fairing, is shown.

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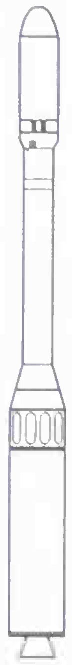
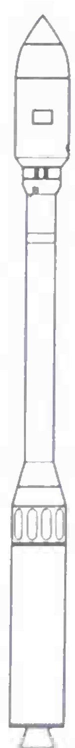
General Description

Taurus



SSLV Taurus

Commercial Taurus



Taurus XL

Summary

The Taurus is a ground-launched derivative of the Pegasus launch vehicle, which combines the Pegasus motors and avionics with a larger first stage. This vehicle was developed to meet the requirements of DARPA's SSLV program, which emphasized rapid-response launch capability from a remote launch site. The SSLV configuration of Taurus uses a government-furnished Peacekeeper missile first stage. The demonstration launch under this contract successfully occurred in 1994. Following this launch, the contract was transferred to the U.S. Air Force where two options for additional SSLV-configuration Taurus missions have been exercised. Following the initial flight of the SSLV, Orbital developed an upgraded version of the Taurus vehicle. This commercial vehicle differs from the SSLV configuration primarily through the use of ATK Thiokol's Castor® 120 solid rocket motor for the first stage in place of the Peacekeeper stage of the SSLV configuration. An enhanced Taurus, called Taurus XL, was launched in 2004. It uses the longer motors from Pegasus XL.

Status

Operational. SSLV Taurus first launch in late 1994. Commercial Taurus first launch in 1998.

Taurus XL operational, first launch in 2004.

Origin

United States

Key Organizations

Marketing Organization	Orbital Sciences Corporation
Launch Service Provider	Orbital Sciences Corporation
Prime Contractor	Orbital Sciences Corporation

Primary Missions

Small LEO and GTO payloads

Estimated Launch Price

$25–47 million (OSC, 2002)

Spaceports

Launch site	Vandenberg AFB, SLC 576E
Location	34.7°N, 120.6°E
Available Inclinations	55–120 deg
Launch site	Cape Canaveral AFS
Location	28.5°N, 81.0°W
Available Inclinations	28.5–40 deg

Performance Summary

Performance is shown with 63-in. (1.6-m) payload fairing. GTO performance includes Star 37 upper stage.

	SSLV Taurus	Commercial Taurus	Taurus XL
200 km (108 nmi), 28.5 deg	1310 kg (2890 lbm)	1370 kg (3020 lbm)	1590 kg (3505 lbm)
200 km (108 nmi), 90 deg	1000 kg (2205 lbm)	1060 kg (2340 lbm)	1100 kg (2450 lbm)
Space Station Orbit: 407 km (220 nm), 51.6 deg	1090 kg (2400 lbm)	1150 kg (2540 lbm)	?
Sun-Synchronous Orbit: 800 km (432 nm), 98.6 deg	660 kg (1460 lbm)	720 kg (1590 lbm)	860 kg (2000 lbm)
GTO: 185×35,786 km (100×19,323 nm), 28.5 deg	400 kg (880 lbs)	495 kg (1090 lbs)	557 kg (1228 lbm)
GEO	No capability	No capability	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	6
Launch Vehicle Successes	4
Launch Vehicle Partial Failures	1
Launch Vehicle Failures	1

Flight Rate

0–1 per year

Nomenclature

There are three basic Taurus configuration types. The Standard Small Launch Vehicle (SSLV) Taurus uses the Peacekeeper ICBM first stage with standard length Orion upper-stage motors, and is used to launch USAF payloads. The Commercial Taurus uses the Castor 120 first stage motor with standard length Orion motors. The Taurus XL uses the Castor 120 with extended length Orion XL motors. For performance enhancement, the ATK Thiokol Star 37 motor can be used in place of, or in addition to, the standard Orion 38 upper-stage motor. Orbital uses the following four-digit nomenclature to identify Taurus configurations.

Example: **Taurus 2210**

First Digit — **Vehicle configuration**

1 = SSLV Taurus with Peacekeeper Stage 0
(has also been referred to as DARPA Taurus)

2 = Commercial Taurus with Castor 120 Stage 0

3 = Taurus XL with Castor120 and extended length Stage 1 and Stage 2

Second Digit — **Fairing size**

1 = 63-in. (1.6-m) fairing

2 = 92-in. (2.3-m) fairing

Third Digit — **Stage 3 motor**

1 = Orion 38

2 = Star 37

Fourth Digit — **Stage 4 motor**

0 = None

3 = Star 37

Because the Orion motors were already numbered stages 1, 2, and 3 in their application on Pegasus, Orbital designated the Castor 120 first stage as Stage 0 on Taurus, allowing the upper-stage motors to retain the same designation. Thus, "Stage 1" is actually the second stage, and so on.

Cost

According to Orbital, launch services on Taurus are typically priced between $25 and $47 million, depending on the configuration and contract terms and conditions. The original SSLV Taurus configuration was developed with funding from a $16.3 million fixed price contract from DARPA for the first launch and approximately $25 million from Orbital.

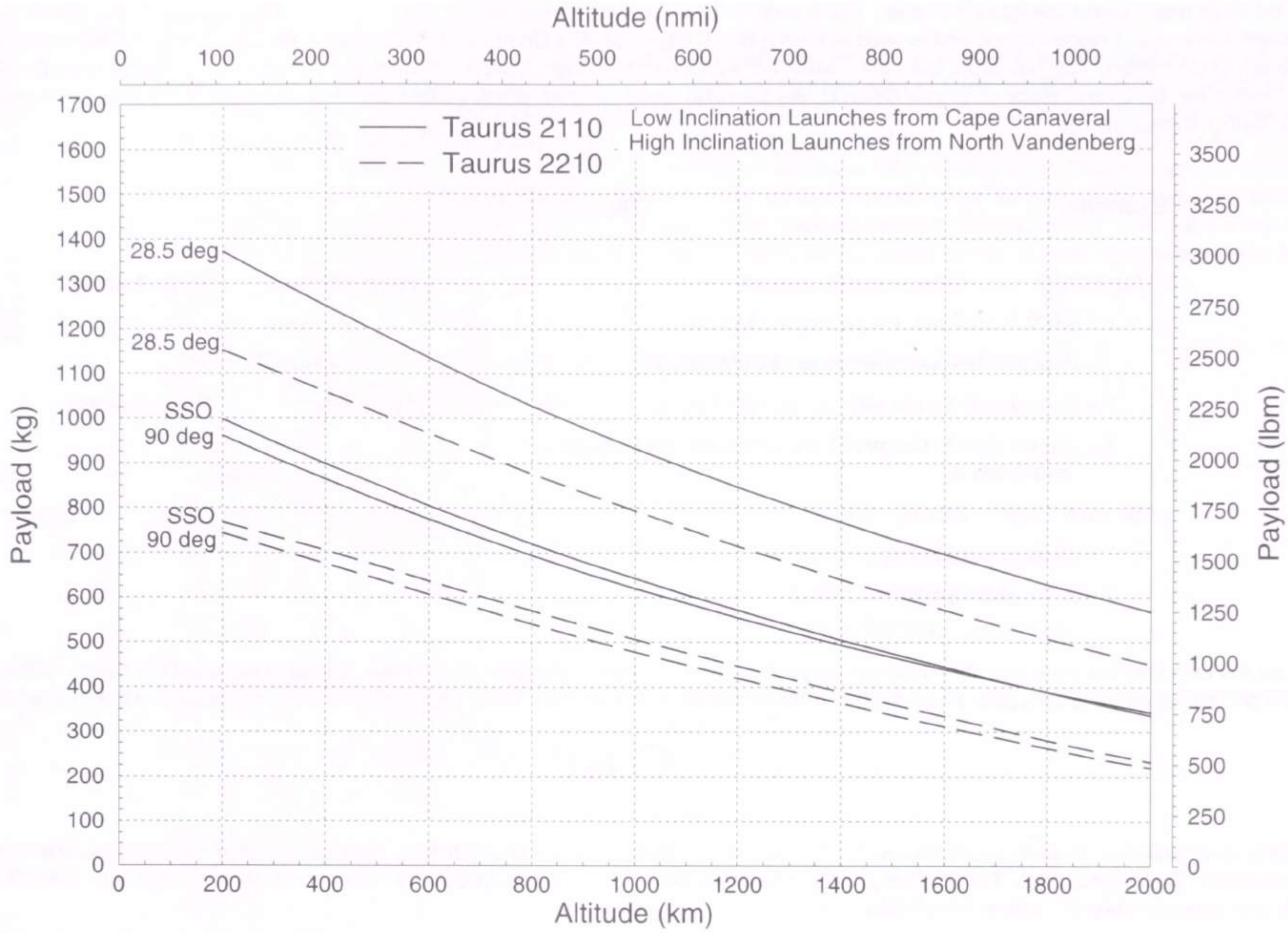
Availability

Taurus is currently in production to support a variety of U.S. government and commercial payloads. The SSLV Taurus is available only for U.S. government payloads because it uses a surplus missile motor. The commercial Taurus configurations are available for all potential customers. Taurus launch services are available directly from Orbital. In addition, options are available to U.S. government customers under the NASA Small Expendable Launch Vehicle Services (SELVS) contract. At the time this publication was going to press, the Taurus XL configuration achieved its intial launch on 20 May 2004. Orbital has not yet determined when or if the standard Commercial Taurus configuration would be phased out.

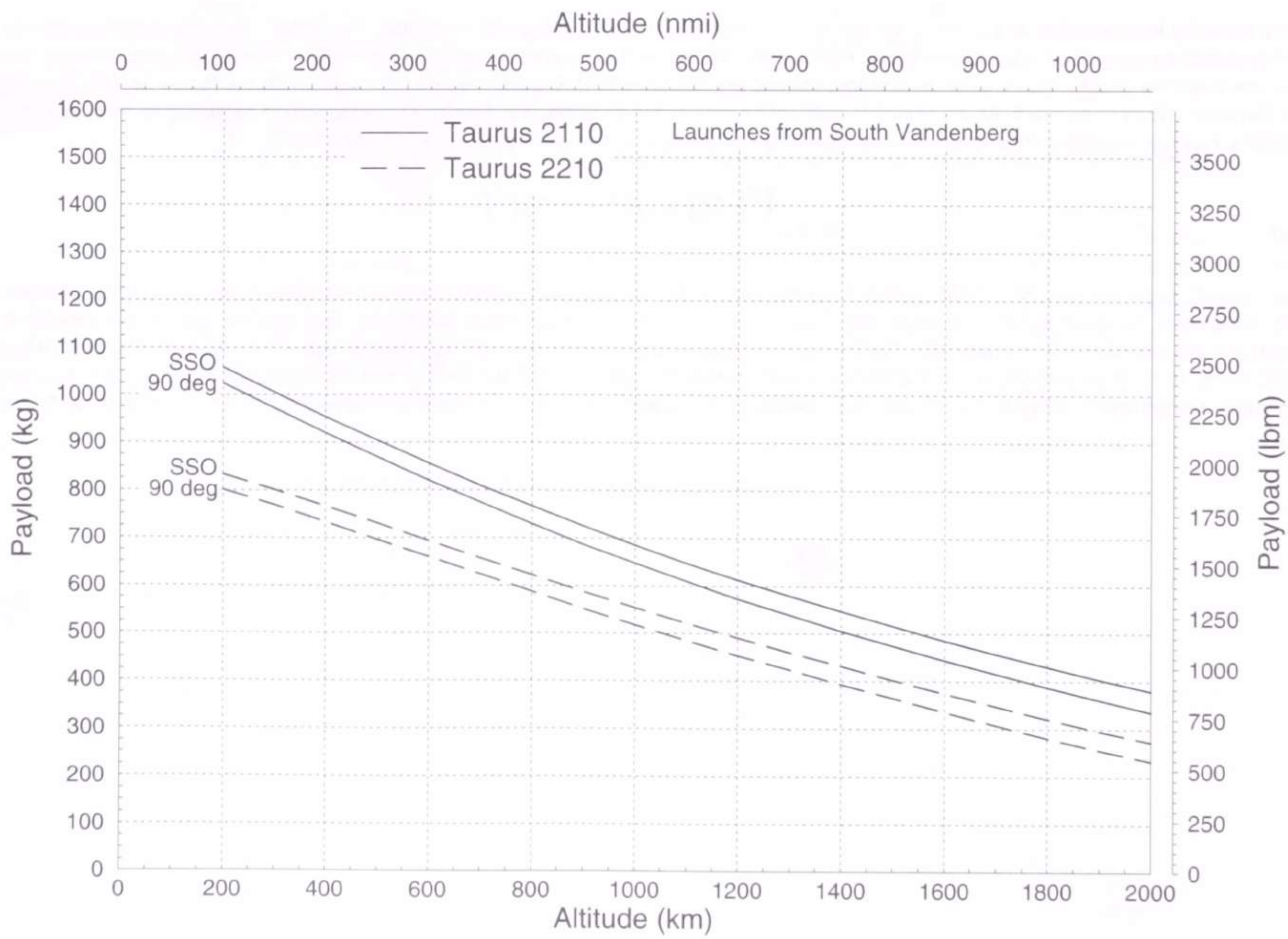
Performance

From its current launch complex SLC-576E on North Vandenberg AFB, Taurus cannot launch due south because it would overfly facilities on the southern part of the base. To reach sun-synchronous and lower inclination orbits, Taurus must fly southwest, then perform a dogleg maneuver to the south at the cost of a modest performance reduction. For this reason, performance to sun-synchronous orbits is higher than 90-deg polar orbits when launched from SLC-576E. Performance is shown for the standard Commercial Taurus both from its current North Base launch site, and from a proposed South Base location. Performance is higher from South Base because the dogleg is not required. Performance is indicated for both payload fairing sizes.

Performance

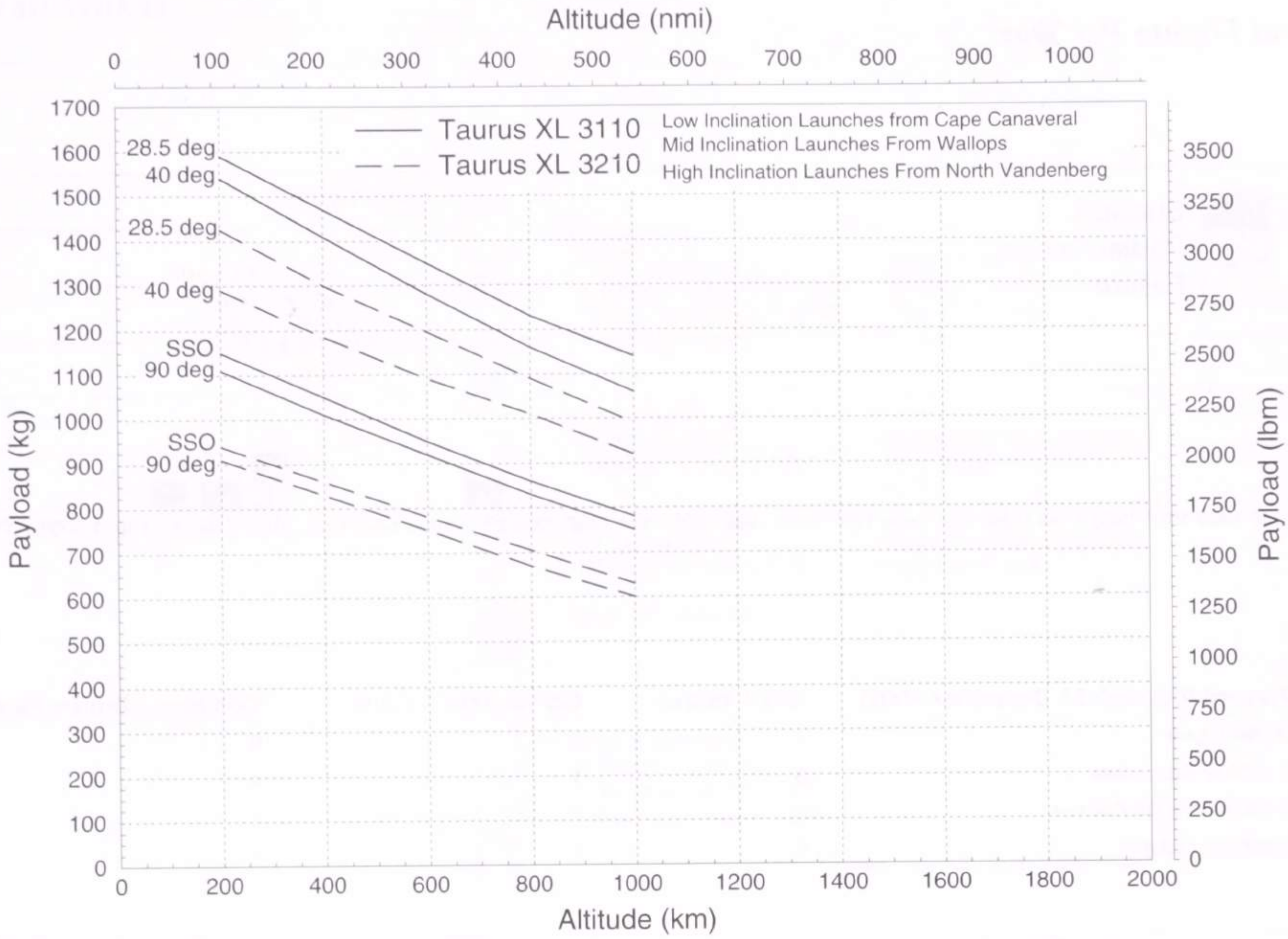


Taurus: Performance to LEO

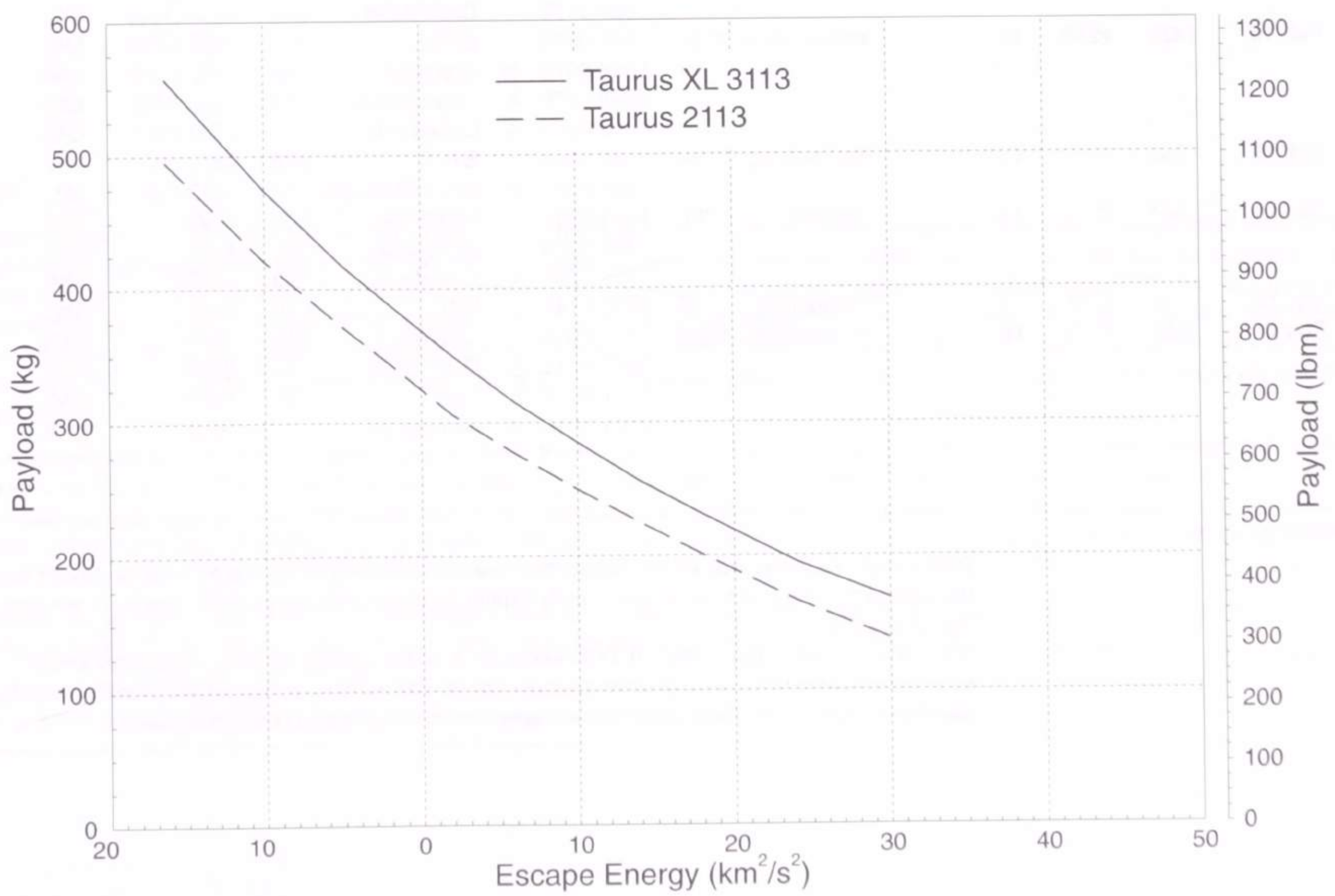


Taurus: Performance to LEO Using South Vandenberg Launch Site

PERFORMANCE



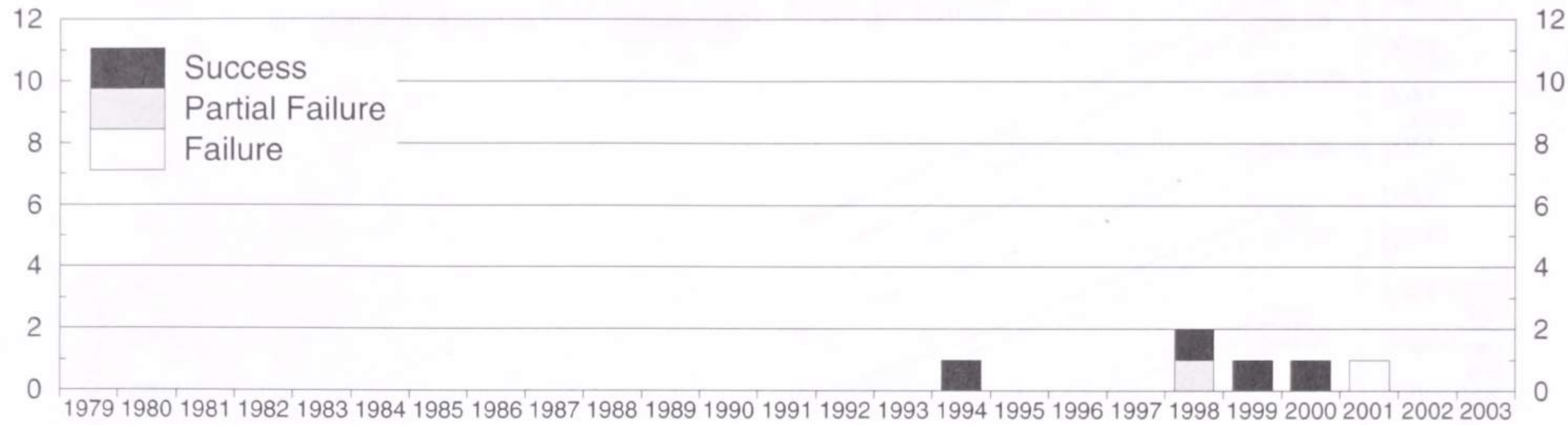
Taurus XL: Performance to LEO



Taurus: Performance for Earth Escape Missions

Performance

Orbital Flights Per Year



Flight Record (through 31 December 2002)	SSLV Taurus	Commercial Taurus	Combined Taurus Family
Total Orbital Flights	3	3	6
Launch Vehicle Successes	3	1	4
Launch Vehicle Partial Failures	0	1	1
Launch Vehicle Failures	0	1	1

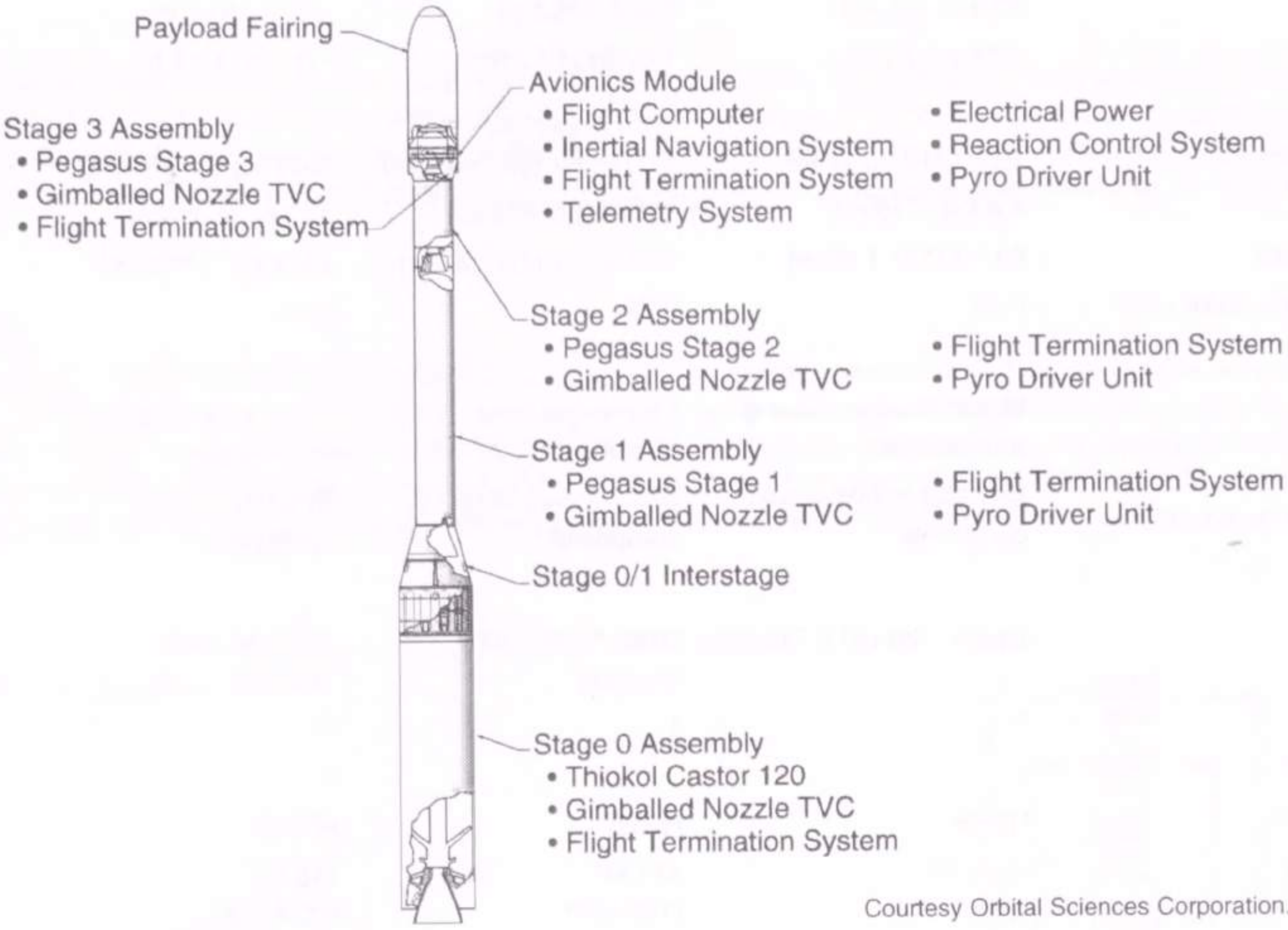
		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	1	1994 Mar 13	—	1110	T1	Vandenberg 576E	1994 017A	STEP 0	502	LEO (105)	MIL	USA
							1994 017B	DARPASat	203	LEO (105)	MIL	USA
P	**2**	**1998 Feb 10**	**1430**	**2210**	**T2**	**Vandenberg 576E**	1998 007A	GFO	367	LEO (108)	MIL	USA
							1998 007B M	Orbcomm 1	43	LEO (108)	CML	USA
							1998 007C M	Orbcomm 2	43	LEO (108)	CML	USA
							1998 007D A	Celestis 02	1	LEO (108)	CML	USA
	3	Oct 03	235	1110	T3	Vandenberg 576E	1998 055A	STEX	678	LEO (85)	MIL	USA
							1998 055C A	ATEx Endmass	13	LEO (85)	MIL	USA
	4	1999 Dec 21	444	2110	T4	Vandenberg 576E	1999 070A	KOMPSat	500	SSO	CIV	South Korea
							1999 070B A	ACRIMSat	114	SSO	CIV	USA
							1999 070C A	Celestis 3	0.9	SSO	CML	USA
	5	2000 Mar 12	82	1110	T5	Vandenberg 576E	2000 014A	MTI	610	SSO	MIL	USA
F	**6**	**2001 Sep 21**	**558**	**2110**	**T6**	**Vandenberg 576E**	2001 F01A	OrbView 4	368	SSO	CML	USA
							2001 F01B A	QuikTOMS	168	SSO	CIV	USA
							2001 F01C A	SBD	73	SSO	CML	USA
							2001 F01C A	Celestis 04		SSO	CML	USA

Failure Descriptions:

P	1998 Feb 10	T2	1998 007	Delivery orbit apogee was 91 km higher than planned. Although considered a partial failure according the definition used in this publication, both Orbital Sciences and the payload customer consider the launch a success.
F	2001 Sep 21	T6	2001 F01	When the second stage ignited at T+83 seconds, a nozzle gimbal actuator drive shaft seized for approximately 5 seconds causing loss of control. The vehicle recovered and continued to fly the mission profile, but failed to reach a stable orbit and reentered near Madagascar.

Vehicle Design

Overall Vehicle



Courtesy Orbital Sciences Corporation.

	Taurus
Height	27.9 m (91.4 ft)
Gross Liftoff Mass	73 t (161 klbm)

Stages

The Taurus first stage is called Stage 0 so that upper stages retain the same numbers as on Pegasus. The SSLV configuration uses a government-furnished Peacekeeper ICBM first-stage motor. Because it is used in an actively deployed missile, data on this motor are not available. The commercial Taurus uses the ATK Thiokol Castor 120 SRM for its first stage. The Castor 120 is a derivative of the Peacekeeper ICBM first-stage motor, designed for space launch applications. It has the same diameter as the Peacekeeper, but is about 0.5 m (1.7 ft) longer to include more propellant. A slight change to the propellant formulation results in a longer burn time that is more suitable for space launch vehicles. The casing is filament-wound graphite–epoxy composite, and TVC is provided by a new blowdown hydraulic nozzle gimbal system similar to those developed for the Castor IVB and SSLV Taurus first stage.

The Taurus upper stages (known as Stages 1, 2, and 3) are the ATK Thiokol Orion 50SG, 50, and 38 SRMs, respectively. These motors were originally developed for Orbital's Pegasus launch vehicle and are fully flight qualified. They have been adapted for use on the both the SSLV and Commercial Taurus configurations, which use the same first, second, and third stages. Common design features, materials, and production techniques are applied to all three motors to maximize reliability and production efficiency. The first stage has an added nozzle gimbal system for thrust vector control. The nozzle of the equivalent Pegasus stage was fixed because attitude control is provided aerodynamically.

In the Taurus XL configuration, the Stage 1 and Stage 2 (that is, the second and third stage) motors are replaced by the extended length Orion 50S-XL and Orion 50-XL. These motors have 24% and 30% more propellant, respectively, than the original motors.

An ATK Thiokol Star 37FM kick motor can be used in place of, or in addition to, the standard Orion 38 as an optional for high energy missions, such as GTO or Earth escape trajectories. The Star 37FM is a spin-stabilized motor with about 1 t of propellant.

Vehicle Design

The following data describe the standard Commercial Taurus configuration. For details on the XL motors used on Stages 1 and 2 of the Taurus XL, see Pegasus chapter. Note that masses shown are for the motors only. Orbital has not provided data on the full-stage gross masses.

	Stage 0	Stage 1	Stage 2	Stage 3
Dimensions				
Length	12.8 m (41.9 ft)	8.6 m (28.3 ft)	3.1 m (10.1 ft)	1.3 m (4.4 ft)
Diameter	2.35 m (7.7 ft)	1.27 m (4.17 ft)	1.27 m (4.17 ft)	1.0 m (3.17 ft)
Mass				
Propellant Mass	48.7 t (107.4 klbm)	12,154 kg (26,800 lbm)	3027 kg (6674 lbm)	771 kg (1700 lbm)
Inert Mass	4.4 t (9.7 klbm)	1088 kg (2400 lbm)	352 kg (776 lbm)	104 kg (230 lbm)
Loaded Motor Gross Mass	53.1 t (117.1 klbm)	13,242 kg (29,200 lbm)	3379 kg (7450 lbm)	875 kg (1930 lbm)
Propellant Mass Fraction (motor only)	0.92	0.92	0.90	0.88
Structure				
Type	Motor: filament-wound monocoque	Filament-wound monocoque	Filament-wound monocoque	Filament-wound monocoque
Material	Motor: graphite–epoxy composite	Graphite–epoxy composite	Graphite–epoxy composite	Graphite–epoxy composite
Propulsion				
Engine Designation	Castor 120 (ATK Thiokol)	Orion 50SG (ATK Thiokol)	Orion 50 (ATK Thiokol)	Orion 38 (ATK Thiokol)
Number of Motors	1	1	1	1
Number of Segments	1	1	1	1
Propellant	HTPB	HPTB	HTPB	HTPB
Average Vacuum Thrust	1615 kN (363.7 klbf)	471 kN (106 klbf)	115 kN (25.9 klbf)	32 kN (7.2 klbf)
Vacuum Isp	277.9 s	285.0 s	290.2 s	286.7 s
Chamber Pressure	96.5 bar (1400 psi)	58.4 bar (847 psi)	58.3 bar (845 psi)	37.8 bar (549 psi)
Nozzle Expansion Ratio	17:1	40:1	65:1	60:1
Attitude Control				
Pitch, Yaw	Hydraulically gimbaled nozzle ±5 deg	Electromechanically gimbaled nozzle	Electromechanically gimbaled nozzle	Electromechanically gimbaled nozzle
Roll	None	Nitrogen cold-gas RCS	Nitrogen cold-gas RCS	Nitrogen cold-gas RCS
Staging				
Nominal Burn Time	82.5 s	72.4 s	75.1 s	68.5 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Burn to depletion
Stage Separation	Linear shaped charge	Linear shaped charge	Frangible Joint	Clampband

Vehicle Design

Attitude Control System

Gimbaled nozzles on all four motors provide pitch and yaw control during powered flight. The actuation system varies—Stage 0 uses a blowdown hydraulic system while Stages 1, 2, and 3 use electromechanical actuators. Three-axis control during coast periods, as well as roll control during Stage 1, 2, and 3 burn, are provided by a regulated cold-gas attitude control system located on the Taurus avionics shelf.

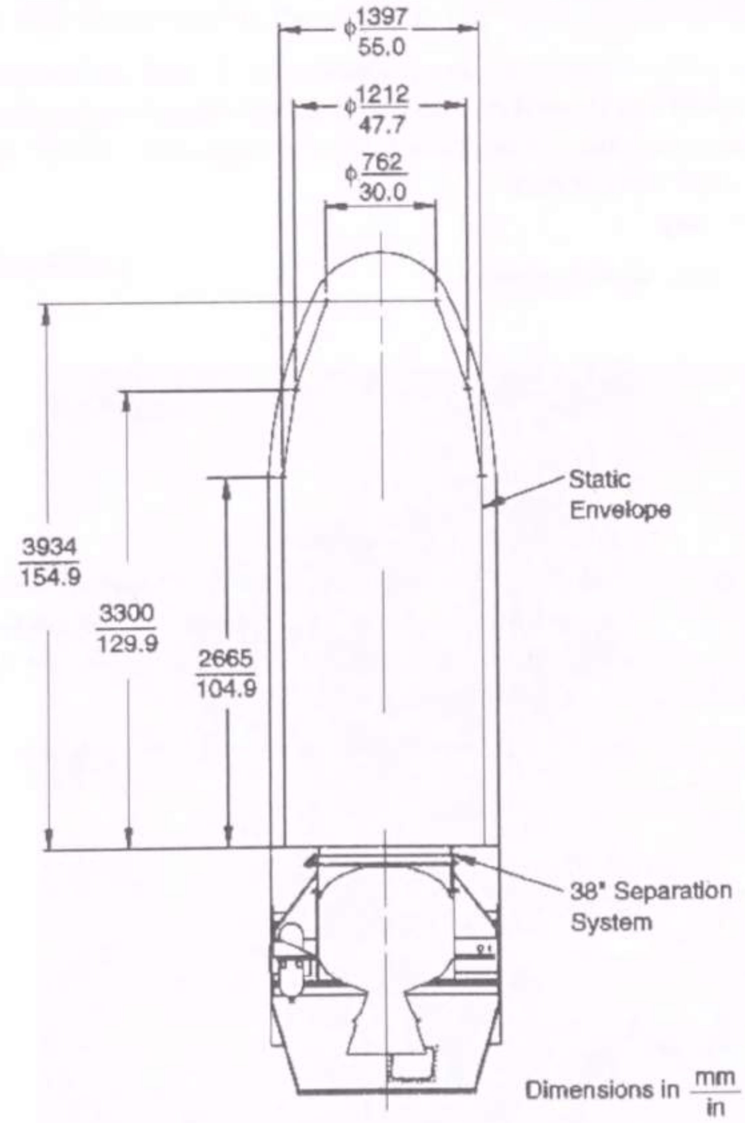
Avionics

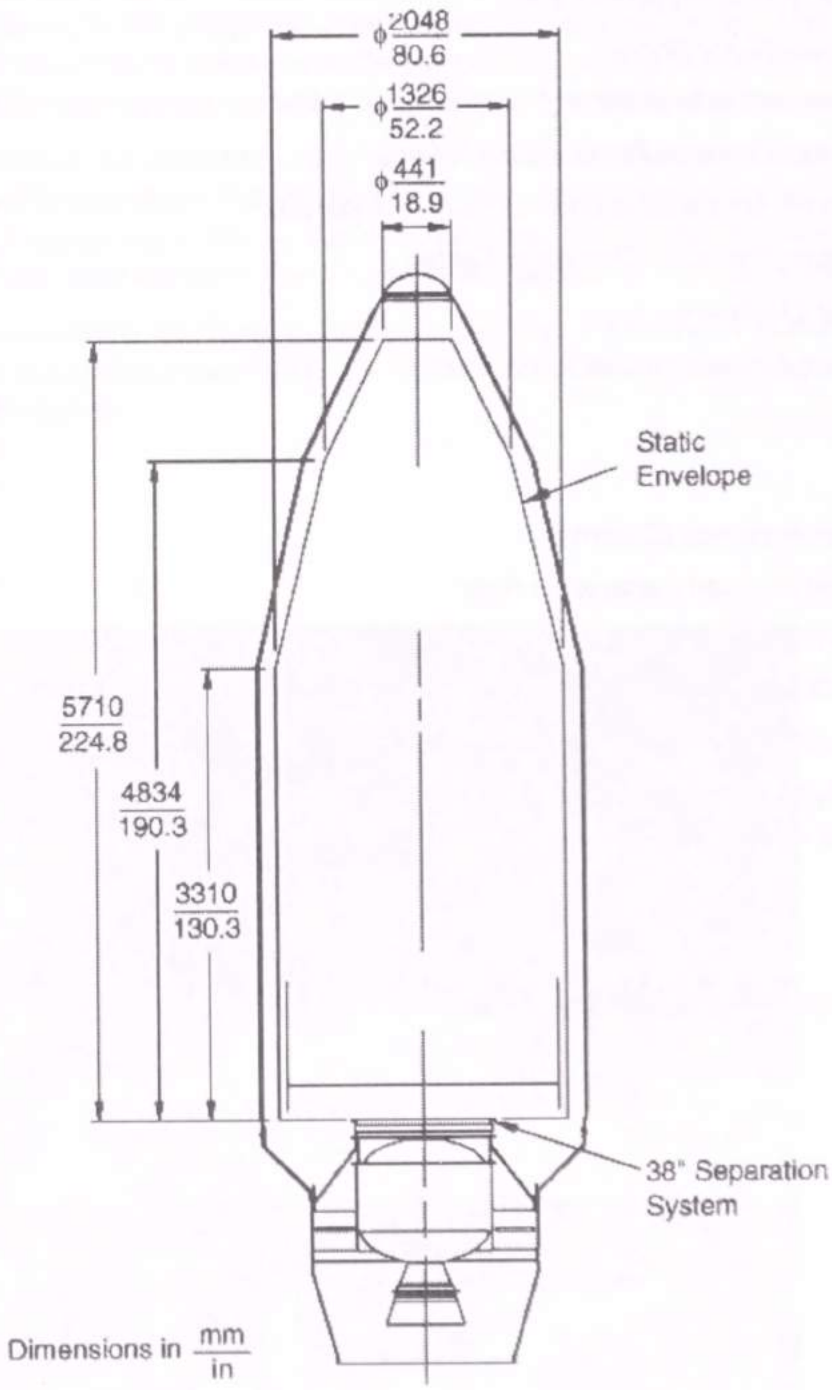
The Taurus avionics system is an all-digital distributed-processor design. Mission reliability is achieved by the use of simple designs, high-reliability components, high design margins and extensive testing at the subsystem and system level. The heart of the avionics system is a multiprocessor, 32-bit flight computer that communicates with all vehicle subsystems and the launch support equipment using standard RS-422 digital serial links. All avionics on the vehicle feature integral microprocessors to perform local processing and to handle communications with the flight computer.

Payload Fairing

Orbital offers two flight-proven payload fairings to encapsulate the payload and provide protection and contamination control during ground handling, integration operations, and flight. Both fairings are bisector shells constructed of graphite–epoxy facesheets with an aluminum honeycomb core. Two payload accommodation structures are available for comanifesting or secondary payloads. Both are mounted directly to the upper stage, enclosing the lower payload, and can be tailored in height to provide the appropriate volume. The Aft Payload Capsule (APC) is 1270 mm (50 in.) in diameter, and up to 1824 mm (71.8 in.) in length. The Dual Payload Attach Fitting (DPAF) has a diameter of 1600 mm (63 in.), and can be up to 2285 mm (90 in.) long.

Courtesy Orbital Sciences Corporation.





	1.6 m (63 in.) Orion 38 Configuration	**2.3 m (92 in.) Orion 38 Configuration**
Length	5.5 m (18 ft)	7 m (23 ft)
Primary Diameter	1.6 m (63 in.)	2.3 m (92 in.)
Mass	360 kg (800 lbm)	400 kg (900 lbm)
Sections	2	2
Structure	Honeycomb	Honeycomb
Material	Graphite–epoxy facesheets over aluminum honeycomb	Graphite–epoxy facesheets over aluminum honeycomb

Payload Accommodations

	Commercial Taurus
Payload Compartment	
Maximum Payload Diameter	63-in. fairing: 1397 mm (55.0 in.) 92-in. fairing: 2048 mm (80.6 in.)
Maximum Cylinder Length	63-in. fairing: 2665 mm (104.9 in.) 92-in. fairing: 3310 mm (130.3 in.)
Maximum Cone Length	63-in. fairing: 1269 mm (50.0 in.) 92-in. fairing: 2400 mm (94.5 in.)
Payload Adapter Interface Diameter	986 mm or 944 mm (38.81 in. or 37.15 in.)
Payload Integration	
Nominal Mission Schedule Begins	T–24 months
Launch Window	
Last Countdown Hold Not Requiring Recycling	T–8 min
On-Pad Storage Capability	30 days
Last Access to Payload	Varies
Environment	
Maximum Axial Limit Load (Payload CG)	–8 *g*
Maximum Lateral Limit Load (Payload CG)	±2.5 *g*
Minimum First Bending Frequency Recommended	25 Hz
Overall Sound Pressure Level	136 dB (63-in. fairing), 138 dB (92-in. fairing)
Maximum Flight Shock	3500 *g* from 1300–10000 Hz
Cleanliness Level in Fairing	Can support Class 10,000
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 $BTU/ft^2/s$)
Maximum Pressure Change in Fairing	3.5 kPa/s (0.5 psi/s)
Payload Delivery	
Injection Accuracy to LEO (3 sigma)	Injection Apse: ±9.26 km (5 nmi) Non-Injection Apse: ±50 km (27 nmi) Mean Altitude: ±29.6 km (16 nmi) Inclination: ±0.15 deg
Attitude Accuracy (3 sigma)	Pitch, Yaw: ±0.7 deg, ±0.4 deg/sec
Nominal Payload Separation Rate	Mission specific
Deployment Rotation Rate Available	Mission specific
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	Yes
Multiple/Auxiliary Payloads	
Multiple or Comanifest	A dual payload attach fitting (DPAF) is available to carry comanifested payloads. The lower payload is contained within a canister structure that supports the upper payload. After separation of the first payload, the upper surface of the DPAF is jettisoned to deploy the second payload.
Auxiliary Payloads	Flight opportunities are available. Contact Orbital for more information.

Production and Launch Operations

Production

Orbital Sciences Corporation is the primary contractor for the Taurus launch vehicle.

Taurus Subcontractors	Responsibility
ATK Thiokol	Stage 1–3 rocket motors, Castor 120 Stage 0 motor
Vermont Composites	63-in. fairing, boattail, avionics skirt, and payload cone
R-Cubed	92-in. fairing
Remmele	Stage 0–1 Interstage
Parker	Stage 2 and 3 TVC
AlliedSignal	Stage 0 and 1 TVC
Litton	IMU
SBS Embedded Computers	flight computer

Launch Operations

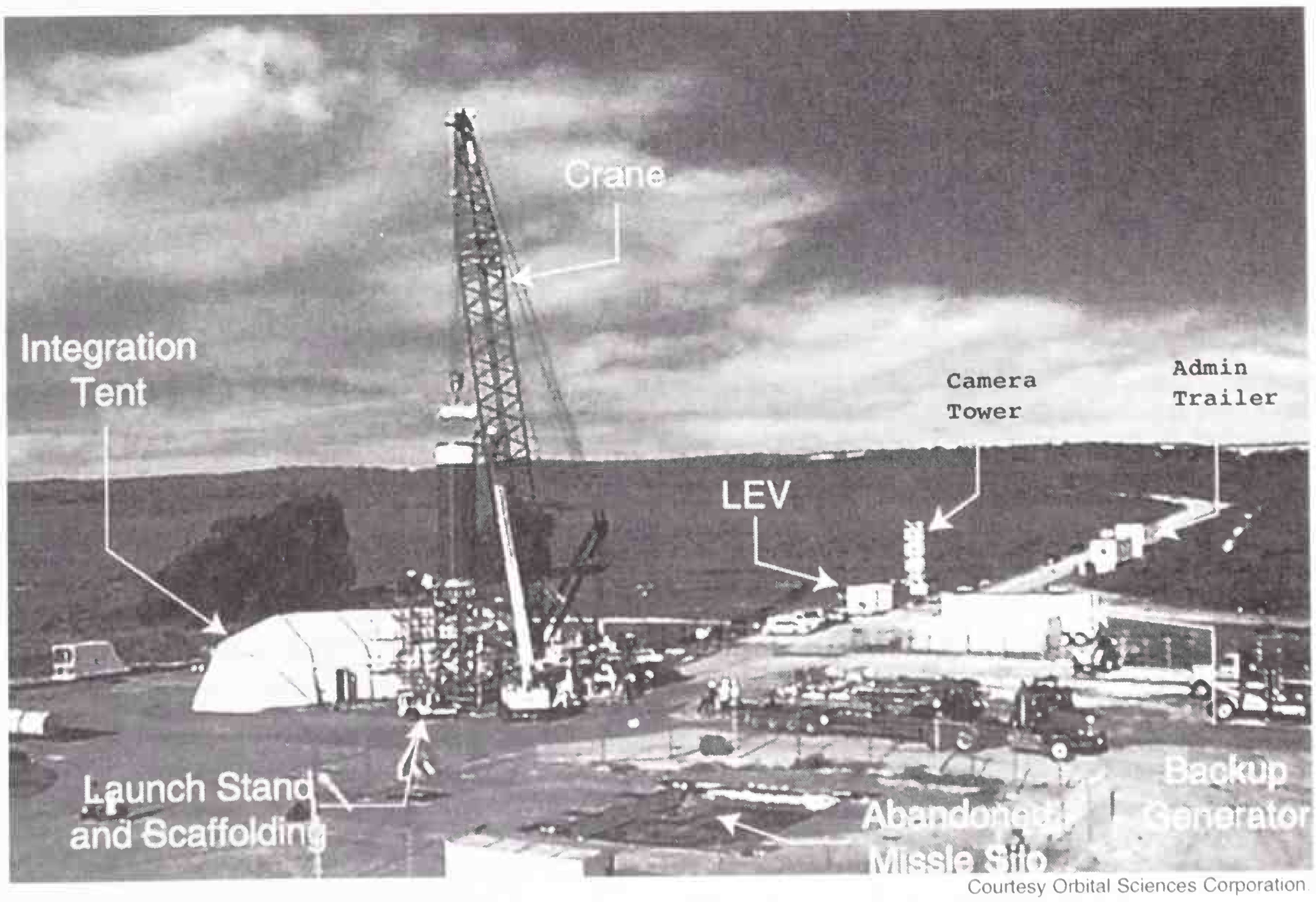
The Taurus system has been designed to minimize vehicle and payload handling complexity and launch base operations time. Ground and launch operations are conducted by Orbital in three major phases: 1) launch vehicle integration, 2) payload processing, and 3) launch operations.

During the launch vehicle integration phase, all Taurus vehicle motors, parts, and completed subassemblies are delivered to the missile assembly building (MAB) on North VAFB from either the assembly vendor or Orbital's Chandler, Arizona, production facility. The transformation of solid rocket motors, avionics, and subassembly structures into an integrated and tested launch vehicle occurs at the MAB. Following the final MAB system test, the vehicle consists of five major assemblies: the Stage 0 motor assembly, the Stage 0–1 interstage, the integrated upper stages, the payload fairing, and the payload cone. At this point, the fairing and payload cone assemblies are released and prepared for shipment to the payload processing facility (PPF) for payload encapsulation operations. The Stage 0 motor, the Stage 0–1 interstage, and the upper-stage assembly are prepared for shipment to the launch site.

Payload processing is conducted independently of Taurus vehicle processing, allowing the payload to determine and control the length of time required for checkout within the PPF. Once the payload has completed its own independent verification and checkout, Orbital delivers the fairing and payload cone assemblies to the PPF. The nominal integrated operations include mating the payload to the Taurus payload cone, performing interface verification tests, and encapsulating the payload within the Taurus fairing. Orbital personnel then transport the encapsulated payload to the launch site.

Launch operations consist of installation and verification of the launch support equipment, facilities, and utilities; delivery of the Taurus launch vehicle components to the launch site for final integration and test; transportation of the encapsulated payload to the launch site for testing and integration with the Taurus vehicle; final system testing and vehicle mate; and vehicle arming and launch.

Launch Facilities

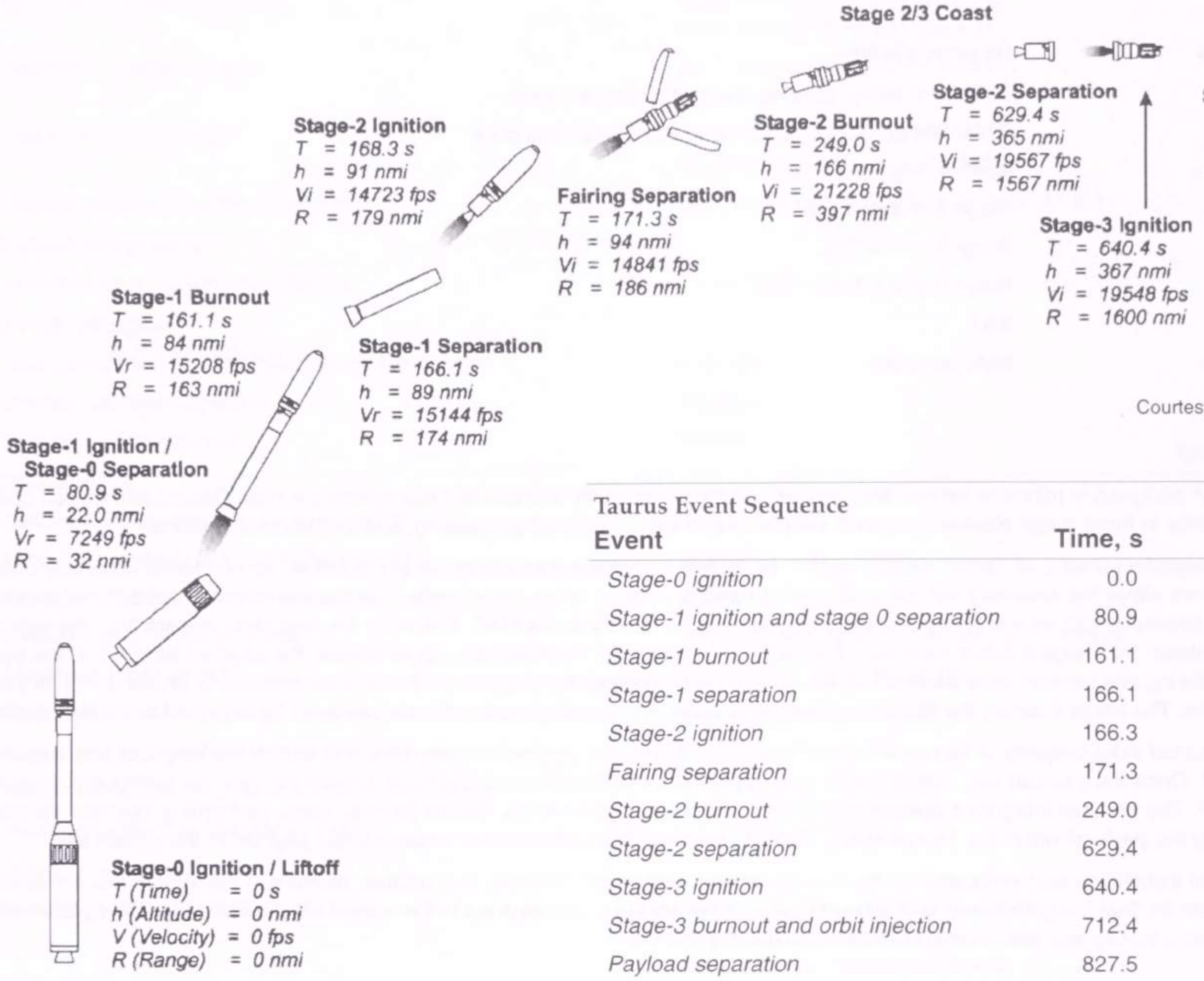


Courtesy Orbital Sciences Corporation

Vandenberg AFB Launch Facilities

Production and Launch Operations

Flight Sequence



Courtesy Orbital Sciences Corporation.

Taurus Mission Profile

Taurus Event Sequence

Event	Time, s	Altitude, km
Stage-0 ignition	0.0	0
Stage-1 ignition and stage 0 separation	80.9	41
Stage-1 burnout	161.1	156
Stage-1 separation	166.1	165
Stage-2 ignition	166.3	169
Fairing separation	171.3	174
Stage-2 burnout	249.0	307
Stage-2 separation	629.4	676
Stage-3 ignition	640.4	680
Stage-3 burnout and orbit injection	712.4	692
Payload separation	827.5	

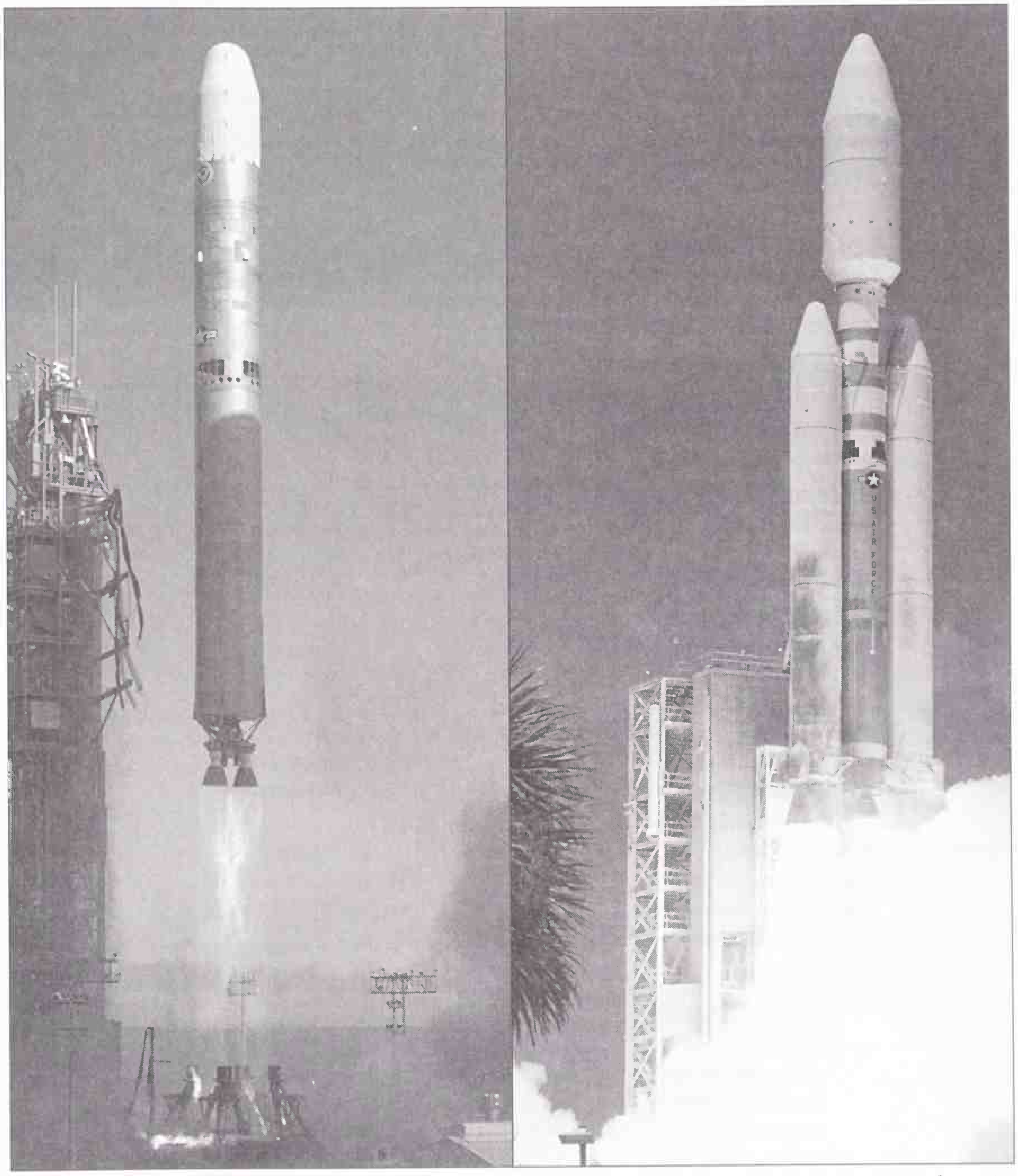
Vehicle History

Historical Summary

The Taurus launch vehicle was privately developed by Orbital to provide a cost-effective, reliable, and flexible means of placing small satellites into LEO. Taurus was originally designed to meet the needs of DARPA's Standard Small Launch Vehicle (SSLV) Program. The requirements of the program stressed system reliability, transportability, and operations from remote launch sites. Following the successful Taurus initial flight, Orbital initiated development work on an upgraded version of the Taurus vehicle. This configuration, which is the version of the Taurus vehicle offered commercially, differs from the SSLV configuration primarily through the use of ATK Thiokol's Castor 120 SRM for the Stage 0 in place of the Peacekeeper stage. This configuration was first launched in 1998.

Taurus's sibling program, the Pegasus, was upgraded to the Pegasus XL configuration with longer motors in 1994. Orbital began studying the equivalent upgrade for Taurus at the same time, but received no orders for the enhanced vehicle. Finally, in 2001, the National Space Program Office of the Republic of China signed a contract for the launch of ROCSAT 2 using a Taurus XL, accomplished in 2004.

TITAN



Courtesy Lockheed Martin Corporation.

The Titan II (left) is a converted ICBM used by the U.S. Air Force to launch small spacecraft. The Titan IVB (right) is the largest expendable launch vehicle in the United States and is used by the U.S. Air Force to launch heavy military spacecraft to LEO and GEO.

Contact Information

Air Force Point of Contact:
Titan System Program Office (ME)
U.S. Air Force Space and Missile Center
2400 El Segundo Blvd.
El Segundo, CA 90245
USA
Phone: +1 (310) 363-1110
Web site: www.laafb.af.mil/SMC/CL

Industry Point of Contact:
Lockheed Martin Space Systems
P.O. Box 179
Denver, CO 80201
USA
Phone: +1 (303) 977-3000
Web site: www.ast.lmco.com/launch_titan.shtml

Titan II 23G

Titan II 23G

General Description

Summary

Titan II launch vehicles are decommissioned ICBMs that have been refurbished and equipped with hardware required for space launch. The Titan II is used by the U.S. Air Force for launches of small U.S. government payloads to low polar orbits. Titan II launch services are not commercially available.

Status

Operational. First launch 1988.

Origin

United States

Key Organizations

Marketing Organization	Not marketed
Launch Service Provider	U.S. Air Force
Prime Contractor	Lockheed Martin Space Systems

Primary Missions

Small U.S. government payloads to polar LEO

Estimated Launch Cost

$30–40 million (FAA, 1999)

Spaceport

Launch Site	Vandenberg AFB SLC-4W
Location	34.7° N, 120.6° N
Available Inclinations	63.4–100 deg

Performance Summary

185 km (100 nmi), 28.5 deg	No capability
185 km (100 nmi), 90 deg	1900 kg (4200 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	No capability
Sun-Synchronous Orbit: 260×800 km (140×432 nmi), 98.6 deg	1100 kg (2425 lbm)
GTO	No capability
Geostationary Orbit	No capability

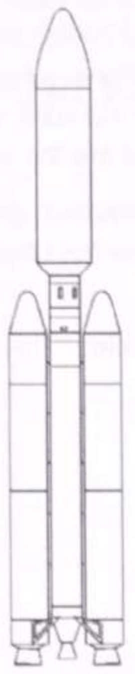
Flight Record (through 31 December 2003)

Total Orbital Flights	13
Launch Vehicle Successes	13
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	0

Flight Rate

0–2 per year

Titan IVB



Titan IVB

General Description

Summary

In 1985 the U.S. Air Force initiated the Titan IV program to upgrade the Titan 34D into a larger vehicle that could provide an alternate means for launching Shuttle-class payloads. Following the *Challenger* accident, the Titan IV became the primary launch vehicle for launching large U.S. Air Force satellites and intelligence payloads for the U.S. National Reconnaissance Office. Beginning in 1997, the original Titan IVA was phased out and replaced by the new Titan IVB configuration. The Titan IVB offers roughly 25% greater performance from new solid strap-on boosters. In addition, the Titan IVB has modernized avionics and a standardized core configuration and interfaces. Titan IV can be launched with no upper stage (NUS) for LEO payloads or with an Inertial Upper Stage (IUS) or Centaur upper stage to deliver payloads directly to GEO. Using the Centaur upper stage, the Titan IVB has higher payload capability than any other active expendable launch vehicle in the world.

Status

Operational. First launch 1997.

Origin

United States

Key Organizations

Marketing Organization	Not marketed
Launch Service Provider	U.S. Air Force
Prime Contractor	Lockheed Martin Space Systems

Primary Missions

Heavy LEO or GEO military payloads

Estimated Launch Cost

$350–450 million (FAA, 1999)

Spaceports

Launch Site	Vandenberg AFB SLC-4E
Location	34.7° N, 120.6° N
Available Inclinations	63.4–113 deg directly, as low as 55 deg with dogleg or in-orbit plane change

Launch Site	Cape Canaveral AFS LC-40
Location	28.5° N, 81.0° W
Available Inclinations	28.5–55 deg directly, up to 65 deg with in-orbit plane change

Performance Summary

Titan IVB performance data are not available for nonstandard reference orbits. For further information, contact the U.S. Air Force program office.

150×175 km (80×95 nmi), 28.6 deg	21,680 kg (47,800 lbm)
185 km (100 nmi), 90 deg	17,600 kg (38,800 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	Nonstandard mission
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	Nonstandard mission
GTO: 28.5 deg 185×35,786 km (100×19,323 nmi)	Nonstandard mission
Geostationary Orbit	IUS: 2380 kg (5250 lbm) Centaur: 5760 kg (12,700 lbm)

Flight Record (through 31 December 2003)

Total Orbital Flights	14
Launch Vehicle Successes	13
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	1

Flight Rate

1–3 per year

Nomenclature

Several distinct naming systems are used to distinguish Titan configurations. The basic nomenclature uses a Roman numeral and a letter, such as Titan IIID or Titan IVB. The Roman numeral indicates the vehicle family, while the letter distinguishes between versions within a family in the approximate order that they were introduced. More subtle distinctions can be indicated with a pair of numerals reflecting the variations of the second and first core stages. For example, the Titan IIIB designation covered Titan versions named 23B, 24B, 33B, and 34B. This nomenclature is generally used only when differences in the core stages exist within a vehicle type. For example, these designations exist for the variations of the Titan IV, but are rarely used because the core stages are the same.

The Titan IV has flown in two basic configurations. The older Titan IVA used two seven-segment SRMs and used avionics based on previous generations of the Titan. The current Titan IVB has updated avionics and uses two improved SRMU strap-on boosters. The A and B names were not used until around the time of the first Titan IVB launch. Previously the two configurations were simply called the Type I and Type II Titan IV.

A three digit number is used to distinguish the configuration of the forward skirts, upper stage, and payload interfaces for Titan IV vehicles. This code does not distinguish between the Titan IVA and Titan IVB vehicles.

401	45E	Centaur upper-stage configuration
402	45D	IUS configuration
403	45F	No upper stage (NUS) configuration for Vandenberg AFB launch
404	45J	NUS configuration for Vandenberg AFB launch
405	45H	NUS configuration for Cape Canaveral AS launch

In the official DoD nomenclature system for aerospace vehicles, Titan II is designated SB-4A and Titan IV is designated SB-5A. These indicate that they are the fourth and fifth design series in the Space Booster category. Although the equivalent designations are widely used for aircraft and missiles (e.g. F-16D or AIM-9X), in practice these names are rarely used for space launch systems.

Cost

According to Lockheed Martin, the total contract value for the Titan II program is $660 million for refurbishment and launch of 14 vehicles. Based on open source information, the FAA estimates the per-flight cost of the Titan II as $30–40 million. The total Titan IV contract is valued at $15.8 billion, including production of 40 launch vehicles and required upper stages, as well as launch services for 39 of the vehicles (the remaining vehicle is a spare). The FAA estimates the per-flight costs of the Titan IVB at $350–450 million. The U.S. Air Force has not announced general Titan IVB cost ranges, but has released costs for certain missions. Flight 4B-27 in April 1999 carried a reported cost of $432 million, including both the Titan and the IUS, which is purchased separately. The 4B-32 Titan/Centaur mission cost $433 million. The Titan IVB/IUS mission 4B-29 cost $432 million, while the same configuration cost $460 million on flight 4B-31. Cost values for USAF launches may or may not include elements such as systems engineering oversight, payload integration, or launch operations, depending on how costs are accounted.

Availability

Both Titan II and Titan IV are owned and operated by the U.S. Air Force and are not available to commercial customers. However, the U.S. Air Force does provide Titan launch services to other U.S. government agencies. For example, some Earth observing satellites for NOAA and NASA are launched on Titan II vehicles. NASA used one Titan IVB to launch the Cassini spacecraft to Saturn, but no further Titan IV launches for NASA are planned.

Titan II space launch vehicles are refurbished ICBMs. When the Titan II missile was withdrawn from military service beginning in 1982, 55 vehicles were placed in storage for possible conversion to space launch vehicles. Under a U.S. Air Force contract, Lockheed Martin has refurbished 14 of these vehicles and converted them into space launch vehicles; 13 have been launched.

The U.S. Air Force has purchased 40 Titan IV launch vehicles from Lockheed Martin. All 22 of the Titan IVA vehicles have been launched. Eighteen Titan IVB vehicles have been built, of which 14 have been launched as of December 2003. The remaining Titan IVB vehicles will continue flying until around 2005, when they will be replaced by the heavy-lift versions of the new Delta IV and Atlas V developed under the U.S. Air Force's Evolved Expendable Launch Vehicle (EELV) program. Payloads have been assigned to all but one Titan IVB vehicle, which will remain as a backup in case of a launch failure or delays in the development of the EELV heavy-lift vehicles.

Performance

The Titan II is launched only from SLC-4W at VAFB and thus cannot be used for low inclination orbits. Inclinations between approximately 63.4 deg and 100 deg can be reached from SLC-4W. The Titan second-stage engine must ignite immediately after first-stage shutdown, and it is not capable of restarting. This prevents the Titan II from reaching circular orbits higher than about 400 km (215 nmi). To deliver payloads to higher circular orbits, Titan II can deploy the payload and a Star 37 solid kick motor on a ballistic trajectory. The solid kick motor fires at apogee to circularize the orbit. Alternatively, the Titan II can carry about 1100 kg (2425 lbm) to a 260×800 km (140×432 nmi) 98.6-deg orbit. From there, the spacecraft can circularize its orbit using onboard propulsion.

The Titan IVB is the largest expendable launch vehicle in the United States. Its performance to LEO is similar to Russia's Proton launch vehicle, but with the high-performance Centaur upper stage the Titan IVB has unmatched capabilities to deliver heavy payloads to high altitudes, including injection directly into GEO. Titan IVB is launched from both CCAFS (LC-40) and VAFB (SLC-4E). From CCAFS, Titan IV can reach inclinations from 28.5 to 55 deg directly. Using the Centaur upper stage to perform an in-orbit plane change, inclinations as high as 65 deg can be reached. From VAFB SLC-4E, inclinations of 63.4–113 deg can be achieved directly, and dogleg maneuvers or in-orbit plane changes can lower the inclination to 55 deg. Either the Centaur or the IUS upper stage can be used to deliver payloads directly to GEO. Because the Titan IVB is not marketed and performs only specific USAF missions, general performance curves have not been created. Instead, performance data for Titan IVB reference missions are included below.

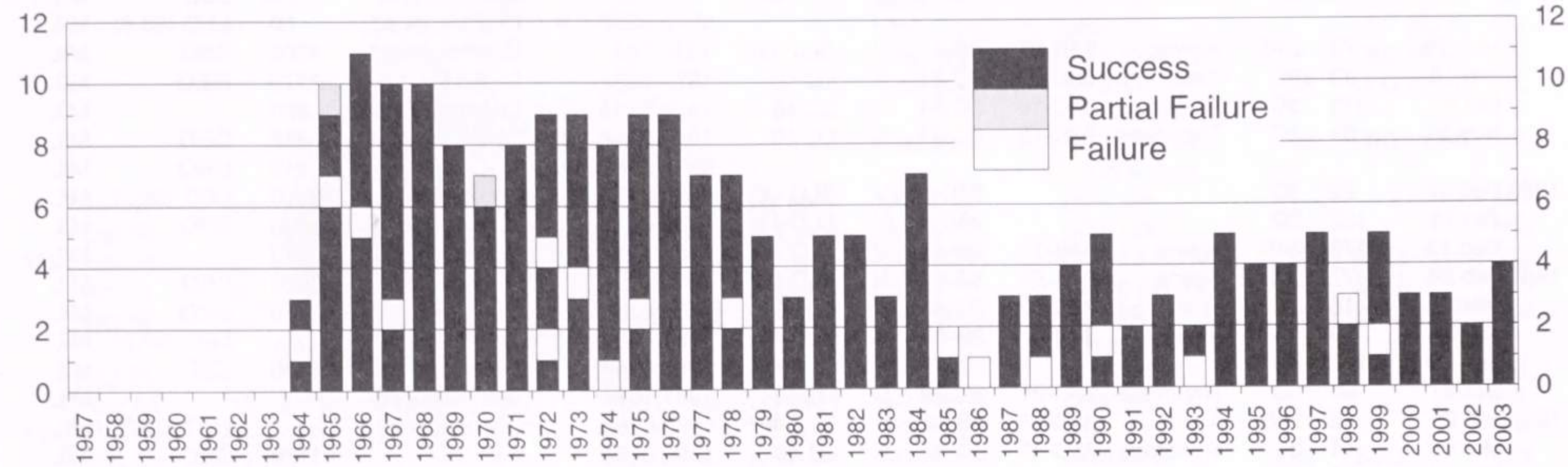
Titan IVB Reference Performance Capabilities

Orbit	Configuration	Capability
150 x 175 km (80×95 nmi) 28.6 deg	405 NUS	21,680 kg (47,800 lbm)
185 km (100 nmi), 90 deg	403 NUS	17,600 kg (38,800 lbm)
Geostationary Orbit	401 (Centaur)	5760 kg (12,700 lbm)

Flight History

In addition to the orbital launches shown below, 81 Titan II ICBMs were launched on suborbital trajectories for development or training. One suborbital Titan II–Gemini launch was performed for NASA in January 1965. Because many Titan payloads are classified, full details are not always available.

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Titan II 23G	Titan IVB	All Titan IV	All Titan
Total Orbital Flights	13	14	36	216
Successes	13	13	33	197
Partial Failures	0	0	0	2
Failures	0	1	3	17

Year	Total T/F	2 Gemini T/F	3A T/F	3B T/F	3C T/F	3D T/F	3E T/F	34D T/F	23G T/F	3 T/F	4A T/F	4B T/F
Total	216/19	11/0	4/1	68/5	36/5	22/0	7/1	15/3	13/0	4/1	22/2	14/1
1964	3/1	1/0	2/1									
1965	10/2	5/0	2/0		3/2							
1966	11/1	5/0		3/0	3/1							
1967	10/1			7/1	3/0							
1968	10/0			8/0	2/0							
1969	8/0			6/0	2/0							
1970	7/1			5/0	2/1							
1971	8/0			5/0	2/0	1/0						
1972	9/2			5/2	1/0	3/0						
1973	9/1			4/1	2/0	3/0						
1974	8/1			3/0	1/0	2/0	2/1					
1975	9/1			3/1	2/0	2/0	2/0					
1976	9/0			4/0	2/0	2/0	1/0					
1977	7/0			2/0	2/0	1/0	2/0					
1978	7/1			2/0	3/1	2/0						
1979	5/0			1/0	3/0	1/0						
1980	3/0			1/0		2/0						
1981	5/0			2/0	2/0	1/0						
1982	5/0			1/0	1/0	2/0		1/0				
1983	3/0			2/0				1/0				
1984	7/0			2/0				5/0				
1985	2/1			1/0				1/1				
1986	1/1							1/1				
1987	3/0			1/0				2/0				
1988	3/1							2/1	1/0			
1989	4/0							2/0	1/0		1/0	
1990	5/1									3/1	2/0	
1991	2/0										2/0	
1992	3/0								1/0	1/0	1/0	
1993	2/1								1/0		1/1	
1994	5/0								1/0		4/0	
1995	4/0										4/0	
1996	4/0										4/0	
1997	5/0								1/0		2/0	2/0
1998	3/1								1/0		1/1	1/0
1999	5/1								2/0			3/1
2000	3/0								1/0			2/0
2001	3/0											3/0
2002	2/0								1/0			1/0
2003	4								2			2

Flight History

		Date (UTC)	Launch Interval (days)	Model	Upper Stage	Vehicle Number	Flight Designation		Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	126	1979 Mar 16	92	3D			23D-21	V	SLC 4E	1979 025A		Unannounced	13300	SSO	MIL	USA
										1979 025B	A	Unannounced	60	LEO (95.8)	MIL	USA
	127	May 28	73	24B	Agena	24B-25	3B-61	V	SLC 4W	1979 044A		Unannounced	4000	SSO	MIL	USA
	128	Jun 10	13	3C	Transtage	23C-13	3C-31	C	LC 40	1979 053A		DSP 11	1170	GEO	MIL	USA
	129	Oct 01	113	3C	Transtage	23C-16	3C-34	C	LC 40	1979 086A		Unannounced	820		MIL	USA
	130	Nov 21	51	3C	Transtage	23C-19	3C-37	C	LC 40	1979 098A	M	DSCS 2 13	612	GEO	MIL	USA
										1979 098B	M	DSCS 2 14	612	GEO	MIL	USA
	131	1980 Feb 07	78	3D			23D-19	V	SLC 4E	1980 010A		Unannounced	13300	LEO (96.9)	MIL	USA
	132	Jun 18	132	3D			23D-16	V	SLC 4E	1980 052A		Unannounced	13300	SSO	MIL	USA
	133	Dec 13	178	34B	Agena	34B-3	3B-53	V	SLC 4W	1980 100A		Unannounced	810		MIL	USA
	134	1981 Feb 28	77	24B	Agena	24B-24	3B-59	V	SLC 4W	1981 019A		Unannounced	3000	SSO	MIL	USA
	135	Mar 16	16	3C	Transtage	23C-22	3C-40	C	LC 40	1981 025A		DSP 10	1170	GEO	MIL	USA
	136	Apr 24	39	34B	Agena	34B-8	3B-60	V	SLC 4W	1981 038A		Unannounced		LEO (62.7)	MIL	USA
	137	Sep 03	132	3D			23D-22	V	SLC 4E	1981 085A		Unannounced	13300	SSO	MIL	USA
	138	Oct 31	58	3C	Transtage	23C-21	3C-39	C	LC 40	1981 107A		Unannounced			MIL	USA
	139	1982 Jan 21	82	24B	Agena	24B-26	3B-62	V	SLC 4W	1982 006A		Unannounced		LEO (97.2)	MIL	USA
	140	Mar 06	44	3C	Transtage	23C-20	3C-38	C	LC 40	1982 019A		DSP 13	1170	GEO	MIL	USA
	141	May 11	66	3D			23D-24	V	SLC 4E	1982 041A		Unannounced		SSO	MIL	USA
	142	Oct 30	172	34D	IUS	04D-5	34D-1	C	LC 40	1982 106A	M	DSCS 2 15	590	GEO	MIL	USA
										1982 106B	M	DSCS 3 1	1040	GEO	MIL	USA
	143	Nov 17	18	3D			23D-23	V	SLC 4E	1982 111A		Unannounced		SSO	MIL	USA
	144	1983 Apr 15	149	24B	Agena	24B-27	3B-63	V	SLC 4W	1983 032A		Unannounced		SSO	MIL	USA
	145	Jun 20	66	34D		04D-3	34D-5	V	SLC 4E	1983 060A		Unannounced	32000	SSO	MIL	USA
	146	Jul 31	41	34B	Agena	34B-9	3B-65	V	SLC 4W	1983 078A		Unannounced			MIL	USA
	147	1984 Jan 31	184	34D	Transtage	05D-1	34D-10	C	LC 40	1984 009A		Unannounced	1043		MIL	USA
	148	Apr 14	74	34D	Transtage	34D-11	34D-11	C	LC 40	1984 037A		DSP 12		GEO	MIL	USA
	149	Apr 17	3	24B	Agena	24B-28	3B-67	V	SLC 4W	1984 039A		Unannounced		SSO	MIL	USA
	150	Jun 25	69	34D		04D-1	34D-4	V	SLC 4E	1984 065A		USA 2	13300	SSO	MIL	USA
										1984 065C	A	USA 3	60	LEO (96.1)	MIL	USA
	151	Aug 28	64	34B	Agena	34B-4	3B-64	V	SLC 4W	1984 091A		USA 4			MIL	USA
	152	Dec 04	98	34D		04D-4	34D-6	V	SLC 4E	1984 122A		USA 6	13000		MIL	USA
	153	Dec 22	18	34D	Transtage	05D-3	34D-13	C	LC 40	1984 129A		DSP 6R	1170	GEO	MIL	USA
	154	1985 Feb 08	48	34B	Agena	34B-10	3B-69	V	SLC 4W	1985 014A		USA 9	550		MIL	USA
F	**155**	**Aug 28**	**201**	**34D**		**04D-6**	**34D-7**	**V**	**SLC 4E**	1985 F02A		USA			MIL	USA
F	**156**	**1986 Apr 18**	**233**	**34D**		**04D-2**	**34D-9**	**V**	**SLC 4E**	1986 F03A		USA			MIL	USA
	157	1987 Feb 12	300	34B	Agena	34B-51	3B-66	V	SLC 4W	1987 015A		USA 21			MIL	USA
	158	Oct 26	256	34D		04D-8	34D-15	V	SLC 4E	1987 090A		USA 27			MIL	USA
	159	Nov 29	34	34D	Transtage	05D-4	34D-8	C	LC 40	1987 097A		DSP 5R	1170	GEO	MIL	USA
F	**160**	**1988 Sep 02**	**278**	**34D**	**Transtage**	**05D-5**	**34D-3**	**C**	**LC 40**	1988 077A		USA 31			MIL	USA
	161	Sep 05	3	2			23G-1	V	SLC 4W	1988 078A		USA 32			MIL	USA
	162	Nov 06	62	34D		04D-7	34D-14	V	SLC 4E	1988 099A		USA 33			MIL	USA
	163	1989 May 10	185	34D	Transtage	05D-6	34D-16	C	LC 40	1989 035A		USA 37			MIL	USA
	164	Jun 14	35	402A	IUS	K-1	45D-1	C	LC 41	1989 046A		DSP 14	2355	GEO	MIL	USA
	165	Sep 04	82	34D	Transtage	05D-7	34D-2	C	LC 40	1989 069	M	USA 43 - 44			MIL	USA
	166	Sep 06	2	2			23G-2	V	SLC 4W	1989 072A		USA 45			MIL	USA
	167	1990 Jan 01	117	3			CT-1	C	LC 40	1990 001A	C	Skynet 4A	1463	GTO	MIL	UK
										1990 001B	C	JCSat 2	2280	GTO	CML	Japan
F	**168**	**Mar 14**	**72**	**3**			**CT-2**	**C**	**LC 40**	1990 021A		Intelsat 603	4215	GTO	CML	Int'l
	169	Jun 08	86	405A		K-4	45H-4	C	LC 41	1990 050	M	USA 59 - 62			MIL	USA
	170	Jun 23	15	3			CT-3	C	LC 40	1990 056A		Intelsat 604	4215	GTO	CML	Int'l
	171	Nov 13	143	402A	IUS	K-6	45D-2	C	LC 41	1990 095A		USA 65	2355		MIL	USA
	172	1991 Mar 08	115	403A		K-5	45F-1	V	SLC 4E	1991 017A		USA 69		LEO (68)	MIL	USA
	173	Nov 08	245	403A		K-8	45F-2	V	SLC 4E	1991 076	M	USA 72, 74, 76, 77			MIL	USA
	174	1992 Apr 25	169	2			23G-3	V	SLC 4W	1992 023A		USA 81			MIL	USA
	175	Sep 25	153	3	TOS		CT-4	C	LC 40	1992 063A		Mars Observer	2573	Mars	CIV	USA
	176	Nov 28	64	404A		K-3	45J-1	V	SLC 4E	1992 083A		USA 86			MIL	USA
F	**177**	**1993 Aug 02**	**247**	**403A**		**K-11**	**45F-9**	**V**	**SLC 4E**	1993 F02A		USA			MIL	USA
S	178	Oct 05	64	2			23G-5	V	SLC 4W	1993 F04A		Landsat 6	1740	SSO	CIV	USA
	179	1994 Jan 25	112	2			23G-11	V	SLC 4W	1994 004A		Clementine 1	424	Moon	MIL	USA
	180	Feb 08	14	401A	Centaur	K-10	45E-3	C	LC 40	1994 009A		Milstar 1	4530	GEO	MIL	USA
	181	May 03	84	401A	Centaur	K-7	45E-1	C	LC 41	1994 026A		USA 103			MIL	USA
	182	Aug 27	116	401A	Centaur	K-9	45E-2	C	LC 41	1994 054A		USA 105			MIL	USA
	183	Dec 22	117	402A	IUS	K-14	45D-3	C	LC 40	1994 084A		DSP 17	2355	GEO	MIL	USA
	184	1995 May 14	143	401A	Centaur	K-23	45E-8	C	LC 40	1995 022A		USA 110			MIL	USA
	185	Jul 10	57	401A	Centaur	K-19	45E-5	C	LC 41	1995 034A		USA 112			MIL	USA
	186	Nov 06	119	401A	Centaur	K-21	45E-7	C	LC 40	1995 060A		Milstar 2	4530	GEO	MIL	USA
	187	Dec 05	29	404A		K-15	45J-3	V	SLC 4E	1995 066A		USA 116			MIL	USA
	188	1996 Apr 24	141	401A	Centaur	K-16	45E-4	C	LC 41	1996 026A		USA 118			MIL	USA
	189	May 12	18	403A		K-22	45F-11	V	SLC 4E	1996 029A	M	USA 119	700	LEO (63.4)	MIL	USA

V = Vandenberg AFB SLC 4E and 4W; C = Cape Canaveral Air Force Station LC40 and 41

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Upper Stage	Vehicle Number	Flight		Pad	Payload	Payload Designation	Name	Mass (kg)	Orbit	Market (Incl)	Country
										1996 029B	M	USA 120		LEO (63.4)	MIL	USA
										1996 029C	M	USA 121		LEO (63.4)	MIL	USA
										1996 029D	M	USA 122		LEO (63.4)	MIL	USA
										1996 029E	A	TiPS Ralph (USA 123)	53	LEO (63.4)	MIL	USA
										1996 029F	A	TiPS Norton (USA 124)	53	LEO (63.4)	MIL	USA
	190	Jul 03	52	405A		K-2	45H-1	C	LC 40	1996 038A		USA 125			MIL	USA
	191	Dec 20	170	404A		K-13	45J-5	V	SLC 4E	1996 072A		USA 129			MIL	USA
	192	1997 Feb 23	65	402B	IUS	K-24	4B-24	C	LC 40	1997 008A		DSP 18		GEO	MIL	USA
	193	Apr 04	40	2			23G-6	V	SLC 4W	1997 012A		DMSP 5D-2 14		SSO	MIL	USA
	194	Oct 15	194	401B	Centaur	K-33	4B-33	C	LC 40	1997 061A		Cassini/ Huygens	5600	Saturn	CIV	USA
	195	Oct 24	9	401A	Centaur	K-18	4A-18	V	SLC 4E	1997 064A		USA 133			MIL	USA
	196	Nov 08	15	401A	Centaur	K-20	4A-17	C	LC 41	1997 068A		USA 136			MIL	USA
	197	1998 May 09	182	401B	Centaur	K-25	4B-25	C	LC 40	1998 029A		USA 139			MIL	USA
	198	May 13	4	2			23G-12	V	SLC 4W	1998 030A		NOAA 15 (NOAA K)	2232	SSO	CIV	USA
F	**199**	**Aug 12**	**91**	**401A**	**Centaur**	**K-17**	**4A-20**	**C**	**LC 41**	1998 F02A		USA			MIL	USA
S	200	1999 Apr 09	240	402B	IUS	K-32	4B-27	C	LC 41	1999 017A		DSP 19 (USA 142)	2382	GEO	MIL	USA
F	**201**	**Apr 30**	**21**	**401B**	**Centaur**	**K-26**	**4B-32**	**C**	**LC 40**	1999 023A		Milstar 2-1 (USA 143)			MIL	USA
	202	May 22	22	404B			4B-12	V	SLC 4E	1999 028A		USA 144			MIL	USA
	203	Jun 20	29	2			23G-7	V	SLC 4W	1999 034A		QuikScat	970	LEO (90)	CIV	USA
	204	Dec 12	165	2			23G-8	V	SLC 4W	1999 067A		DMSP 15	794	SSO	MIL	USA
	205	2000 May 08	148	402B	IUS		4B-29	C	LC 40	2000 024A		DSP 20	2350	GEO	MIL	USA
	206	Aug 17	101	403B			4B-28	V	SLC 4E	2000 047A		USA 152	13611	LEO (68)	MIL	USA
	207	Sep 21	35	2			23G-13	V	SLC 4W	2000 055A		NOAA 16 (NOAA L)	1946	SSO	CIV	USA
	208	2001 Feb 27	159	401B	Centaur	K-30	4B-41	C	LC 40	2001 009A		Milstar 2-2	4536	GEO	MIL	USA
	209	Aug 06	160	402B	IUS		4B-31	C	LC 40	2001 033A		DSP 21	2300	GEO	MIL	USA
	210	Oct 05	60	404B			4B-34	V	SLC 4E	2001 044A		USA 161	16650	SSO	MIL	USA
	211	2002 Jan 16	103	401B	Centaur		4B-38	C	LC 40	2002 001A		Milstar 2-3	4550	GEO	MIL	USA
	212	Jun 24	159	2			23G-14	V	SLC 4W	2002 032A		NOAA 17 (NOAA M)	2232	SSO	CIV	USA
	213	2003 Jan 06	196	2			23G-4	V	SLC 4W	2003 001A		Coriolis	827	SSO	MIL	USA
	214	Apr 08	92	401B	Centaur		4B-35	C	LC 40	2003 012A		Milstar 6	4500	GEO	MIL	USA
	215	Sep 09	154	401B	Centaur		4B-36	C	LC 40	2003 041A		USA 171	4500	GEO	MIL	USA
	216	Oct 18	39	2			23G-9	V	SLC 4W	2003 048A		DMSP F-16	1154	SSO	MIL	USA

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Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Failure Descriptions:

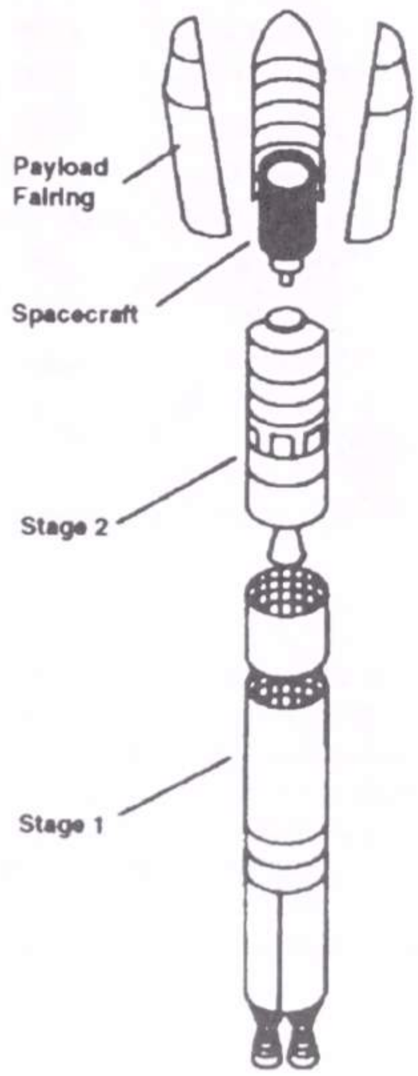
F	1964 Sep 01	3A-2	1964 F10	Third stage (Transtage) pressurization failure caused premature shutdown and failure to reach orbit.
S	1965 Feb 11	LES 1	1965 08C	LES 1 satellite perigee kick motor failed to fire, but some planned tests were accomplished.
F	1965 Oct 15	3C-4	1965 082	Transtage engine failed to shut down, causing the vehicle to tumble, then explode.
P	1965 Dec 21	3C-8	1965 108	Transtage attitude control valve stuck open causing propellant depletion and loss of attitude control; three payloads operated in incorrect orbit, the fourth (LES 3) failed to deploy.
F	1966 Aug 26	3C-12	1966 F08	Payload fairing broke up 78 s after launch, causing vehicle to pitch and break up.
F	1967 Apr 26	3B-5	1967 F04	Stage 2 engine lost thrust.
P	1970 Nov 06	23C-1	1970 093A	Guidance platform not properly aligned before launch, causing delivery to lower than planned orbit.
S	1971 Nov 03	DSCS 2 2	1971 095B	Antenna deployment caused dynamic instability.
F	1972 Feb 16	33B-2	1972 F01	Agena upper stage failed to reach orbit.
F	1972 May 20	24B-4	1972 F03	Agena upper stage failed to reach orbit.
F	1973 Jun 26	24B-9	1973 F04	Agena upper stage failed to reach orbit.
F	1974 Feb 11	23E-1	1974 F01	Centaur LOX pump failure prevented ignition (First Centaur flown on Titan).
F	1975 May 20	23C-7	1975 040	Transtage gyros lost power, causing loss of attitude control and injection into low orbit.
F	1978 Mar 25	23C-17	1978 F01	Second-stage hydraulic pump failure.
F	1985 Aug 28	34D-7	1985 F02	Destroyed by range safety officer after first-stage propellant feed system leaked and one engine turbopump failed.
F	1986 Apr 18	34D-9	1986 F03	Ignition pressure caused SRM case insulation to debond between segments, resulting in case burn through and vehicle explosion 8.5 s after liftoff, damaging launch pad.
F	1988 Sep 02	34D-3	1988 077	Damage to Transtage resulted in hydrazine and helium leaks, which prevented ignition of the second burn for injection into GEO.
F	1990 Mar 14	CT 2	1990 021A	Second stage reached correct orbit but failed to deploy payload because of incorrect interface wiring; Intelsat 603 separated itself from its kick stage and was rescued and reboosted by astronauts on STS-49 flight.
F	1993 Aug 02	K-11	1993 F02	A radial cut in the propellant of one SRM segment during repairs permitted combustion propagation to the motor case, causing the motor to explode 101 s after liftoff, destroying the vehicle.

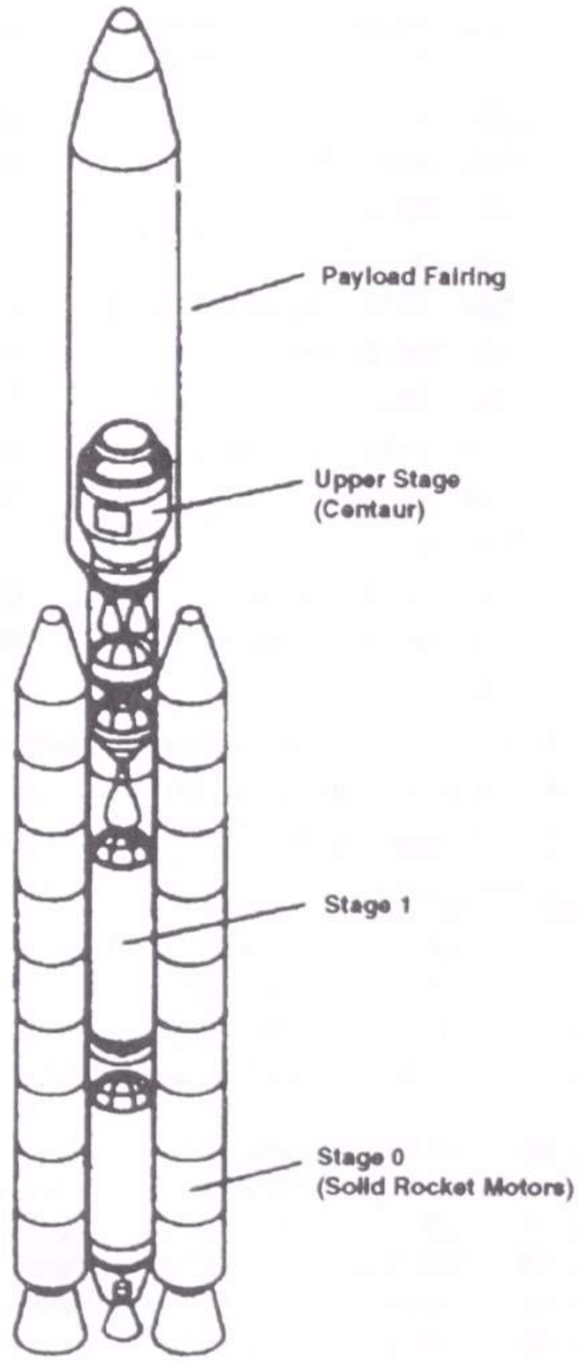
Vehicle history

S	1993 Oct 05	Landsat 6	1993 F04A	Because of a ruptured manifold, which prevented flow of fuel to reaction engine assemblies, satellite was not properly oriented when kick motor fired 820 s after launch, causing failure to acheive orbit.
F	1998 Aug 12	4A-20	1998 F02	Intermittent power shorts, possibly caused by a damaged cable, caused the inertial guidance unit to lose its reference attitude and begin generating improper steering commands; vehicle pitched over 40 s into flight and was destroyed by aerodynamic forces.
S	1999 Apr 09	DSP 19	1999 017A	Titan performed correctly, but following burnout of first IUS stage, the IUS second stage failed to completely separate because of thermal tape overwrapped an electrical connector. As a result the second-stage nozzle extension could not properly deploy, and the payload was left stranded in an elliptical orbit.
F	1999 Apr 30	4B-32	1999 023A	Centaur attitude control propellant depleted prematurely, causing deployment of payload in incorrect low orbit. Fault traced to incorrect roll rate parameter in Centaur flight software, a decimal point misplaced by human error during manual data entry.

Vehicle Design

Overall Vehicle



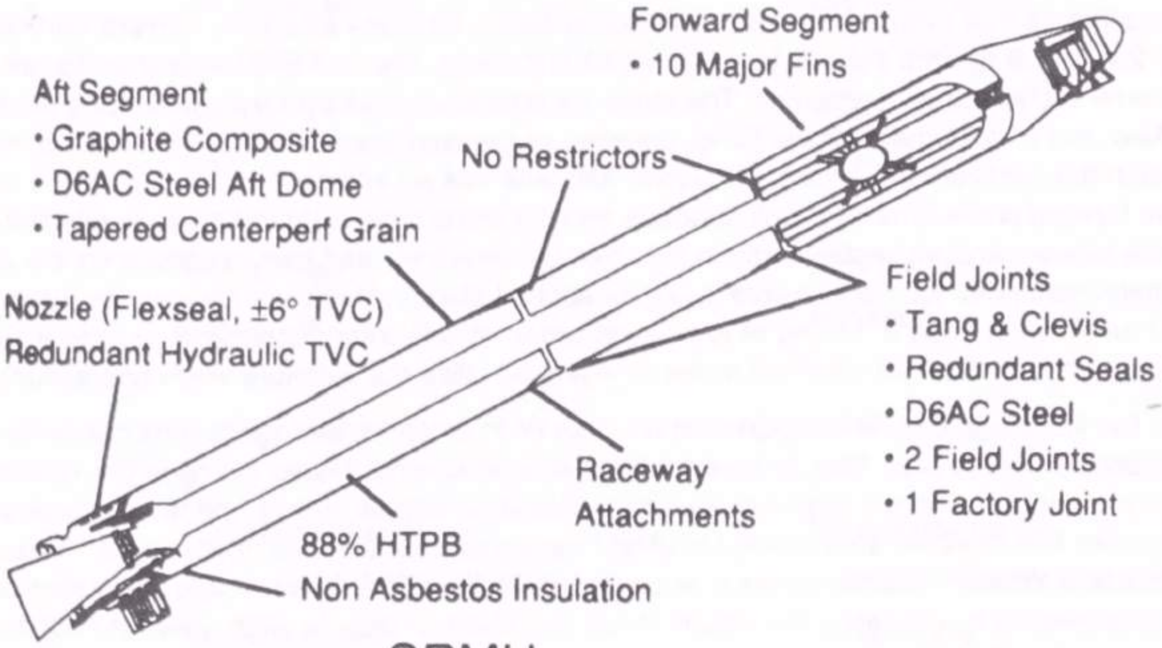


	Titan II	Titan IVB
Height	Up to 42.9 m (148 ft)	Up to 62.2 m (204 ft)
Gross Liftoff Mass	155 t (340 klbm)	925 t (2040 klbm)
Thrust at Liftoff	2090 kN (474 klbf)	Peak vacuum: 15 MN (3400 klbf)

VEHICLE DESIGN

Solid Rocket Motor Upgrade (SRMU)

The liftoff thrust of the Titan IVB is provided solely by the two solid strap-on boosters, which together are referred to as Stage 0. The motors ignite at liftoff and provide thrust and flight control until the Titan IVB first stage ignites shortly before booster burnout. One of the primary changes for the Titan IVB is the use of new motors designated SRMUs, which replace the SRMs used since the 1960s on earlier Titan III and Titan IV vehicles. The SRMU motors have higher performance because of an increased propellant load and lighter composite cases, replacing the steel cases used previously. The SRMU motors have flexseal nozzles that are gimballed by hydraulic actuators. These replace the LITVC systems used in the SRMs. The number of segments is reduced from seven to three to improve reliability.



SRMU

	Titan IVB
Dimensions	
Length	34.3 m (112.4 ft)
Diameter	3.20 m (10.5 ft)
Mass	
Propellant Mass (each)	313 t (688 klbm)
Inert Mass (each)	37 t (81 klbm)
Gross Mass (each)	350 t (769 klbm)
Propellant Mass Fraction	0.89
Structure	
Type	Filament-wound monocoque
Material	Graphite–epoxy composite
Propulsion	
Motor Designation	SRMU
Number of Motors	2
Propellant	88% HTPB
Number of Segments	3
Peak Thrust (each)	Vacuum: 7500 kN (1700 klbf)
Isp	Vacuum: 285.6 s
Chamber Pressure	86.9 bar (1260 psi)
Nozzle Expansion Ratio	16:1
Attitude Control	
Pitch, Yaw	Hydraulic gimbaling ±6 deg
Roll	Hydraulic gimbaling ±6 deg
Staging	
Nominal Burn Time	137.8 s
Shutdown Process	Burn to depletion
Stage Separation	6 solid retro-rockets each (3 forward and 3 aft)

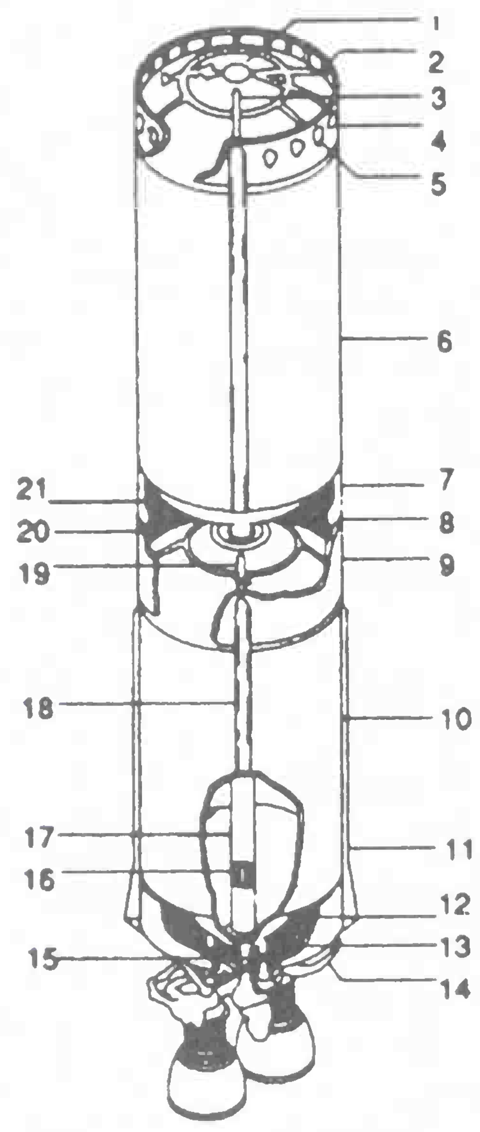
Vehicle Design

Stage 1

Because of the Titan's heritage as an ICBM, both the first and second stage use storable hypergolic propellants–N_2O_4 oxidizer and Aerozine-50 fuel. Aerozine-50 contains approximately 50% UDMH and 50% hydrazine by weight. It combines the stability of UDMH and higher performance of hydrazine, but is one of the most expensive liquid fuels in use. Use of hypergolic propellants, which burn spontaneously when mixed, eliminates the need for an igniter, thus improving staging ignition reliability. These propellants are storable and can remain in a launch-ready state for extended periods. The propellant tanks are pressurized on the ground with nitrogen. An autogenous pressurization system maintains the in-flight pressure requirements using cooled fuel-rich turbine exhaust gas for the fuel tank and vaporized N_2O_4 for the oxidizer tank. The first stage is powered by an LR87 liquid-propellant rocket engine, which consists of two independent subassemblies, each with one turbopump set for fuel and oxidizer and one thrust chamber. Therefore, the LR87 is effectively a two-engine system. An upgraded version of the Titan IV LR87-AJ-11 engine is used for most Titan IVB flights. Because the new solid boosters burn out at a higher altitude, the engine expansion ratio is increased slightly resulting in increased performance.

The engines are attached to an airframe that includes the fuel and oxidizer tanks, intertank structure, forward skirt, and aft skirt. The total length of the Titan IV first stage is stretched 2.4 m (7.9 ft) from the previous Titan 34D first stage. The fuel and oxidizer tanks are welded structures consisting of a forward dome, a skin-stringer barrel section, and an aft dome. The tanks are structurally independent, minimizing the hazard of propellant mixing should a leak develop in either tank. Also, this arrangement allows filling, draining, or pressurization of either tank without jeopardizing structural integrity. The tanks are mounted in tandem with the oxidizer tank on top. The lower fuel tank has an internal conduit to duct the oxidizer to the rocket engine. Each tank has an access cover in the forward dome. The intertank structure and the skirts have welded frames to which the aerodynamic surface is riveted. Access doors are provided in the forward and aft sections and in the intertank structure, and four longerons on the aft skirt allow for attachment of the solid boosters. (The forward attachment point for the boosters is on the second stage). Aluminum alloy is used throughout for skin areas with gauges about twice as thick in Titan IV as those in Titan II. Milling of low-stress areas of tank interiors reduces vehicle weight. Integrally milled T-shape skin stringers extend longitudinally in each tank. Frames attached to the stringers stabilize the structure when unpressurized.

A boat-tail heat shield encloses the first-stage engine compartment on Titan IV to protect first-stage engine components from radiant heat produced by the exhaust plumes of the solid boosters. (Because Titan II does not have solid strap-ons, it does not have this boat tail.) The boat tail is aluminum and covers those portions of the engine above the thrust chamber assembly. Separate engine covers, made of refrasil sandwiched between two layers of Inconel, protect each thrust chamber. Exit closures are provided to shield the thrust chamber interior from heat. These closures are explosively separated when first stage engine operation is initiated. Rubber covers over the turbine exhaust stacks are blown off by start-cartridge gas pressure upon engine start. Thrust vector control is accomplished by gimbaling the engine thrust chambers to provide pitch, yaw, and roll corrections. Hydraulic actuators, driven from the engine turbopump and controlled by electrical signals from the guidance and flight control systems, provide the gimbal force. The first stage can be shut down using one of three different methods—upon command from the guidance software, when tank sensors detect low propellant levels, or when either one of the propellants is exhausted. The first stage is separated using a fire-in-the-hole technique in which the second stage engine ignites in the interstage to separate the stages. Large vent areas in the interstage prevent overpressurization during ignition. On at least one Titan II flight, the ignition of the second stage caused the propellant remaining in the first stage to explode, but this did not affect the outcome of the mission.



1. Tank Entry Dome
2. Forward Dome
3. Oxidizer Tank Autogenous Pressurization Line
4. Forward Skirt Oxidizer Dome
5. Blast Ports
6. Oxidizer Tank Barrel
7. Aft Skirt Oxidizer Tank
8. Tension Splice
9. Fuel Tank Forward Tank Skirt
10. Fuel Tank Barrel
11. Tank Panel Longeron
12. Aft Cone
13. Fuel Suction Line
14. Air Scoop (TIIIB Only)
15. Oxidizer Suction Line
16. Oxidizer Suction Line
17. Internal Conduit
18. External Conduit
19. Fuel Tank Augotenous Pressurization Line
20. Forward Dome Fuel Tank
21. Aft Dome

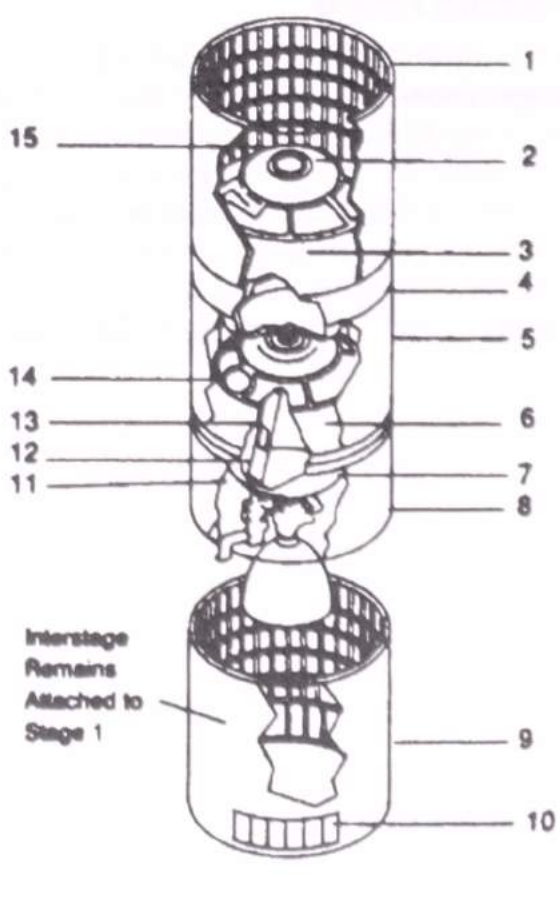
Vehicle Design

Stage 1

	Titan II	Titan IVB
Dimensions		
Length	21.5 m (70.7 ft)	26.4 m (86.5 ft)
Diameter	3.05 m (10.0 ft)	3.05 m (10.0 ft)
Mass		
Propellant Mass	118 t (260 klbm)	155 t (340 klbm)
Inert Mass	4 t (9 klbm)	8 t (19 klbm)
Gross Mass	122 t (269 klbm)	163 t (359 klbm)
Propellant Mass Fraction	0.97	0.95
Structure		
Type	Skin-stringer	Skin-stringer
Material	Aluminum	Aluminum
Propulsion		
Engine Designation	LR87-AJ-5	LR87-AJ-11
Number of Engines	2	2
Propellant	N_2O_4/Aerozine-50	N_2O_4/Aerozine-50
Average Thrust	Vacuum: 2090 kN (474 klbf)	Vacuum: 2452 kN (551.2 klbf)
Isp	Vacuum: 296 s	Vacuum: 302 s
Chamber Pressure	57.2 bar (829 psi)	57.2 bar (829 psi)
Nozzle Expansion Ratio	12:1	15:1
Propellant Feed System	Gas generator	Gas generator
Mixture Ratio (O/F)	1.9:1	1.9:1
Throttling Capability	100% only	100% only
Restart Capability	No	No
Tank Pressurization	Turbine exhaust gas for fuel tank and vaporized oxidizer for oxidizer tank	Turbine exhaust gas for fuel tank and vaporized oxidizer for oxidizer tank
Attitude Control		
Pitch, Yaw	Hydraulic gimbaling (2 nozzles)	Hydraulic gimbaling (2 nozzles)
Roll	Hydraulic gimbaling (2 nozzles)	Hydraulic gimbaling (2 nozzles)
Staging		
Nominal Burn Time	147 s	164 s
Shutdown Process	Command shutdown, low-propellant sensor, or burn to depletion	Command shutdown, low-propellant sensor, or burn to depletion
Stage Separation	Stage 2 fire in the hole	Stage 2 fire in the hole

Stage 2

The second stage uses an LR91 liquid-propellant rocket engine attached to an airframe similar in construction to that of the first stage. From top to bottom the airframe consists of a transition assembly for attachment of the upper stage or payload, an oxidizer tank, an intertank structure, a fuel tank, and an aft skirt. An interstage structure connects the first and second stages. Construction of the interstage and intertank structures is similar to the first stage, with aluminum skin riveted to a welded frame. The propellant tanks are similar in structure to those of the first stage. Pitch and yaw control is accomplished by gimbaling the single thrust chamber, and roll control is provided by ducting pump turbine exhaust through a swiveled nozzle to produce torque. The Titan II second stage contains a guidance truss and an instrumentation truss in the intertank section that support the avionics and two small propellant tanks for the ACS thrusters. On Titan IV these trusses are installed in the second-stage forward skirt.



1. Transition Assembly
2. Tank Entry Cover
3. Forward Dome Oxidizer Tank
4. Oxidizer Tank
5. Between Tanks Structure
6. Forward Dome Fuel Tank
7. Aft Dome
8. Aft Skirt
9. Interstage Structure
10. Blast Ports
11. Fuel Autogenous Pressurization Line
12. Internal Conduit
13. Oxidizer Suction Line
14. Tank Entry Cover
15. Oxidizer Autogenous Pressurization

Vehicle Design

Stage 2

The second stage separates from the first stage using the fire-in-the-hole technique. When the first stage shuts down the second stage is commanded to ignite while still attached to the first stage. The overpressure is vented through interstage blast ports until the pyrotechnics separate the two stages. Depending upon the mission requirements, the second stage can be shut down by command upon achievement of payload target velocity or by a signal from low propellant level sensors, or by detection of propellant depletion. On both vehicles, retro-rockets in the base of the second stage are used to back the stage away from the separated payload.

	Titan II	Titan IVB
Dimensions		
Length	7.3 m (24 ft)	10.0 m (32.7 ft)
Diameter	3.05 m (10.0 ft)	3.05 m (10.0 ft)
Mass		
Propellant Mass	27 t (59 klbm)	35.0 t (77.2 klbm)
Inert Mass	3 t (6 klbm)	4.5 t (9.8 klbm)
Gross Mass	30 t (65 klbm)	39.5 t (87.0 klbm)
Propellant Mass Fraction	0.91	0.89
Structure		
Type	Skin-stringer	Skin-stringer
Material	Aluminum	Aluminum
Propulsion		
Engine Designation	LR91-AJ-5	LR91-AJ-11
Number of Engines	1	1
Propellant	N_2O_4/Aerozine-50	N_2O_4/Aerozine-50
Average Thrust	440 kN (100 klbf)	472.2 kN (106.15 klbf)
Isp	316 s	316 s
Chamber Pressure	57.0 bar (827 psi)	57.0 bar (827 psi)
Nozzle Expansion Ratio	49:1	49:1
Propellant Feed System	Gas-generator turbopump	Gas-generator turbopump
Mixture Ratio (O/F)	1.8:1	1.8:1
Throttling Capability	100% only	100% only
Restart Capability	No	No
Tank Pressurization	Turbine exhaust gas for fuel tank and vaporized oxidizer for oxidizer tank	Turbine exhaust gas for fuel tank and vaporized oxidizer for oxidizer tank
Attitude Control		
Pitch, Yaw	Hydraulic gimbaling	Hydraulic gimbaling
Roll	Gas-generator exhaust	Gas-generator exhaust
Staging		
Nominal Burn Time	182 s	223 s
Shutdown Process	Command shutdown, low-propellant sensor, or burn to depletion	Command shutdown, low-propellant sensor, or burn to depletion
Stage Separation	4 solid retro-rockets	4 solid retro-rockets

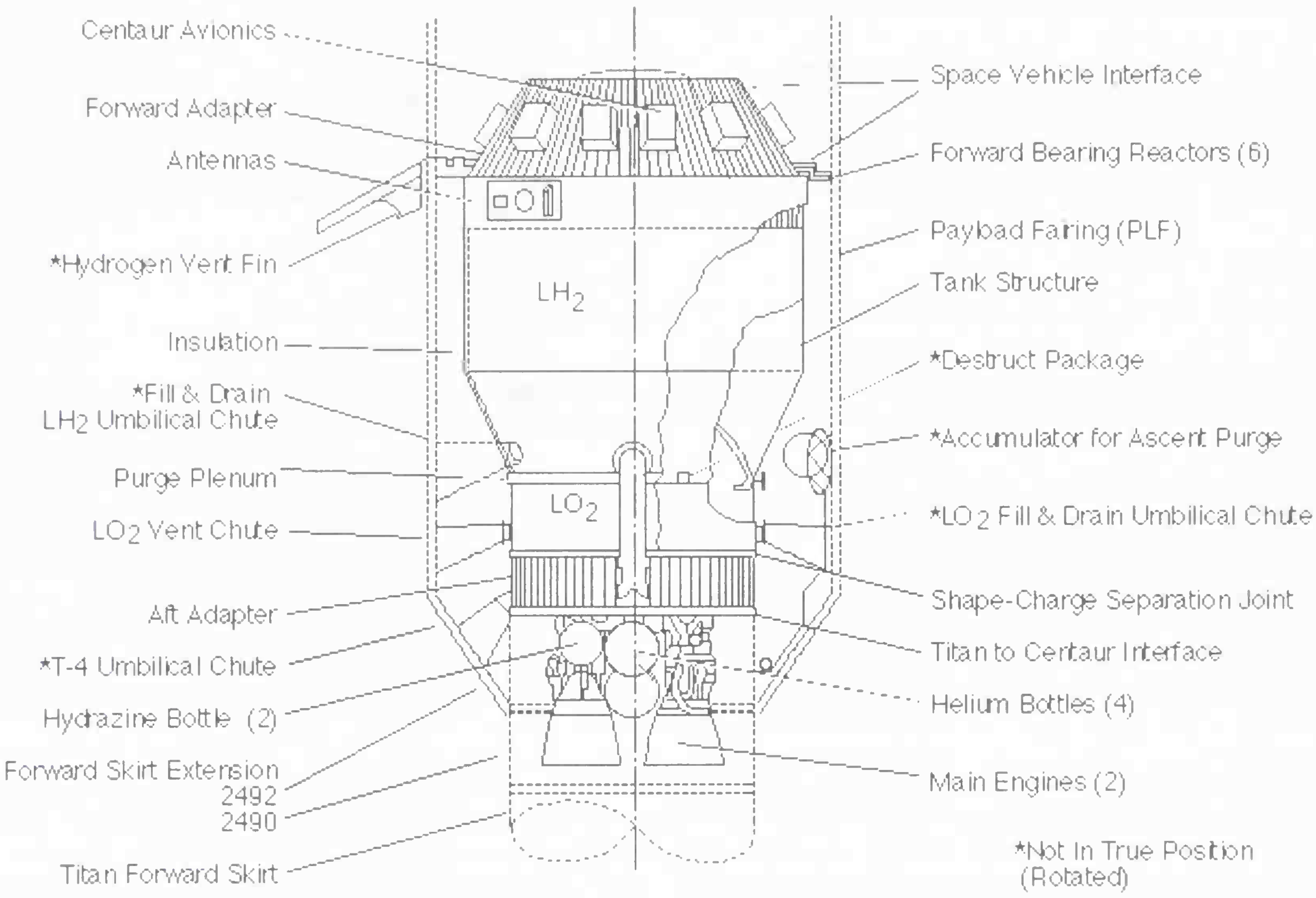
Vehicle Design

Upper Stages

The Titan IVB can carry the Centaur upper stage, the IUS, or no upper stage. The Titan II does not include an upper stage, but some missions use a Star 37-class solid kick motor that is not considered part of the launch vehicle.

Centaur

The Titan IV Centaur upper stage is manufactured by Lockheed Martin Space Systems. It is derived from the Centaur G and G-Prime stages developed for the Space Shuttle. These in turn evolved from the Centaur upper stage developed for Atlas in the early 1960s. The Titan IV Centaur differs from the Atlas Centaur stage in two major ways. First, the propellant capacity is expanded by a longer oxygen tank and wider hydrogen tank. Second, the stage power and thermal systems are upgraded to provide a longer on-orbit lifetime, allowing Centaur to perform the injection burn into GEO rather than delivering the payload to GTO as on Atlas. The Centaur consists of two stainless steel tanks, a forward adapter section that carries the avionics, and an aft propulsion section that houses the twin RL10 engines. The forward adapter is structurally connected to the payload fairing with six forward-bearing reactors, which reduce flexing of the Centaur–payload stack within the fairing during ascent. The Centaur stage contains its own avionics and ACS, which are separate from Titan's.



IUS

The IUS is produced by Boeing. Unlike Centaur, it is treated as a separate vehicle, and is not considered part of the Titan. The IUS was developed as an upper stage for the Space Shuttle and Titan to take payloads from LEO to GEO. It was originally named the Interim Upper Stage, with the expectation that it would be replaced by a reusable orbital transfer vehicle as part of the Space Transportation System architecture. When it became clear that the follow-on system would not be built, the acronym was reverse engineered to Inertial Upper Stage, referring to the inertial guidance system. The IUS is a two-stage solid system using a Pratt & Whitney CSD SRM-1 motor (also known as an Orbus® 21) to provide injection from LEO to GTO, and a smaller SRM-2 motor (Orbus 6) to circularize from GTO to GEO. Both motors are qualified for up to 50% propellant offload. The SRM-2 motor includes a standard three-piece extendable exit cone (EEC), which is deployed before ignition to increase the nozzle expansion ratio. The two motors are joined by a skin-stringer interstage. Total height of the IUS is 5.2 m (17 ft). The second stage includes the equipment support section (ESS), which supports the payload and contains the avionics bay and a hydrazine RCS system.

Vehicle Design

	Centaur	Inertial Upper Stage	
		Stage 1	Stage 2
Dimensions			
Length	9.0 m (29.5 ft)	3.5 m (11.5 ft)	2.1 m (6.9 ft)
Diameter	4.3 m (14 ft)	2.34 m (7.7 ft)	2.9 m (9.5 ft)
Mass			
Propellant Mass	20,950 kg (46,185 lbm) +150 kg (340 lbm) RCS hydrazine	9790 kg (21,580 lbm)	2750 kg (6060 lbm) solid +110 kg (245 lbm) RCS hydrazine
Inert Mass	2775 kg (6125 lbm)	1115 kg (2460 lbm)	1005 kg (2215 lbm)
Gross Mass	23,880 kg (52,650 lbm)	10,900 kg (24,040 lbm)	3865 kg (8520 lbm)
Propellant Mass Fraction	0.88	0.90	0.74
Structure			
Type	Pressure-stabilized monocoque	Motor case: filament-wound monocoque Interstage: skin-stringer	Motor case: filament-wound monocoque ESS: skin-stringer
Material	Stainless steel	Motor case: Kevlar® Interstage: aluminum	Motor case: Kevlar® ESS: aluminum
Propulsion			
Engine Designation	RL10A-3-3A (Pratt & Whitney)	SRM-1, Orbus 21 (Pratt & Whitney CSD)	SRM-2, Orbus 6E (Pratt & Whitney CSD)
Number of Engines	2	1 (1 segment)	1 (1 segment)
Propellant	LOX/LH_2	HTPB	HTPB
Average Thrust	73 kN (16.5 klbf) each	186 kN (41.8 klbf)	76.5 kN (17.2 klbf)
Isp	444 s	294 s	With EEC: 303.8 s Without EEC: 289.6 s
Chamber Pressure	32.1 bar (465 psi)	45.2 bar (655 psi)	42.1 bar (611 psi)
Nozzle Expansion Ratio	61:1	63.8:1	With EEC: 181:1 Without EEC: 49.3:1
Propellant Feed System	Split-expander turbopump	—	—
Mixture Ratio (O/F)	5:1	—	—
Throttling Capability	None	No	No
Restart Capability	Multiple: 3 burns are standard	No	No
Tank Pressurization	Fuel tank: autogenous hydrogen Oxidizer tank: Helium gas	—	—
Attitude Control			
Pitch, Yaw	Hydraulic nozzle gimbal	Electromechanical nozzle gimbal ± 4 deg	Electromechanical nozzle gimbal ± 7 deg
Roll	Hydraulic nozzle gimbal	Hydrazine RCS	Hydrazine RCS
Staging			
Nominal Burn Time	600 s	153 s	104 s
Shutdown Process	Command shutdown	Burn to depletion	Burn to depletion
Stage Separation	—	Redundant pyrotechnic nuts	—

VEHICLE DESIGN

Attitude Control System

Titan

Three-axis attitude control of the Titan IVB from liftoff to SRMU burnout is provided by gimbaling of the SRMU nozzles. The nozzles are gimbaled with hydraulic actuators driven by redundant pumps powered by gas generators. The gimbal has a range of ±6 deg, but is limited to ±5 deg during periods of high dynamic pressure to reduce aerodynamic loads on the vehicle. The gimbal angle is also limited to 1 deg inboard to reduce heating of the core stage. The first-stage ignition sequence begins shortly before strap-on booster burnout so that the first-stage engine can provide attitude control as the solid motor thrust tails off. The two engine subassemblies of the first stage provide pitch, yaw, and roll control. The engines are gimballed with hydraulic actuators powered by a hydraulic pump driven by the gearbox of the number two engine subassembly turbopump. After second-stage ignition, pitch and yaw control is provided by gimbaling of the single main engine. Roll is controlled by the turbine exhaust nozzle.

An optional hydrazine ACS kit can be installed in the Titan II second-stage intertank compartment. The system performs a wide range of propulsion functions including vehicle pitch, yaw, and roll control after second-stage burnout; orbit vernier control, and multiple payload deployment. There are six rocket engine modules, each with two thruster assemblies. Thrust varies between 71 N (16 lbf) and 120 N (27 lbf) during the mission. The hydrazine propellant is stored in a single tank with a rubber diaphragm and is pressurized by nitrogen.

Centaur

During powered flight, pitch, yaw, and roll control are provided by hydraulically actuated nozzle gimbaling of the twin RL10 engines. Centaur also has twelve 27 N (6 lbf) monopropellant hydrazine thrusters that are used for attitude control during coast phases, propellant settling, and the collision avoidance maneuver after payload separation. The hydrazine is stored in two positive-expulsion tanks with a capacity of 75 kg (170 lbm) each.

IUS

Each of the IUS solid motors has an electromechanically gimbaled nozzle that rotates on a Techroll® joint to provide pitch and yaw control. In addition, a hydrazine RCS provides roll control during solid-motor burns, three-axis control during coast phases, precision velocity corrections at orbit injection, and collision avoidance following payload separation. The RCS system contains 110 kg (240 lbm) of hydrazine in two tanks and has six rocket engine modules, each with two 133-N (30-lbf) thrusters. All RCS components are housed in the ESS surrounding the SRM-2 motor.

Avionics

Titan

Titan II guidance and electrical and ordnance systems are copied from the Titan 34D design, which was also used on the Titan IVA. Titan II guidance and instrumentation units are carried on two truss structures in the intertank compartment of the second stage. In addition, an aft rate gyro is located between the first-stage propellant tanks.

Upgraded avionics are one of two primary new features on the Titan IVB (the other being the SRMU strap-on boosters). Previous Titan III and IVA vehicles used an avionics system based on the earlier Transtage upper stage guidance system. This included a navigation unit using four gimbaled gyros. The old avionics have been replaced by a new system based on the Centaur design. This single-string system uses a strapdown platform with three Honeywell ring laser gyros and three accelerometers for inertial navigation. The new computer uses 1750A standard architecture and has twice the processing capacity of the older unit. The new guidance and control system is also less than half the cost of the older system. The instrumentation and telemetry equipment includes a data bus pulse modulation encoder and S-band transmitter. The tracking and flight safety subsystem has been redesigned for compatibility with range safety requirements. Electrical power is provided by silver–zinc batteries. Titan IVB avionics are located primarily in the second-stage forward skirt.

Centaur

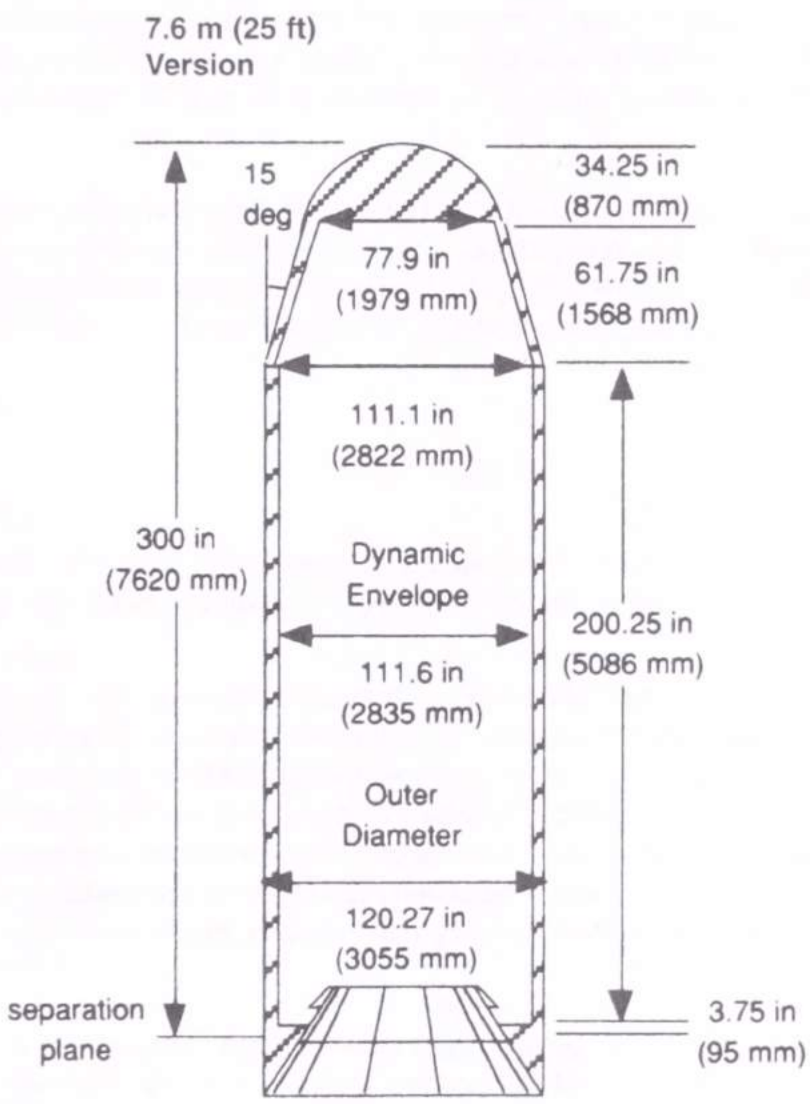
The Centaur upper stage has separate avionics systems, which are used after separation from the Titan second stage. Guidance is provided by the INU, which consists of a ring laser gyro inertial measurement unit (IMU) and a 1750A standard flight control processor (FCP). Telemetry data are collected by the data acquisition system (DAS), which includes a master data unit (MDU) and a remote data unit (RDU). Encrypted telemetry data are transmitted by a 10-W S-band transmitter with four antennas—one pair mounted on the payload fairing and one pair on the Centaur itself. Centaur software can store a vehicle state vector at any time during the mission for later transmission to a ground station. A 400-W C-band transponder is used to assist in tracking the upper stage. The flight termination system includes both automatic and command destruct systems.

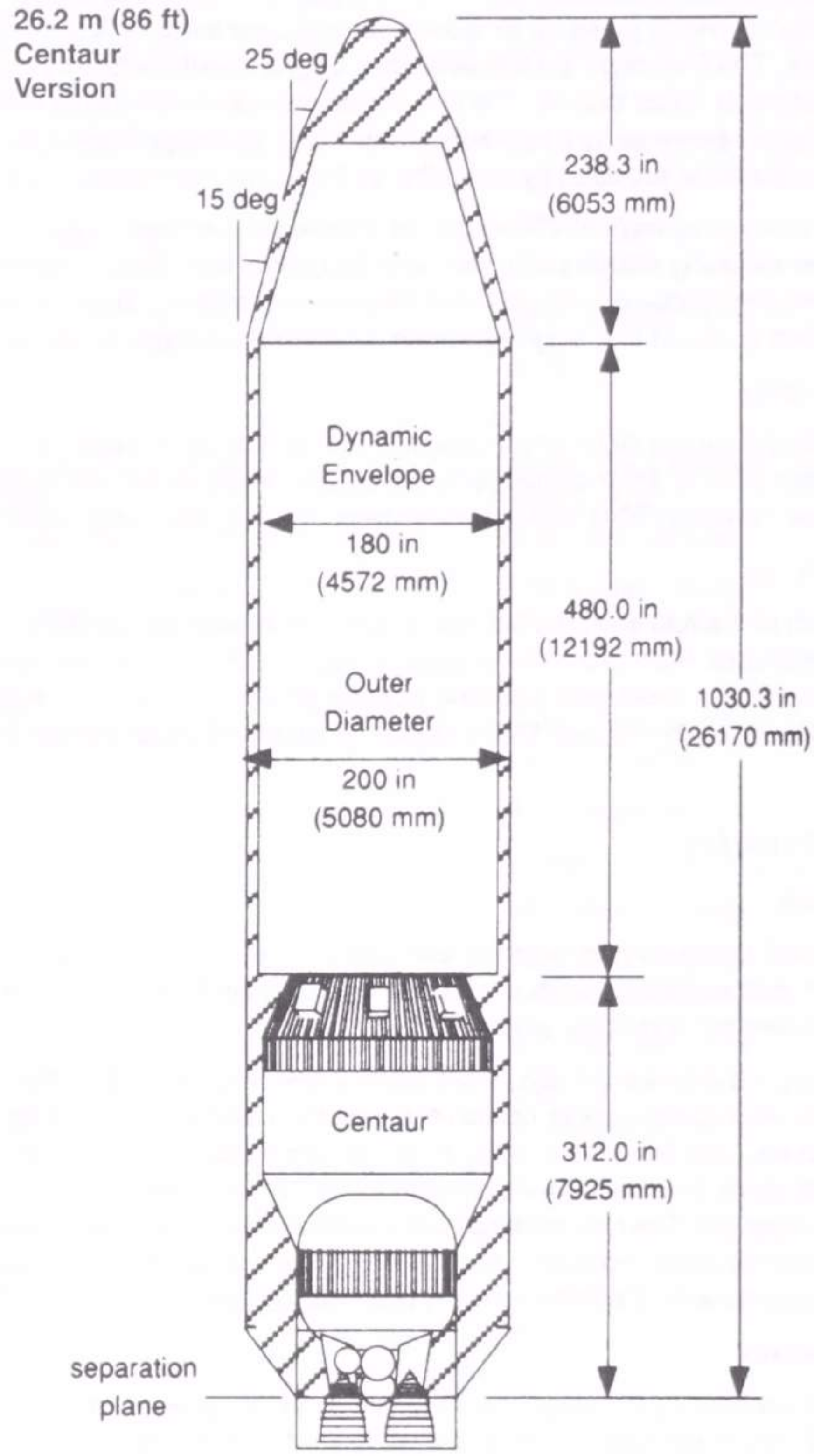
IUS

The IUS is guided by its own guidance and navigation system (GNS), which uses a strapdown redundant inertial measurement unit (RIMU) with five gyros and five accelerometers to feed measurements to the two 16-bit computers of the data management subsystem (DMS). The DMS performs the guidance calculations, controls the ignition of the two solid motors and the RCS system, and performs data processing and telemetry formatting. Other avionics systems include the TTC subsystem, the TVC system, and the power supply system.

Vehicle Design

Payload Fairing





	Titan II	Titan IVB
Length	6.1, 7.6, or 9.1 m (20, 25, or 30 ft)	15.2, 17.1, 20.1, 23.2, or 26.2 m (50, 56, 66, 76, or 86 ft)
Primary Diameter	3.05 m (10.0 ft)	5.08 m (16.7 ft)
Mass	652, 750, or 909 kg (1435, 1650, or 2000 lbm)	3.6, 5.0, 5.5, 5.9, or 6.3 t (8, 11, 12, 13, or 14 klbm)
Sections	3	3
Structure	Skin-stringer	Isogrid
Material	Aluminum	Aluminum
Remarks	The fairing is an existing design used for Titan 34D called a universal payload fairing. It can be extended in 1.5 m (5 ft) increments. The fairing has noncontaminating separation joints that contain any debris in longitudinal bellows along each trisector upon fairing jettison during flight. Access doors are standard.	The Titan IV payload fairing is derived from the design used on Titan IIIE, but is wider and longer to provide payload volume compatible with the Space Shuttle. The payload fairing is available in several lengths in 3-m (10-ft) increments. When used with the Centaur upper stage, the payload fairing provides structural support to the Centaur through attachments at the forward adapter. The fairing is separated into three parts at a speed of 4.57 m/s (15 ft/s) by ignition of a mild detonating fuse in thrusting joints. Contamination is avoided by retaining smoke, gases, and metal particles in bellows that form an integral part of each thrust structure.

Payload Accommodations

	Titan II	Titan IVB
Payload Compartment		
Maximum Payload Diameter	2832 mm (111.5 in.)	4570 mm (180 in.)
Maximum Cylinder Length	3658 mm (144 in.) for 20 ft PLF 5182 mm (204 in.) for 25 ft PLF 6706 mm (264 in.) for 30 ft PLF	9700, 12,750, 15,800, or 18,850 mm (382, 502, 622, or 742 in.)
Maximum Cone Length	1568 mm (61.75 in.)	6045 mm (238 in.)
Payload Adapter Interface Diameter	1426 mm (56.15 in.) 915 mm (36.0 in.)	?
Payload Integration		
Nominal Mission Schedule Begins	T−36 months	T−33 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T−3 min	T−5 min
On-Pad Storage Capability	30 days fueled	30 days fueled
Last Access to Payload	T−15 days (approx.) or T-5 h through access doors	?
Environment		
Maximum Axial Load	+4.0 to +10.0 *g*	+5.0 *g*
Maximum Lateral Load	±2.5 *g*	±1.5 *g*
Minimum Lateral/Longitudinal Payload Frequency	2–10 Hz avoid <6 Hz/12–24 Hz	>2.5 Hz avoid 6–10 Hz/17–24 Hz
Maximum Acoustic Level	129 dB at 500 Hz w/o blankets 125.5 dB at 250 Hz w/ blankets	129 dB at 100–600 Hz
Overall Sound Pressure Level	139.6 dB w/o blankets 134.4 dB w/ blankets for 20 and 25 ft PLF 130.0 dB w/ blankets 30 ft PLF	139.3 dB (full octave)
Maximum Flight Shock	200 *g* at 500 Hz	2000 *g* at 5000 Hz
Maximum Dynamic Pressure on Fairing	35.9 kPa (750 lb/ft^2)	47 kPa (975 lb/ft^2)
Maximum Aeroheating Rate at Fairing Separation	Payload dependent	Centaur and NUS: 315 W/m^2 (100 BTU/ft^2/h) IUS: 473 W/m^2 (150 BTU/ft^2/h)
Maximum Pressure Change in Fairing	3.5 kPa/s (0.5 psi/s)	3.5 kPa/s (0.5 psi/s)
Cleanliness Level in Fairing	Class 10,000	Class 5000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	Altitude: 185 km +1.8–0.9 km (100 nmi +1, -0.5 nmi) Inclination: 90.0 ± 0.15 deg	Perigee: 111 ± 2.0 km (60 ± 1.1 nmi) Apogee: 328 ± 8.1 km (177 ± 4.4 nmi) Inclination: 28.6 ± 0.01 deg
Attitude Accuracy (3 sigma)	All axes: ±2.0 deg; ±1.0 deg/s	?
Nominal Payload Separation Rate	0.6 m/s (2 ft/s) w/ booster retros	0.6 m/s (2 ft/s)
Deployment Rotation Rate Available	0–2 rpm (higher with spin table)	0–2 rpm
Loiter Duration in Orbit	1.5 h (batteries optional)	?
Maneuvers (Thermal/Collision Avoidance)	Yes	Yes
Multiple/Auxiliary Payloads		
Multiple Manifest	No examples of dual or multiple manifest payloads are known. Contact the USAF program office for information.	Titan IV has carried more than one primary payload on some flights, but details are not available.
Auxiliary Payloads	No examples of secondary payloads are known. Contact the USAF program office for information.	Titan IV has carried at least one auxiliary payload, the TiPS tether experiment. Availability of other flight opportunities is unknown.

Production and Launch Operations

Production

The process of converting a Titan II ICBM to a space launch vehicle begins with removal from storage at Davis–Monthan AFB, Arizona. The engines (first and second stages) are removed and shipped to Aerojet Technical Systems in Sacramento, California, where they enter a refurbishment program and are test fired to characterize the baseline engine performance.

After the engines are removed, the first- and second-stage core structures are shipped to Lockheed Martin Space Systems in Denver, Colorado, where they undergo the following modifications: 1) remanufacture of the second-stage oxidizer tank to provide additional capacity; 2) manufacture of adapter rings to allow mating of payload or upper-stage adapters; 3) fabrication of an additional equipment truss to carry unique space launch systems (destruct systems, telemetry systems, etc.); 4) installation of electrical cabling, umbilical connectors, and other unique space flight items; and 5) receipt and inspection of various new subsystems from vendors.

After completing the Lockheed Martin factory processing in Denver, the Titan II kit (except engines and guidance subsystems) is ready for shipment to VAFB where final assembly occurs. New fairings are manufactured by Boeing and are also shipped to VAFB. At VAFB, as at Lockheed Martin, resources assigned to the Titan programs are shared. That is, the core facilities and manpower support both the Titan II and Titan IV programs. The common facilities consist of the vehicle assembly building (VAB) and the payload fairing processing facility (PLFPF). The final assembly process begins in the VAB when the Titan II kit is assembled up to the state where it can either be shipped to SLC-4W for launch processing or be placed into storage pending a launch call-up.

Unlike the Titan II, all Titan IV components are built new. The Titan IV core and Centaur upper stage are produced at the Lockheed Martin Space Systems plant at Waterton, Colorado, south of Denver. Boeing serves as an associate contractor to provide the integrated IUS. Boeing also manufactures the Titan payload fairings. The following companies are involved in production of the Titan II and Titan IVB.

Organization	**Responsibility**	
	Titan II	**Titan IVB**
Lockheed Martin Space Systems	Prime contractor, core stage refurbishment, and mission integration	Prime contractor, core stages, Centaur, and mission integration
Aerojet	First- and second-stage engines	First- and second-stage engines
ATK Thiokol		SRMU boosters
Boeing	Payload fairing	IUS and payload fairing
Cincinnati Electronics	Command receivers	Command receivers
Delco Electronics	Guidance system	
Honeywell		Guidance systems
Pratt & Whitney		Centaur RL10 engines
SCI	Instrumentation	Instrumentation
United Technologies Chemical Systems Division		IUS solid motors

Production and Launch Operations

Launch Operations—Cape Canaveral Air Force Station

Titan IV launch vehicles at CCAFS are processed using an integrate–transfer–launch (ITL) concept. This permits maximum use of the launch complex because most prelaunch activities are conducted at a remote integration building. ITL consists of the SRMU processing area, vertical integration building (VIB), solid motor assembly building (SMAB), solid motor assembly and readiness facility (SMARF), launch complexes (LC) 40 and 41, a Titan transporter, and a railway system. The primary purpose of the SRMU processing area is to receive, inspect, assemble, and test inert components and store and prepare the solid boosters. The VIB is used to build-up and mate the core stages. In the SMARF, both SRMUs are stacked and mated to the core vehicle. For upper stages, the Centaur is processed in one cell of the VIB and the IUS in the SMAB. The Titan transporter not only moves the launch vehicle to the pad via railway but also serves as the launch platform. Initial spacecraft operations are performed in a building assigned in the industrial area or an offsite location for commercial payloads.

The VIB is used to inspect, erect, and checkout the Titan core (first and second stages). The VIB has four cells for assembly and checkout, a center section for an inspection area, and a launch control center for monitoring vehicle operations. Cells 1, 2, and 4 are used for Titan and cell 3 for Centaur checkout. The first stage is removed from its transportation trailer and positioned on a rotation fixture in the VIB low bay area for receiving inspection. After engine installation, the first stage is hoisted to a vertical position and placed on a transporter in a cell. First-stage components are then installed. The second-stage horizontal operations in the VIB low bay are similar to the first stage. The second stage is then hoisted and positioned on top of the first stage. Vertical alignment of the stages and stage electrical interface checks are then conducted.

A low bay area near each VIB cell provides space for checkout and instrumentation of aerospace ground equipment (AGE) vans that are attached to the transporter. After checkout, the core vehicle is prepared for transport to the SMARF. The transporter moves at 5–6 km/h (3–4 mph) and it takes 45–60 min to reach the SMARF.

The SRMUs are processed in parallel with the core in the SRMU processing area. The segments are delivered to the segment arrival storage (SAS) area. Inert parts are inspected, assembled, and tested in the motor inert storage (MIS) building. The segments are received and visually inspected for damage. In the ready inspection building (RIS), they undergo laser and video imaging, dimensional measurements, ultrasonic inspection, infrared imaging, and finally x-ray inspection. The segment ready storage (SRS) building provides a controlled environment for SRMU components before they are individually transferred by rail to the SMARF.

During the Titan III program, and for initial flights of the Titan IV, the SMAB was used for preliminary assembly of the solid motor segments and for mating the boosters to the core stages. However, the increased size of the Titan IVA SRMs, and particularly the Titan IVB SRMUs required a new facility. Therefore, the new SMARF facility is now used to stack the SRMU segments together and to mate these to the Titan core. The vehicle is then transported to the launch pad by rail. The SMAB is still used for IUS assembly and checkout.

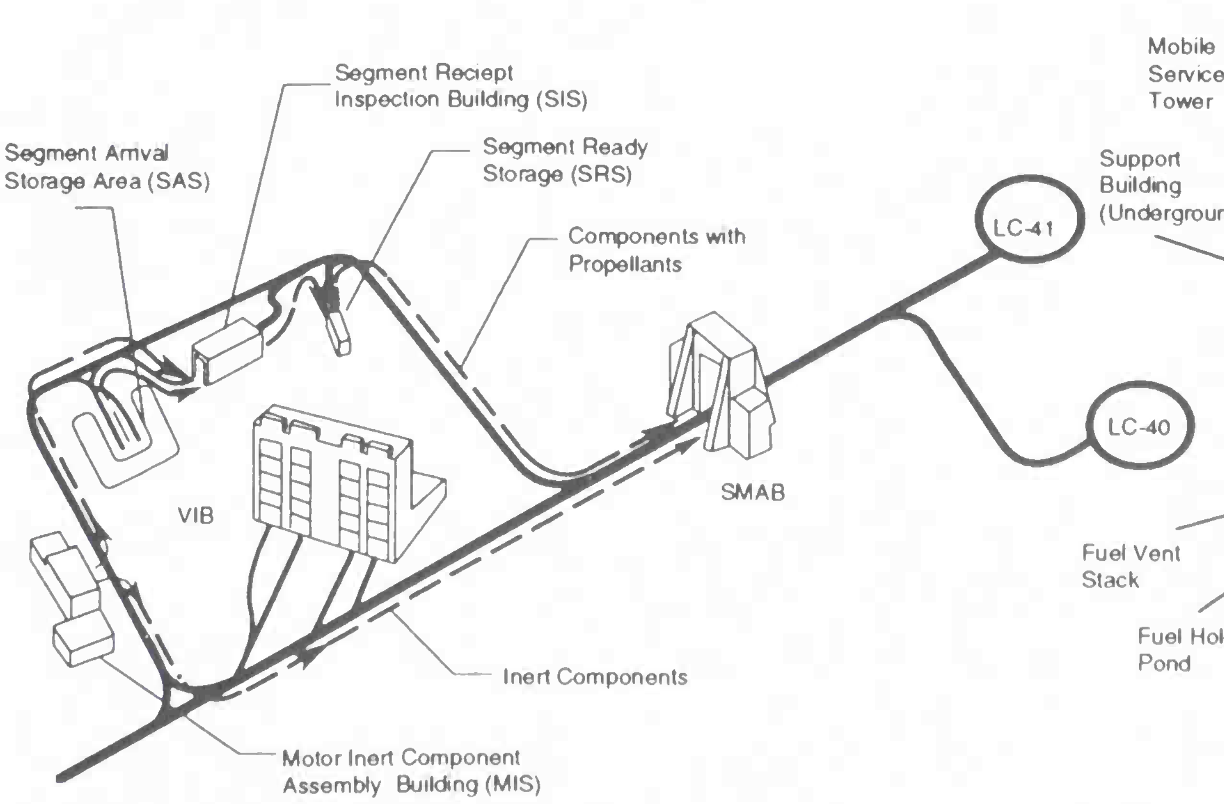
In parallel to booster processing, the Titan IV PLF is received and processed in the VIB annex. In the annex, all prelaunch processing for the PLF is accomplished. This includes mechanical preparations, thermal coating application, electrical functional checks, final cleaning, and weighing. The forward and aft PLF sections are then transported to the pad and stored in the environmental shelter.

Titan IV has been launched from LC-41 and LC-40. However, Titan operations at LC-41 ended in 1999, and the complex has been modified to support the Atlas V program. LC-40 consists of the launch pad, mobile service tower (MST), umbilical tower, AGE building, air conditioning shelter, gas storage area, propellant holding areas, and other service facilities. For Titan IV, the spacecraft and upper stage are separately hoisted and mated at the launch pad (except for IUS, which is mated at the SMAB) inside the MST. An environmental shelter above the vehicle is provided to ensure a clean and secure area for spacecraft checkout activities. Once payload activities are complete, the payload fairing segments are mated to the launch vehicle over the payload.

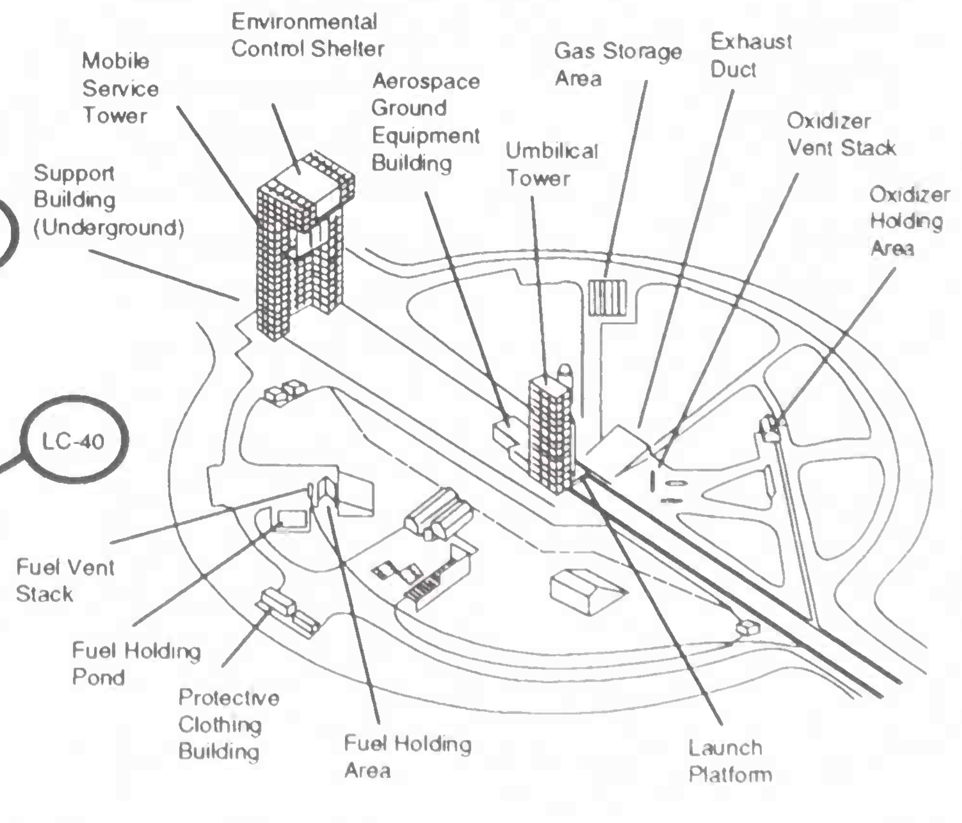
In preparation for launch, a combined systems test (CST) is performed. This includes verification checks, a terminal countdown, and simulated launch and flight. After a successful launch CST data review, the readiness countdown is started. Some of the major events in the readiness countdown are: power-on payload test, MST and umbilical tower prelaunch preparations, installation of flight ordnance and checkout, connection of core and SRMU batteries, loading of core oxidizer then fuel, pressurization of propellant tanks, and Titan guidance system launch preparations.

Launch Facilities

Cape Canaveral Air Force Station



CCAFS LC-40/41 Launch Facilities Layout



Titan CCAFS Launch Pad Layout

Production and Launch Operations

Launch Operations—Vandenberg Air Force Base

Titan II and Titan IV vehicles launched from VAFB are processed using a build-on-pad concept. Space Launch Complex-4 (SLC-4) includes two launch pads. Titan IV is launched from SLC-4E (East), while Titan II is launched from SLC-4W (West). Each launch pad area contains an MST, a propellant storage area, a high-pressure gas storage area, a launch service building on which a fixed umbilical tower is mounted, a launch mount, and a MST track system. The pads share a common launch operations building, which includes the launch control center.

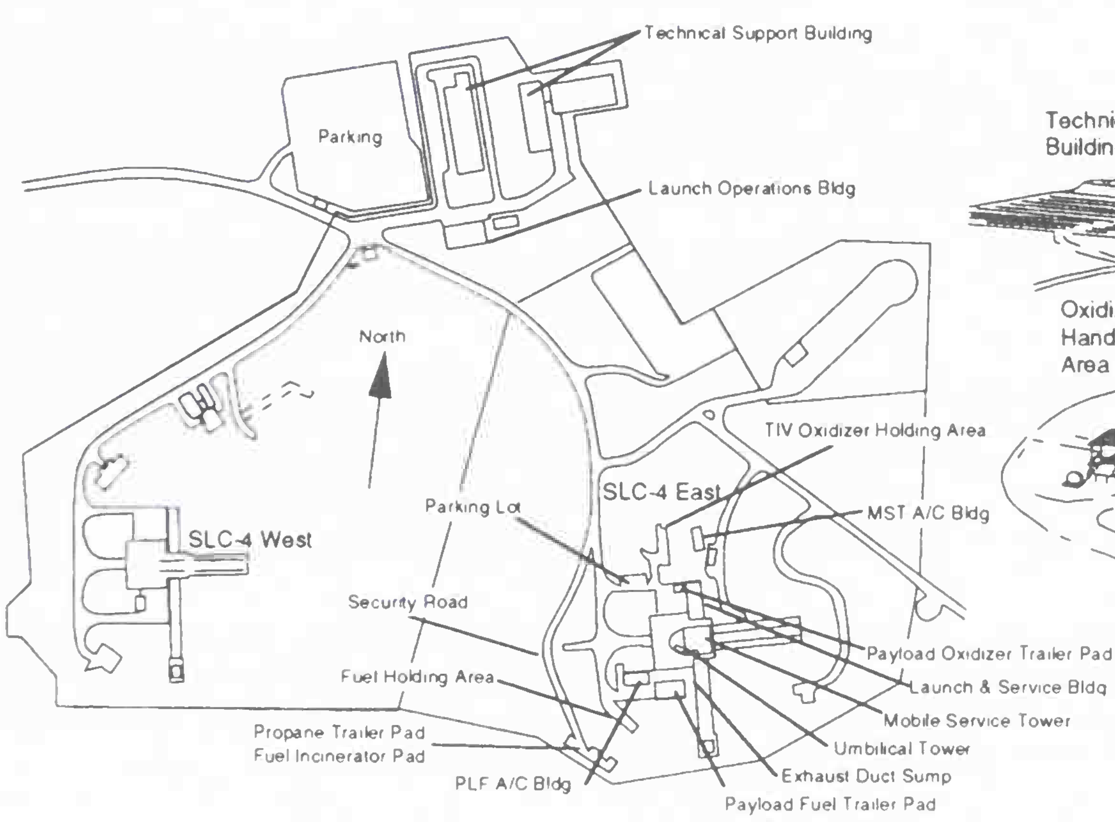
Titan IVB SRMU segments arrive via rail. The core arrives via air and the liquid engines via surface transportation. The components are transported to the VAB horizontal test facility at Building 8401. The payload fairing arrives via air and is trucked to the payload fairing cleaning facility at Building 8337. Each of these vehicle components are received, inspected, and tested to various degrees at these facilities. Next, the components are transported to SLC-4E where they are assembled and checked out on the launch mount.

Assembly begins with the SRMUs, which mechanically interface with the launch mount, followed by the first and second stages, which mechanically interface with the SRMUs. Checkout of the launch vehicle is accomplished systematically by a building block approach beginning with components testing followed by subsystem and system testing and culminates in a CST to verify the launch vehicle integrity before mating of the payload.

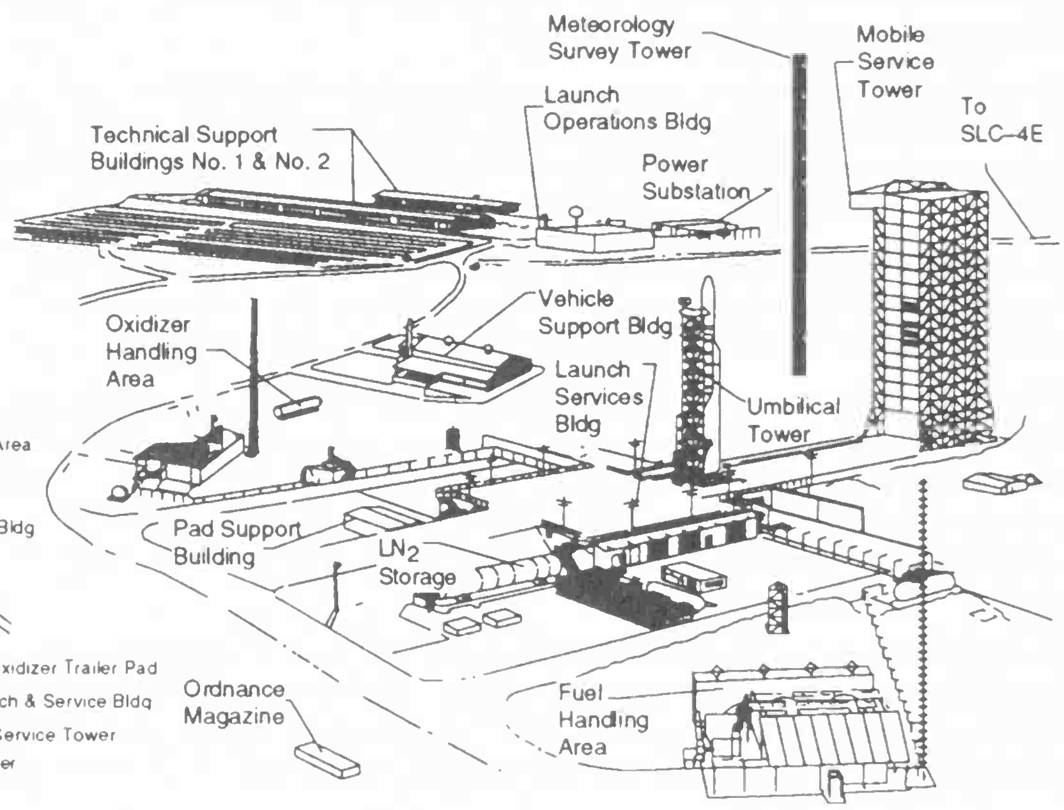
Titan II conducts the same processing procedures as Titan IV except for the SRMUs.

Launch Facilities

Vandenberg Air Force Base



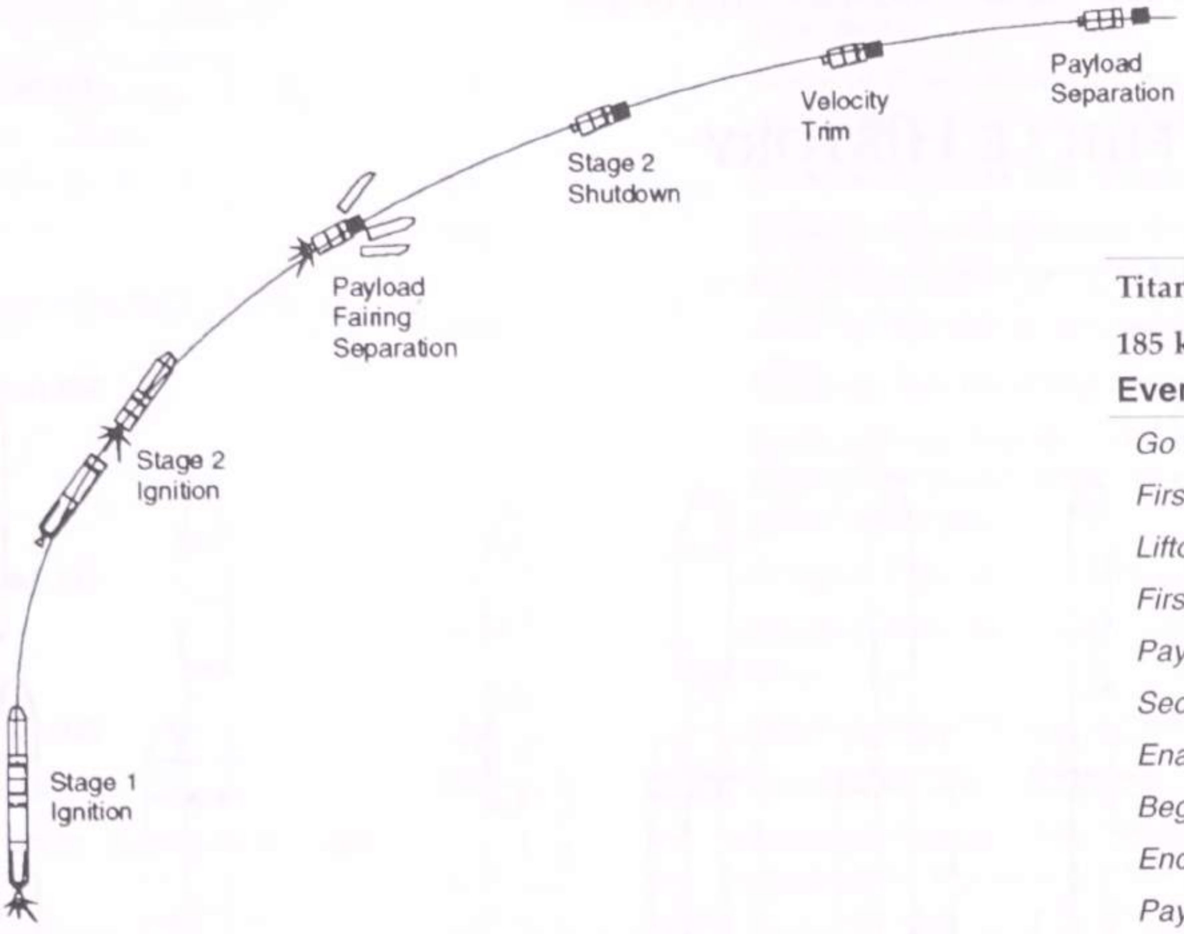
VAFB SLC-40 Launch Facilities Layout



Titan VAFB Launch Pad Layout

Production and Launch Operations

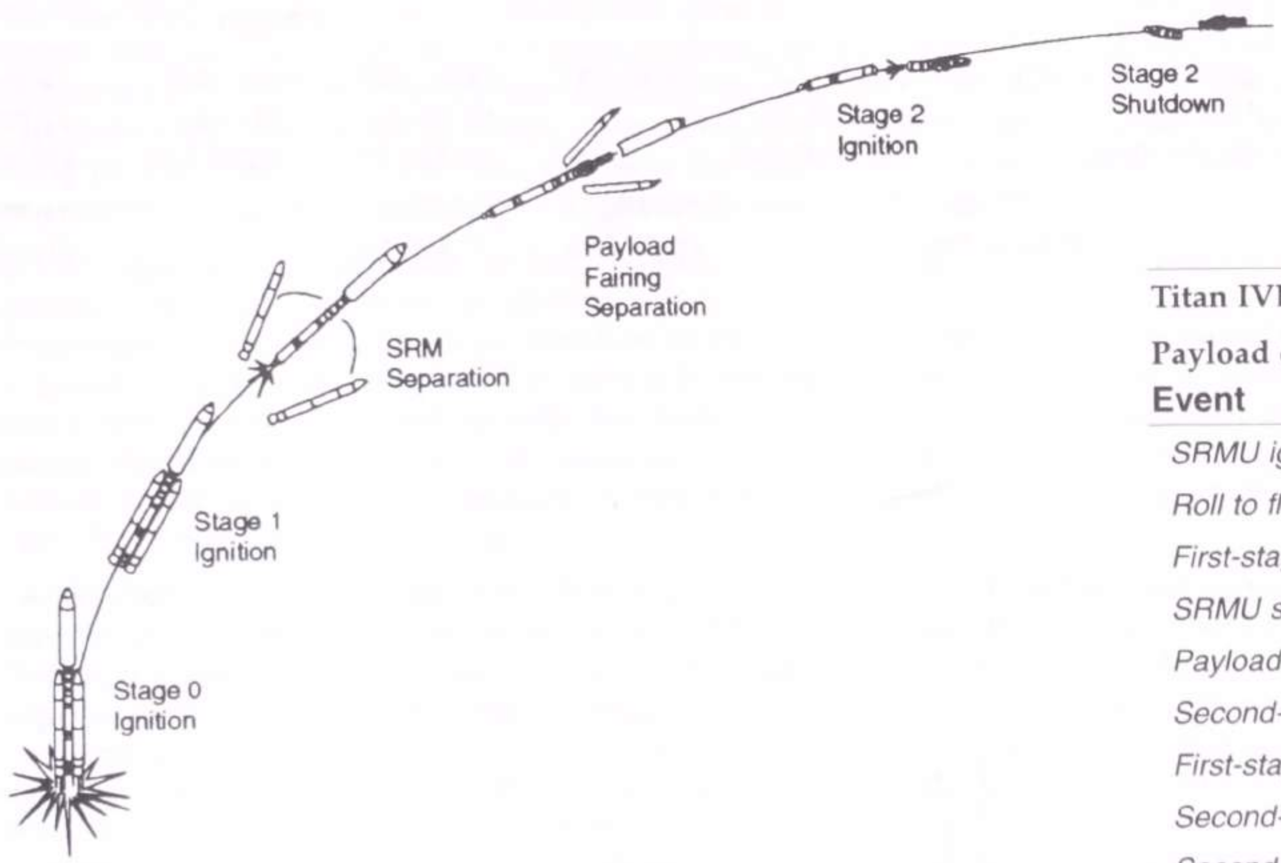
Flight Sequence



Titan II Flight Sequence

185 km (100 nmi) circular orbit, 90-deg inclination

Event	Time, s
Go inertial	−5
First-stage ignition signal	−2
Liftoff	0
First-stage separation	158
Payload fairing separation	211
Second-stage shutdown	333
Enable ACS	349
Begin velocity trim	378
End velocity trim (nominal)	406
Payload separation maneuver	434
Payload separation	519



Titan IVB Flight Sequence

Payload delivered to GEO with IUS

Event	Time, s
SRMU ignition	0
Roll to flight azimuth	6
First-stage ignition	131
SRMU separation	146
Payload fairing separation	306
Second-stage ignition	621
First-stage separation	622
Second-stage shutdown	527
Second-stage separation	536
IUS SRM 1 ignition	4407
IUS RCS 1 burn	4856
IUS SRM 2 ignition	23,588
IUS RCS 1 burn	23,822
Spacecraft separation	24,892

Vehicle Upgrade Plans

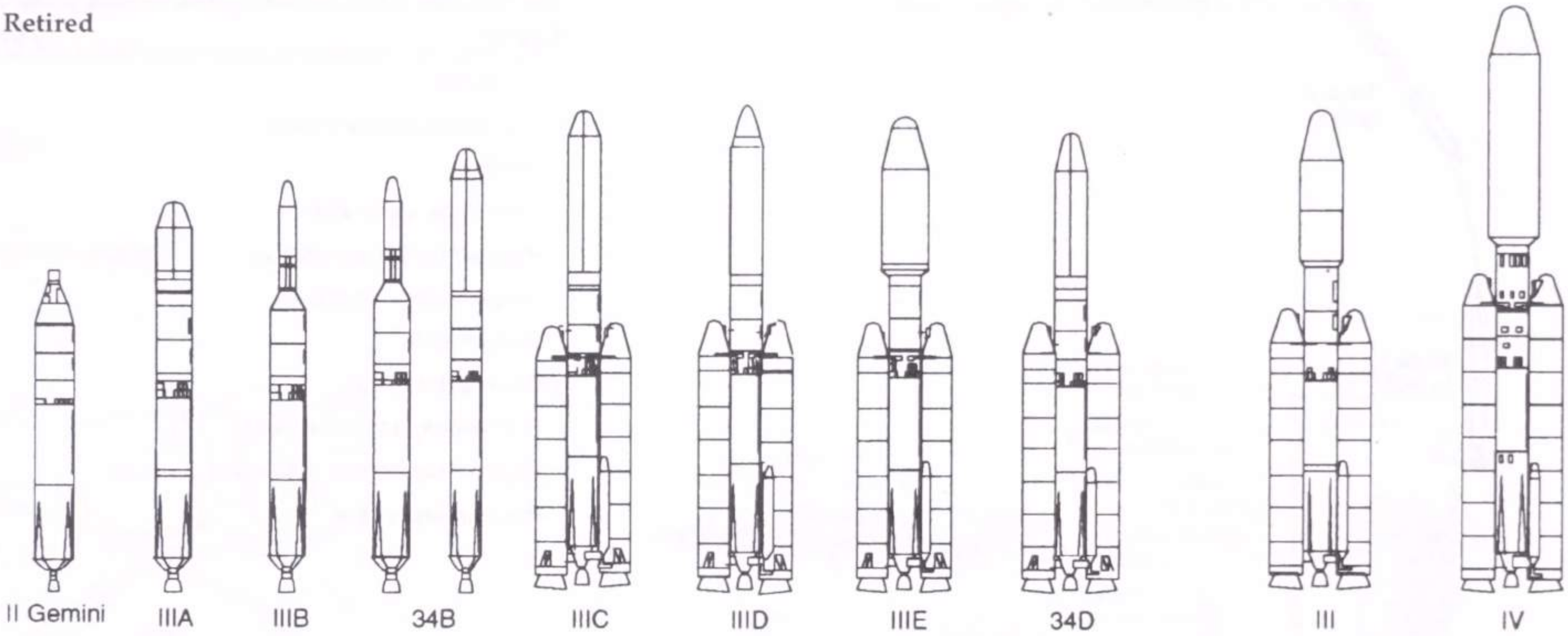
Titan II upgrade plans have been developed for adding solid strap-on boosters similar to Delta II or to enhance the propulsive capability of the attitude control system to increase circular orbit performance. Neither of these proposals are being actively pursued.

No further upgrades of the Titan IVB are planned. The U.S. Air Force will replace the Titan with launch vehicles developed under the EELV program. See the Atlas chapter for information on the Atlas V and the Delta chapter for information on the Delta IV.

Vehicle History

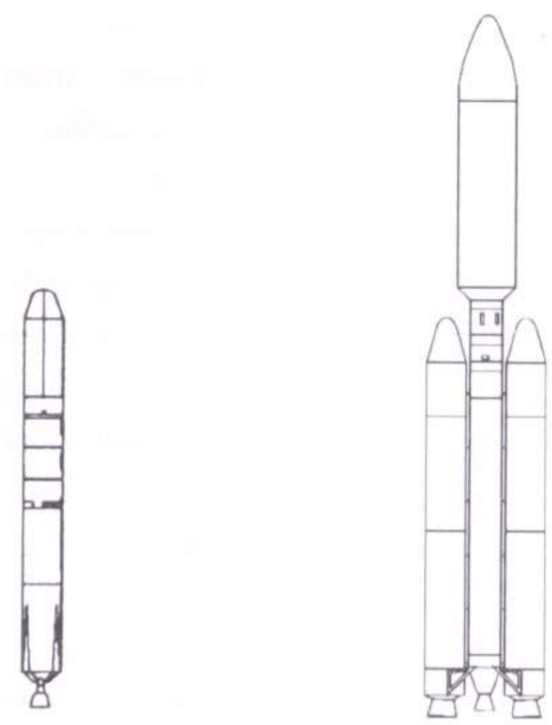
Vehicle Evolution

Retired



Vehicle	**Titan II Gemini**	**Titan IIIA**	**Titan IIIB**	**Titan IIIC**	**Titan IIID**	**Titan IIIE**	**Titan 34B**	**Titan 34D**	**Titan III (Commercial Titan)**	**Titan IVA**
Period of Service	1964–1966	1964–1965	1966–1984	1965–1982	1971–1982	1974–1977	1975–1987	1982–1989	1989–1992	1989–1998
Payload: 185 km (100 nmi), 28.5 deg	3600 kg (7900 lbm)	4100 kg (9000 lbm)		13,300 kg (29,200 lbm)		13,800 kg (30,400 lbm)		15,350 kg (33,800 lbm)	14,515 kg (32,000 lbm)	17,700 kg (39,000 lbm)
Payload: 185 km (100 nmi), 90 deg			3700 kg (8200 lbm)		11,200 kg (24,600 lbm)		3600 kg (7950 lbm)	12,550 kg (27,600 lbm)		14,100 kg (31,100 lbm)

Operational



	Titan II 23G	**Titan IVB**
Period of Service	1988–Present	1997–Present
Payload: 185 km (200 nmi), 28.5 deg		21,680 kg (47,800 lbm)
Payload: 185 km (200 nmi), 90 deg	1905 kg (4200 lbm)	17,600 kg (38,800 lbm)

Vehicle History

Vehicle Description

•**Titan II Gemini**		Titan II ICBM converted to a human-rated space launch vehicle.
•**Titan IIIA**		Similar to Titan II Gemini except stretched first- and second-stages and integral Transtage upper stage.
•**Titan IIIB**	23B	Similar to Titan IIIA except Agena upper stage instead of Transtage, elimination of human-rated systems, new radio guidance, and 1.5-m (5.0-ft) diameter fairing.
	24B	Similar to Titan 23B with stretched first stage.
	33B	Similar to 23B with guidance provided by Agena upper stage inside new 3.0-m (10-ft) payload fairing. Ascent Agena was used for launch only and did not remain attached to payload.
	34B	Similar to 33B with lengthened first stage.
•**Titan IIIC**		Similar to Titan IIIA except 5-segment SRMs and human-rated.
	23C	Similar to initial Titan IIIC, with man-rated systems removed, upgraded first-stage main engine, new digital avionics, blowdown SRM TVC system instead of pressure regulated, simplified Transtage ACS.
•**Titan IIID**	23D	Similar to Titan IIIC except no upper stage.
	34D	Similar to Titan 34B except used 5-1/2-segment SRM. Used either Transtage or IUS upper stage.
•**Titan IIIE**	23E	Similar to Titan IIID except Centaur upper stage and 4.3-m (14-ft) diameter payload fairing.
•**Titan II**	23G	Refurbished Titan II ICBM with 3.0-m (10-ft) diameter payload fairing.
•**Titan III (Commercial Titan)**		Similar to Titan 34D except stretched second stage, single or dual payload carrier, enhanced liquid rocket engines and 4.0-m (13.1-ft) diameter payload fairing.
•**Titan IVA**	45D, E, F, H, and J	Similar to Titan 34D except stretched first and second stages, 7 segment SRM, and 5.1-m (16.7-ft) diameter payload fairing. Performance capability and payload volume compatible with Space Shuttle. Used either IUS, Centaur, or NUS.
•**Titan IVB**	45D, E, F, H, and J	Similar to Titan IVA, with new 3-segment SRMU, new avionics, standardized core configution.

Historical Summary

The Titan family was established in October 1955, when the U.S. Air Force awarded the then Martin Company a contract to build a heavy-duty missile system. It became known as Titan I, the nation's first two-stage ICBM and first underground silo-based ICBM. Titan I served as a backup to the development of the Atlas ICBM. It proved many structural and propulsion techniques that were later incorporated into Titan II. The Titan II was a heavy-duty missile using storable propellants instead of the LOX/kerosene used in Titan I. In addition to being deployed as an ICBM for three decades, and providing the technological legacy for Titan III, Titan II also became a human-rated space booster for NASA's Gemini program. Twelve successful Gemini missions were flown on Titan II through 1966 (11 orbital and 1 suborbital).

In 1961, representatives from both the DoD and NASA made an in-depth study of the nation's space-booster needs. Testimony presented to the group pointed to mission requirements that could not be met by any system then in existence or in the development stage. These missions included U.S. Air Force manned space flights using the DynaSoar spaceplane. In addition, the study suggested that a high order of cost effectiveness and an increase in system reliability could be achieved by using a flexible standardized system instead of a variety of boosters, each modified specifically for the mission at hand. As a result, Titan III was born. The Titan III was to be an outgrowth of propulsion technology developed in both the Titan II and Minuteman ballistic missile programs, combining two liquid stages from the Titan II with large solid strap-on boosters. It was the first space launcher to combine the features of both liquid and solid propulsion systems. A variety of upper stages, including Agena and the new Transtage would be used. On 1 December 1962, the development program began.

Development of the Titan III was rapid. Following the go-ahead, development and fabrication of the booster began immediately at the then Martin Marietta Denver installation. Dredging operations for the Titan III ITL facilities began at CCAFS on 1 February 1963. In the spring of 1963, AC Electronics Division of General Motors Corporation completed the basic design of the airborne and ground components for Titan III's inertial guidance system. The engines for Titan III's new upper stage, Transtage, were fired for the first time in July 1963, by Aerojet-General Corporation of Sacramento, California. The same month, near Sunnyvale, California, Chemical Systems Division of United Technologies Corporation successfully fired the 5.3-MN (1.2-million lbf) thrust solid-propellant booster motors of the type used for Titan IIIC. In May 1964, the U.S. Air Force accepted the first completed Titan IIIA, which would test the core only, without using the strap-on boosters. The maiden flight occurred on 1 September 1964. The two core Titan stages performed correctly, but a pressurization failure in the Transtage caused the engines to shut down before achieving orbit. Three more successful test flights were performed. In October the U.S. Air Force accepted the first Titan IIIC, which consisted of the core plus two five-segment solid strap-on boosters. The maiden flight of the Titan IIIC occurred 18 June 1965.

By this time the DynaSoar spaceplane had been canceled (the program ended in December 1963), and the U.S. Air Force began to redirect its manned space efforts toward the manned orbiting lab (MOL), which would be launched on Titan, and serviced with Titan-launched Gemini capsules. The delayed schedule and heavier mass requirements meant that MOL would need a new Titan, larger than the Titan IIIC. The Titan IIIC vehicles were instead used to launch early communications satellites and Vela nuclear detonation detection spacecraft. Meanwhile, design work began on the Titan IIIM (manned), which would have upgraded main engines on a longer first stage, larger seven-segment solid motors, upgraded avionics, and other improvements. In November 1966, A Titan IIIC flight with a simulated MOL and refurbished NASA Gemini-2 capsule was used to test reentry of the modified capsule, which had an access door in its heatshield to allow astronauts to pass into the MOL. Development of MOL and Titan IIIM continued for several years, including ground tests of the new main engine and seven-segment booster. However, the program was canceled on 10 June 1969. The Titan IIIM was never built, but many of the systems developed for it were transferred to other Titan configurations. The improved -11 version of the main engine first flew in 1970. The stretched first stage and improved avionics were incorporated into several Titan versions a few years later. The seven-segment solid motors were not used in flight until 1989, on the maiden launch of the Titan IVA.

Vehicle History

Meanwhile, Titan was becoming the U.S. Air Force's primary launch vehicle for unmanned satellites, with several versions in use simultaneously. The Titan IIIB was a core-only vehicle similar to Titan IIIA except the Titan IIIB used radio guidance and had an Agena upper stage instead of Titan IIIA's inertial guidance and Transtage upper stage. The Agena upper stage was considered separate from the Titan vehicle, and three failures of the Agena were therefore not counted against Titan. A total of 68 Titan IIIB vehicles flew from VAFB between 1966 and 1987, primarily carrying classified reconnaissance payloads. Heavier military satellites were launched with the Titan IIID. Titan IIID was similar to Titan IIIC with two solid boosters, but lacked the Transtage upper stage. In a change from the usual U.S. Air Force operations, NASA managed the Titan IIIE program, adding a Centaur D-1T upper stage and 4.3-m (14-ft) diameter payload fairing to the Titan IIID configuration. While the Titan IIIE flew only seven times, it made a significant contribution to planetary exploration, launching the Viking 1 and 2 spacecraft to Mars in 1975, and Voyager 1 and 2 spacecraft to the outer planets in 1977. In the late 1970s, the U.S. Air Force ordered an additional batch of Titan III vehicles to provide launch services until the Space Shuttle became available. The new Titan 34D was similar to the Titan IIID, with a stretched core and longer five-and-a-half-segment solid motors. Only seven vehicles were ordered initially, but a total of 15 were eventually built as payloads were removed from the Space Shuttle manifest as a result of schedule slips.

One final Titan III configuration was developed in the late 1980s. Shortly after the *Challenger* accident in 1986 when the U.S. government decided to offload commercial payloads from the Space Shuttle, Martin Marietta announced plans to develop a Titan III commercial launch vehicle on its own funds. The Commercial Titan, designated simply Titan III, was a derivative of the Titan 34D designed to carry dual commercial payloads, similar to the successful Ariane program in Europe. In fact, the dual payload structure was provided by Dornier of Germany, and the fairing was provided by Contraves of Switzerland based on their experience with similar structures for Ariane 4. The commercial Titan was not a successful venture however. Martin Marietta found it difficult to line up two customers for each launch, particularly after a failure on the second flight when Titan failed to deploy the single Intelsat 603 spacecraft. The payload deployment wiring was designed for two payloads, and engineers had not properly coordinated which channel was to carry the signal for the single payload. The spacecraft separated itself from its kick motor, but was stranded in LEO. Space Shuttle astronauts later rescued the spacecraft, connected it to a new kick motor, and boosted it into GTO during the STS-49 Shuttle mission in 1992. The last of four commercial Titans carried NASA's Mars Observer the same year.

In 1984 the DoD called for a launch system that would complement the Space Shuttle and better ensure access to space for certain national security payloads. In 1985, the U.S. Air Force contracted with Martin Marietta Corporation for 10 launch vehicles, originally called the complementary expendable launch vehicle (CELV) or Titan 34D-7 (i.e., Titan 34D with seven-segment motors). This vehicle was eventually named Titan IV. The Titan IV was derived from a Titan 34D, with stretched first and second stages and the seven-segment solid motor originally designed for Titan IIIM. A new 5.1-m (16.7-ft) diameter payload fairing was needed to provide payload volume compatible with the Space Shuttle. The Titan IV program initially started as a short-term program to acquire and launch 10 Titan IV vehicles and Centaur upper stages off one CCAFS pad at a cost of $2 billion. However, after the *Challenger* accident in 1986, the program grew to 41 Titan IV vehicles with a mix of upper stages to eventually be launched off one West and two East Coast pads at an estimated cost of $15 billion. Following the removal of DoD payloads from the Space Shuttle manifest, Titan IV became the DoD's main access to orbit for its heaviest and most important payloads. The first Titan IV launch occurred successfully on 14 June 1989. Even before the first flight, the U.S. Air Force was considering ways to upgrade the Titan IV. In October 1987, Hercules Aerospace (now Alliant Techsystems) was awarded a contract to supply 15 flight sets of upgraded SRBs. For the first time in the Titan program, a supplier other than United Technologies Chemical Systems Division would produce the solid motors. The SRMU motors would have a new propellant formulation, new graphite-composite cases, and hydraulically gimbaled nozzles to replace the LITVC system used since the first Titan IIIC. The development of the SRMU took five years longer than expected, however, because of an explosion during the first test firing. The first flight of the Titan IV with the new motors, now designated Titan IVB, did not occur until 1997. On its second flight in 1997 the Titan IVB Centaur launched NASA's Cassini spacecraft on its way to Saturn.

After a long hiatus, Titan II returned as a space launch vehicle in the late 1980s when retired ICBMs were converted to deliver payloads to orbit. The U.S. Air Force initiated the Titan II Space Launch Vehicle (SLV) at the same time as Titan IV. The Titan II 23G configuration was developed from refurbished Titan II ICBMs with technology and hardware incorporated from the Titan III program. The goal of this USAF-sponsored program is to maximize the use of Titan II ICBM resources, which were deactivated from 1982 to 1987. Deactivated Titan II ICBMs are stored at Davis–Monthan AFB, Arizona, until selected for flight. Fifty-five missiles were made available for refurbishment, although the current USAF contract is for only 14 flights. The Titan II SLV is launched from VAFB and the first flight occurred on 5 September 1988. In 1994, a Titan II launched the Clementine spacecraft to the moon.

VEGA

Courtesy ESA.

The Vega is a small, solid-propellant launch vehicle currently being developed. The program is managed by Italy with participation by other ESA nations.

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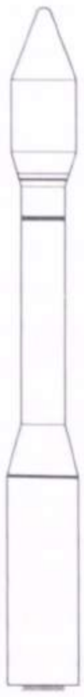
Vega



General Description

Summary

The Vega program is the result of Italy's long-held desire for a national small launch vehicle. After many years of work within Italy, the program was adopted by ESA as a multinational program under Italian leadership. Vega is a small, solid-propellant launch system designed to carry small European science satellites. It will also be marketed commercially.

Status

In development. First launch planned 2006

National Origin

Italy/Europe

Key Organizations

Marketing Organization	Arianespace
Launch Service Provider	Arianespace
Prime Contractor	ELV (Fiat Avio and ASI)

Primary Missions

European small satellites to LEO

Estimated Launch Price

$20 million (ESA, 2002)

Space Port

Launch Site	Guiana Space Center, ELA-1
Location	5.2° N, 52.8° W
Available Inclinations	5.2–100 deg

Performance Summary

200 km (108 nmi), 5.2 deg	?
200 km (108nmi), 90 deg	?
Space Station Orbit, 407 km (220 nmi), 51.6 deg	?
Sun-Sychronous Orbit: 800 km (432 nmi), 98.6 deg	1395 kg (3075 lbm)
GTO: 200 x 35,786 km (108x19,323 nmi), 0 deg	No capability
Geostationary Orbit	No Capability

Flight Record (through 31 December 2003)

Total Orbital Flights	0
Successes	0
Partial Failures	0
Failures	0

Flight Rate

Planned for 2 per year initially, possibly increasing to 4 per year

Nomenclature

The name Vega is a reference to the second brightest star visible from the Northern Hemisphere, and an acronym for Vettore Europeo di Generazione Avanzata (Advanced Generation European Vector). The name Vega was considered in 1973 for Europe's L3S launch vehicle study, but was rejected because it was too similar to the name of a brand of beer. That vehicle was instead named Ariane.

Vega stage designations indicate the approximate propellant mass, in metric tons. For example, the Zefiro 23 has roughly 23 metric tons of propellant.

Cost

To be commercially competitive, the Vega program goal is to offer launch services at a price 15% less than comparable U.S. launch vehicles. The Vega program does not attempt to compete with the price of Russian small launch systems.The target launch price for Vega is therefore $20 million or less per flight. The ELV joint venture has guaranteed that it will produce Vega launch vehicles for Arianespace for $15.5 million. Adding launch service costs will increase the actual price to customers into the $18–20 million range.

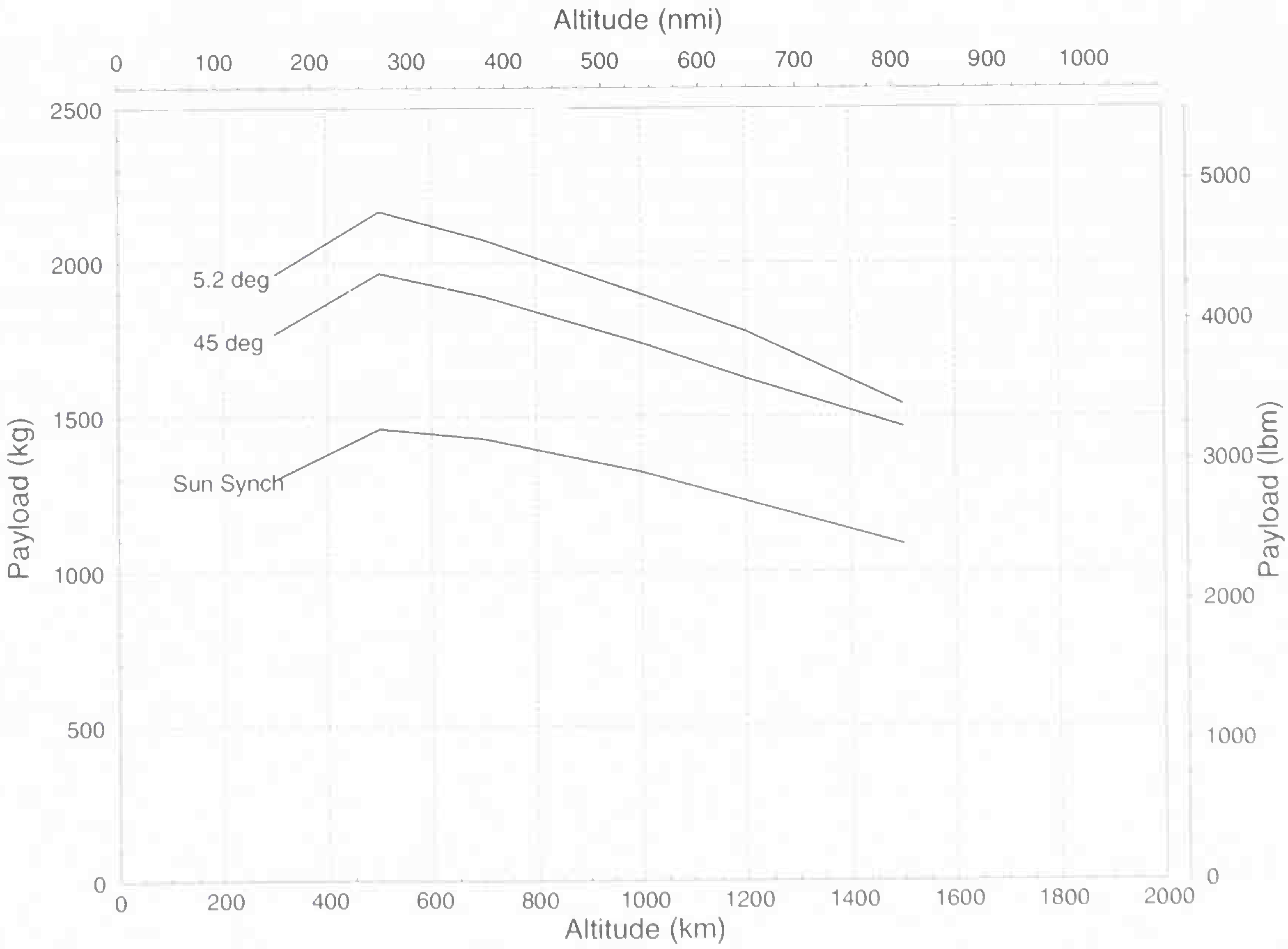
The Vega development program is funded by participating ESA member states: Italy (65%), France (12.4%), Belgium (5.6%), Spain (5%), the Netherlands (3.5%), Switzerland (1.3%), and Sweden (0.8%). The development budget for Vega, approved in December 2000, amounts to 335 million ($329 million). This includes 44 million ($43 million) spent between 1998 and 2000 during the "Step 1" initial studies. The development of the launch vehicle itself amounts to 260 million ($255 million), while 45 million ($44 million) is to be spent on ground infrastructure. A separate but related program for development of the P80 FW first-stage motor was allocated an additional 123 million ($120 million). Fiat Avio is providing 63 million ($62 million) of this funding, France provides 45 million ($44 million), with the remainder provided by the governments of Belgium, Italy, the Netherlands, and Spain. The combined Vega and P80 programs will cost a total of 463 million ($454 million). Substantial funds were also invested by ASI (Agenzia Spaziale Italiana – the Italian Space Agency) and Fiat Avio in precursor Italian programs, as described in the Vehicle History section.

Availability

Vega is scheduled to perform its first launch in mid-2006. It will be marketed commercially by Arianespace. The flight rate will depend on demand, but is expected to range from two to six flights per year, with an average production rate of four per year. Launches can be conducted one month apart. The first demonstration launch is planned to carry a spacecraft for a paying customer, at a reduced price.

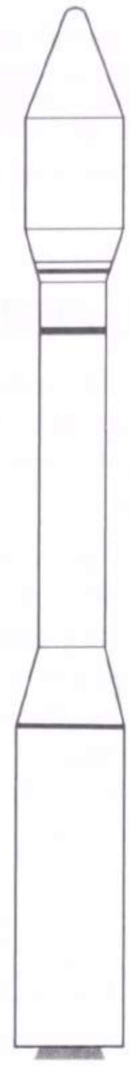
Performance

The Vega program performance requirement is to deliver a 1500 kg (3300 lbm) payload to a 700 km (378 nmi) 90-deg orbit. The vehicle is designed to reach orbits between 300 km and 1500 km (162 to 810 nmi). Performance is reduced for very low orbits, presumably because of the low thrust of the upper stage. For this reason, performance data for a typical 200-km reference orbit is not available.



Vega: LEO Performance

Vehicle Design



Height	30 m (98 ft)
Gross Liftoff Mass	130 t (287 klbm)
Thrust at Liftoff	2700 kN (600 klbf)

Stages

Previously studied Vega configurations included a first stage based on a shortened, single-segment version of the Ariane 5 strap-on booster solid motors. The final design configuration instead uses a more advanced motor to achieve higher payload capability goals and to advance the state of the art in European solid propulsion. The motor is designated the P80 FW, indicating that it has 80 metric tons of solid propellant, with a filament-wound composite case. The P80 FW shares some technical heritage with the Ariane 5 motors, and it is intended that the improvements demonstrated will be applied to future Ariane 5 motor upgrades. The new motor will include a filament-wound carbon composite case, which is lighter and enables higher chamber pressure than the metal case it replaces. The nozzle is designed with a low-torque flex joint and electromechanical gimbal actuation. The design of the nozzle is simplified and uses lower cost materials compared to the Ariane 5 technology baseline. The motor has a finocyl-shaped grain that provides a thrust bucket that reduces the maximum dynamic pressure during flight.

The Zefiro 23 second stage and Zefiro 9 third stage are derivatives of the Zefiro 16 motor developed by Italy beginning in the 1990s. Originally intended as the second stage for Vega, the Zefiro 16 was stretched to create the Zefiro 23 when the Vega performance requirement was increased. Both motors have composite cases, new low-density EPDM insulation, and simplified nozzle designs.

The fourth stage of the Vega is the AVUM (Attitude and Vernier Upper Module). It is divided into two sections, the APM (AVUM Propulsion Module) and AAM (AVUM Avionics Module). The propulsion module is powered by a Yuzhnoye RD-869 bipropellant engine from the Ukraine. This performs orbit injection and serves to balance out the performance dispersions of the solid motors.

Vehicle Design

The design of Vega is still being finalized. The final design may therefore differ from data presented below. Note that stage mass data reflects the mass of the motor only.

	Stage 1 P80 FW	Stage 2 Zefiro 23	Stage 3 Zefiro 9	Stage 4 AVUM
Dimensions				
Length	12.2 m (40.0 ft)	7.5 m (24.6 ft)	3.2 m stage + 0.4 m interstage (10.5 + 1.3 ft)	1.6 m (5.2 ft)
Diameter	3 m (9.8 ft)	1.9 m	1.9 m (6.2 ft)	1.9 m (6.2 ft)
Mass				
Propellant Mass	88.4 t (195 klbf)	23.9 t (52.8 klbm)	8996 kg (19,833 lbm)	250–400 kg (550–880 lbm)
Inert Mass	7408 kg (16,332 lbm)	1863 kg (4107 lbm) (motor only)	731 kg (1612 lbm) (motor only)	336 kg (740 lbm) (motor only)
Gross Mass	95.8 t (211.2 klbm)	25.8 t (26.9 klbm)	9727 kg (21,444 lbm)	586–736 kg (1292–1623 lbm)
Propellant Mass Fraction	0.92	0.93	0.92	0.43–0.54
Structure				
Type	Motor: filament-wound monocoque Interstages: stiffened skin	Motor: filament-wound monocoque Interstages: stiffened skin	Motor: filament-wound monocoque Interstages: stiffened skin	?
Material	Motor: CFRP Interstage: Aluminum	Motor: CFRP Interstage: Aluminum	Motor: CFRP Interstage: Aluminum	Aluminum?
Propulsion				
Motor Designation	P80 FW (Fiat Avio)	Zefiro 23 (Fiat Avio)	Zefiro 9 (Fiat Avio)	RD-869 (Yuzhnoye)
Number of Motors	1 (1 segment)	1 (1 segment)	1 (1 segment)	1
Propellant	HTPB 1912	HTPB 1912	HTPB 1912	UDMH/N2O4
Thrust	2261 kN (508 klbf) average 3062 kN (688 klbf) maximum	900 kN (200 klbf) average 1200 kN (270 klbf) maximum	230 kN (51.7 klbf) average 280 kN (53 klbf) maximum	2.2–2.5 kN (495–560 lbf)
Vacuum Isp	279.4 s	287 s	294 s	317 s
Chamber Pressure	88.6 bar (1285 psi)	95 bar (1378 psi)	67 bar (970 psi)	?
Nozzle Expansion Ratio	16:1	25:1	56:1	?
Propellant Feed System	—	—	—	Pressure-fed
Mixture Ratio (O/F)	—	—	—	?
Throttling Capability	None	None	None	None?
Restart Capability	None	None	None	Multiple
Tank Pressurization	—	—	—	Helium
Attitude Control				
Pitch, Yaw	Nozzle gimbal with electromechanical actuation ±6.5 deg	Nozzle gimbal with electromechanical actuation ±6.5 deg	Nozzle gimbal with electromechanical actuation ±6.5 deg	Six 50-N (11-lbf) cold GN_2 thrusters
Roll	?	?	AVUM cold-gas thrusters	Six 50-N (11-lbf) cold GN_2 thrusters
Staging				
Nominal Burn Time	105 s	71 s	117 s	620 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Command shutdown
Stage Separation	?	?	?	?

Vehicle Design

Attitude Control System

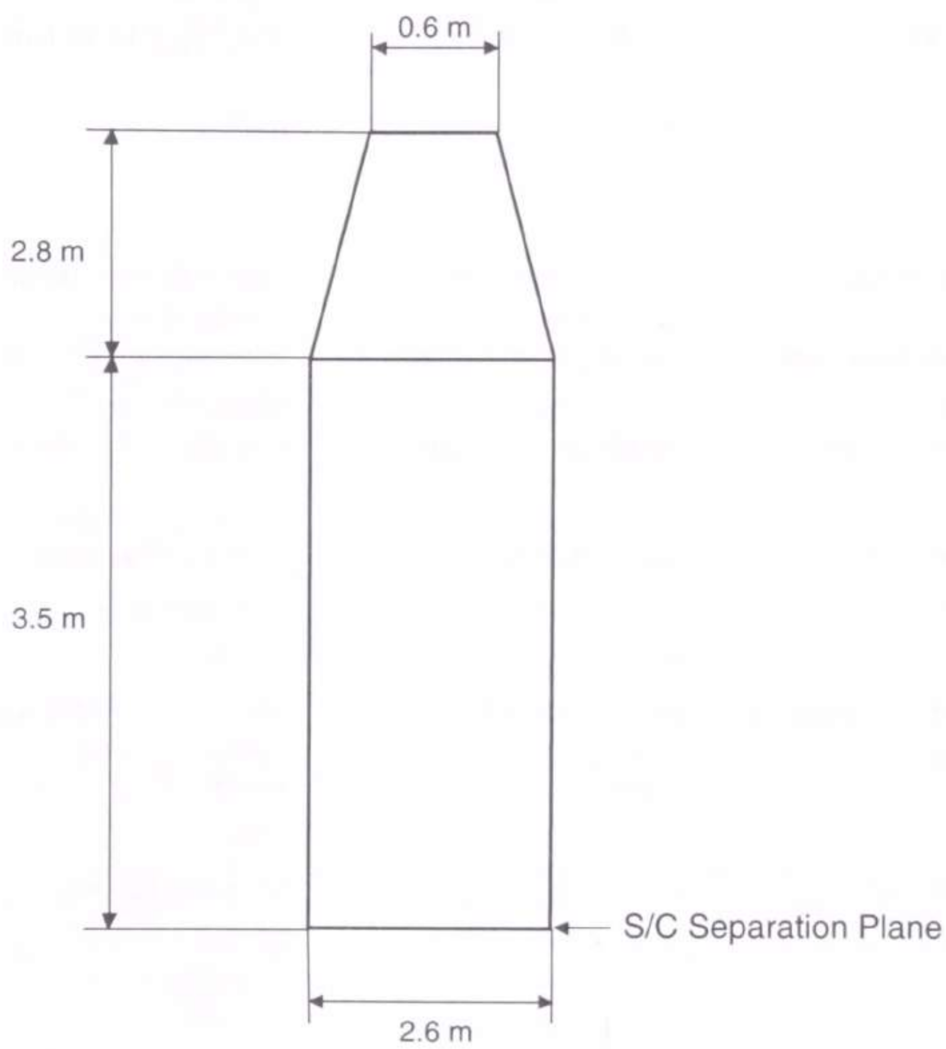
All three solid stages use electromechanically actuated nozzle gimbal systems to control pitch and yaw. The method of roll control, if any, for these stages is unclear. The AVUM stage has six cold-gas thrusters that control its attitude, as well as providing roll control during third-stage flight.

Avionics

The avionics system is being designed to use existing designs where possible, not only to reduce development costs, but also to maintain compatibility with the same ground systems used by Ariane 5. The avionics architecture is divided into four subsystems. The Flight Control Subsystem (FCS) uses an Ariane 5-derived IMU, a flight computer, and thrust vector control avionics. The Telemetry Subsystem (TMS) provides for downlink of vehicle data. The Electrical Power and Distribution Subsystem (EPDS) provides power to the various components. The Safeguard System (SAS) uses Ariane 5 heritage transponders and antennas for tracking, with newly designed components for the destruct functions.

Payload Fairing

After considering a single-piece payload fairing, a two-piece design was selected because it was lighter. Access doors are provided. No specific acoustic attenuation is provided in the standard configuration.



Payload Envelope	
Length	7.5 m (24.6 ft)
Primary Diameter	2.6 m (8.5 ft)
Mass	440 kg (970 lbm)
Sections	2
Structure	Honeycomb
Material	Carbon composite face sheets with aluminum core

Payload Accommodations

Payload Compartment	
Maximum Payload Diameter	2300 mm (90 in.)
Maximum Cylinder Length	3500 mm (138 in.)
Maximum Cone Length	2800 mm (110 in.)
Payload Adapter Interface Diameter	937 mm (36.9 in.)
Payload Integration	
Nominal Mission Schedule Begins	?
Launch Window	
Last Countdown Hold Not Requiring Recycling	?
On-Pad Storage Capability	?
Last Access to Payload	?
Environment	
Maximum Axial Load	5.5 *g*
Maximum Lateral Load	1 *g*
Minimum Lateral/Longitudinal Payload Frequency	?
Maximum Acoustic Level	139 dB at 250 Hz
Overall Sound Pressure Level	142 dB
Maximum Flight Shock	5000 *g* at 8–10 kHz
Maximum Dynamic Pressure on Fairing	?
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 BTU/ft^2/s)
Maximum Pressure Change in Fairing	?
Cleanliness Level in Fairing	Class 100,000
Payload Delivery	
Standard Orbit Injection Accuracy (3 sigma)	±10 km (5.4 nmi) altitude, ±0.05 deg inclination, ±0.1 longitude of ascending node
Attitude Accuracy (3 sigma)	?
Nominal Payload Separation Rate	?
Deployment Rotation Rate Available	?
Loiter Duration in Orbit	?
Maneuvers (Thermal/Collision Avoidance)	Yes, including deorbit
Multiple/Auxiliary Payloads	
Multiple or Comanifest	Multiple manifest options are available
Auxiliary Payloads	Vega can carry up to three microsatellites, with mass <100 kg (220 lbm) and dimensions 600 × 600 × 800 mm (23.6 × 23.6 × 31.5 in.)

Production and Launch Operations

Production

Development and production of Vega is managed by ELV SpA, a company owned jointly by FiatAvio (70%) and ASI (30%). Manufacturing of the motors and other vehicle hardware is performed at the FiatAvio facility in Colleferro, near Rome. The Colleferro plant was built to produce explosives in 1913, and today is a major production center for solid propulsion systems. Test firing of the Zefiro motors is performed at the Italian test range Salto di Quirra on Sardinia.

Launch Facilities

Vega will be launched from the ELA-1 launch complex at the Guiana Space Center (GSG) near Kourou, French Guiana. The ELA-1 complex was used for launches of Ariane 1, 2, and 3, but has been inactive for many years. Engineers also considered using the ELA-3 complex where Ariane 5 is launched, because the Vega first stage has much in common with the Ariane 5 solid booster. However, ELA-1 was selected instead to avoid interference between the two programs. The complex will be fitted with a new mobile integration building called the BIV (Bâtiment d'Intégration Vega). The vehicle will be assembled at the pad inside the BIV, which will then roll away from the pad before launch. Launches will be controlled from the CDL-3 launch control center, which also control Ariane 5 launches.

Vehicle History

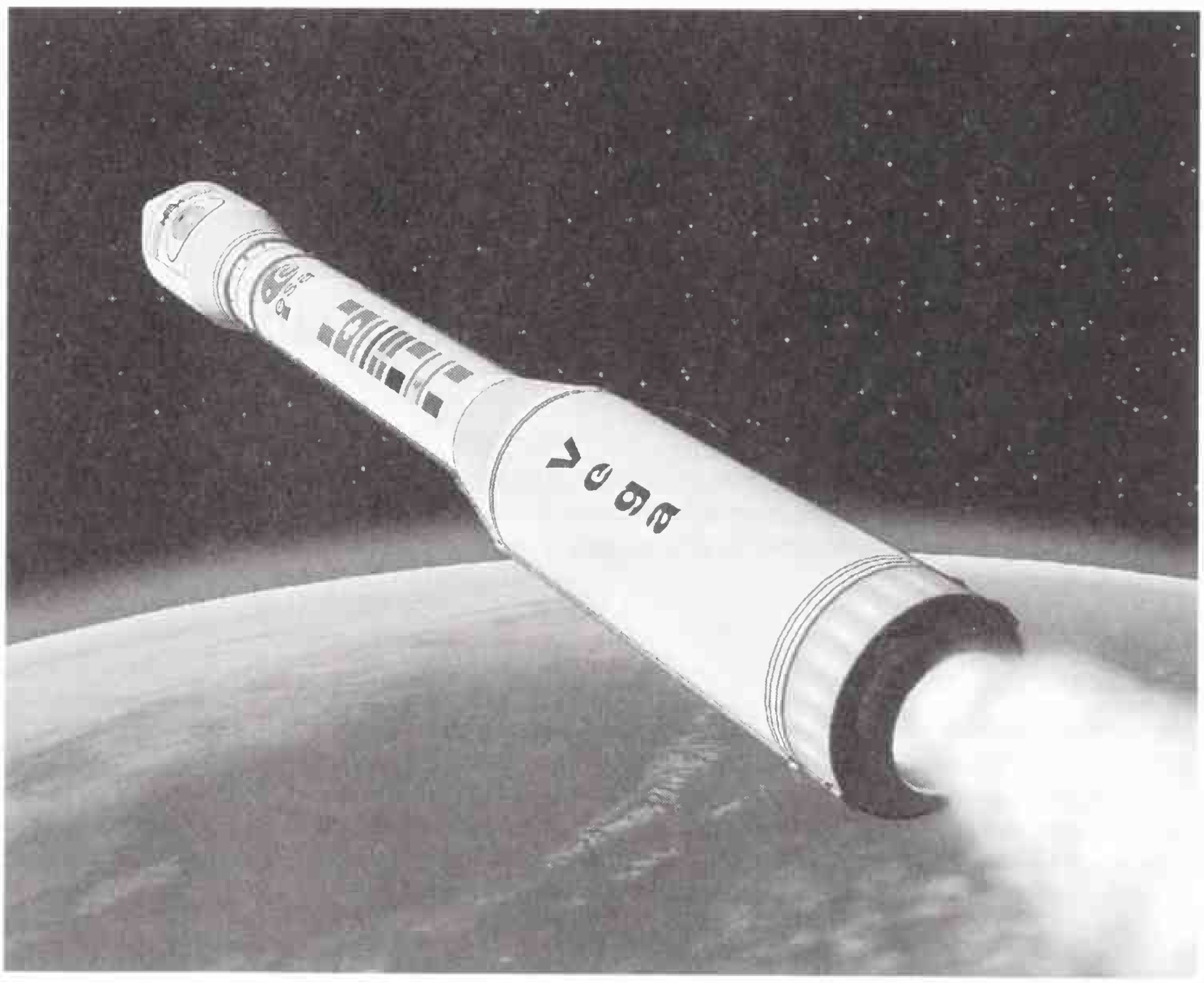
The first precursor to the Vega program was a study initiated in 1988 to provide Italy with a domestic launch system. Between 1967 and 1988, Italy had launched nine U.S.-built Scout launch vehicles from its San Marco platform on the coast of Kenya, in a program managed by the University of Rome. In March 1988 the Italian aerospace company BPD Difesa e Spazio signed an agreement with LTV, the makers of the Scout, to jointly develop an Enhanced Scout with Italian components. In November 1988 BPD submitted a proposal to Italy's space agency, ASI, to build this upgraded derivative, dubbed San Marco Scout. The vehicle would use BPD's Zefiro solid motors, which were similar to the solid strap-on motors the company built for Ariane 4. The proposal was in competition with a University of Rome plan for an enhanced Scout that would be built primarily in the United States. ASI preferred to spend its resources on Italian products, so in December 1991, BPD was awarded an initial $14 million contract to develop its concept, with a potential budget of $75–80 million through first launch in 1995. The new San Marco Scout would be able to lift up to 800 kg (1760 lbs), with a launch price of $10–15 million. Despite the contract award to BPD, the University of Rome effort continued to receive government funding. A political and legal battle between the two programs broke out during 1992. To eliminate redundancy, ASI terminated funding for the University of Rome project. The university filed and won a lawsuit, and ASI was ordered to pay 90 billion lire ($56 million at 1992 exchange rates) to the university for its program. ASI provided only half the amount ordered. The situation was not resolved until late 1993, with the government suspending the University of Rome program and funding only the BPD program, which by this time had been renamed Vega to distinguish it from the university's Scout upgrade program.

Technical progress on Vega was continuing despite the political battles. Four static tests of the Zefiro motor were conducted during 1991 at the Salto di Quirra test facility in Sardinia. A test flight was conducted in March 1992, using a single Zefiro first stage with two dummy strap-ons and a nose cone. The test vehicle failed and had to be destroyed 10 s after liftoff. The design of Vega fluctuated for several years, changing size and configuration multiple times. The Scout approach was dropped in 1993 in favor of a configuration consisting of the 16-ton Zefiro motor, a 5-ton Sesamo second stage, and an IRIS third stage. Strap-on motors were optional. In 1995 this configuration was replaced by one that used Zefiro motors for the first and second stage. In 1997 participation with Ukraine's Yuzhnoye was considered, with Yuzhnoye to provide a liquid upper stage.

Meanwhile, several European programs for small launch vehicles were being promoted during the 1990s. These included Spain's Capricornio, the Anglo-Scandinavian LittLEO, France's SLBM-derived Petit Lanceur de Satellites, and a host of multinational programs including Medium Ariane ("Marianne"), Ariane Light Derivatives (ALD), European Small Launcher (ESL), and Ariane Complementary Launch Vehicle (ACLV). The programs were motivated by a combination of three factors. First, a boom in small satellites was anticipated, which was expected to increase demand for small launch vehicles. Second, it was recognized that Europe would have no practical launcher for small- and medium-sized spacecraft once Ariane 5 became operational and Ariane 4 was phased out. And finally, U.S. companies appeared well positioned to dominate this market segment with new launch vehicles like Pegasus, Taurus, Conestoga, and Athena. However, the combined challenges of pulling together multiple European partners and securing funding while Ariane 5 development was still being paid for prevented any of these programs from solidifying during the 1990s, particularly after more money was allocated to recover from the first Ariane 5 failure.

Recognizing the broader European interest, Fiat Avio's BPD division began in 1997 to work jointly with France's Aerospatiale to study a merger of both companies' small launch vehicle designs. This study would be the genesis of the modern Vega design and program. A new vehicle configuration resulted from the study, using a P85 motor – essentially a single segment of an Ariane 5 solid booster – as its first stage, with the 16-t Zefiro for a second stage, and a new 7-ton motor for a third stage. The vehicle would carry a 1000-kg payload to polar orbit and be ready by 2002. Italy proposed that the program be "Europeanized," and in July 1998 the ESA Council approved a 42 million ($41 million) preliminary design study, as the first step in a possible 310 million ($305 million) program.

Although the vehicle continued to progress technically, a major political battle was brewing for the second time in Vega's history. France, the traditional leader in European launch vehicles, refused to support full funding for development of Vega throughout 1999. The French government argued that Vega was too small, that the anticipated market for such a vehicle was disappearing, and that since the vehicle lacked any new technology, it should be developed by industry, not by government. Italy continued to champion the Vega program, and threatened to withdraw its funding for Ariane 5 upgrades if Vega was not funded. Finally a compromise was reached. A new Vega configuration was developed with higher performance. In addition to larger upper-stage motors, the performance increase would be achieved with a more advanced first-stage motor. The composite-case P80 motor would increase Vega's capability, but also prove out new technologies that could be applied to future upgrades of the Ariane 5 solid motors. France agreed to contribute to the P80 development program, but initially chose not to participate in the development of Vega itself. The new configuration increased Vega performance to 1500 kg (3300 lbm) to a 700-km (378-nmi) sun-synchronous orbit. However, the need to develop new motors also increased the cost of the program and pushed the target date for the first launch to late 2005. The preliminary design review for this configuration was held in July 2001, and the review for the P80 motor was held in late 2001. France elected to rejoin the vehicle development effort in 2001. The critical design review is scheduled for 2004 with the first launch now planned for mid-2006.



VLS AND VLM

Courtesy CTA/IAE.

The first launch of Brazil's VLS-1 occured at the Alcântara Launch Center in November 1997.

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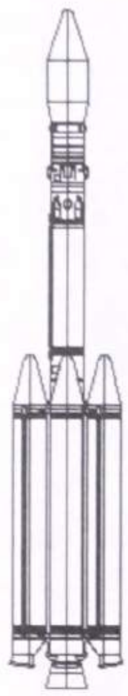
VLS Web site: www.iae.cta.br/AtividadesEspaciais/vls1.html

www.agespacial.gov.br/foglancadores.htm

Alcântara Web site: www.cla.aer.mil.br

VLS-1

General Description



VLS-1

Summary

The VLS-1 (Veiculo Lançador de Satelites) is a small, all-solid propellant launch vehicle that has been under development in Brazil since the early 1980s. The VLS-1 is intended to provide Brazil with independent space access for small environmental payloads and to develop aerospace technology in the country. The first two flights ended in failure, but development of the VLS-1 is continuing, with two more test flights planned.

Status

Flight test. First launch in 1997.

Origin

Brazil

Key Organizations

Marketing Organization	CTA/IAE (Centro Técnico Aeroespacial/Instituto de Aeronáutica e Espaço)
Launch Service Provider	CTA/IAE
Prime Contractor	CTA/IAE

Primary Missions

Small Satellites to LEO

Estimated Launch Price

$8 million (CTA/IAE)

Spaceport

Launch Site	Alcântara Launch Center
Location	44.4° W, 2.3° S
Available Inclinations	4–100 deg

Performance Summary

200 km (108 nmi), 5 deg	380 kg (840 lbm)
200 km (108 nmi), 90 deg	250 kg (550 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	260 kg (570 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	80 kg (175 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 5 deg	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	2
Launch Vehicle Successes	0
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	2

Flight Rate

0–1 per year

VLM

General Description

VLM

Summary

The VLM (Veiculo Lançador de Microsatelites) is a smaller variant of the VLS-1, intended to launch microsatellites at a lower cost.

Status

In development. First launch to be determined.

Origin

Brazil

Key Organizations

Marketing Organization	CTA/IAE
Launch Service Provider	CTA/IAE
Prime Contractor	CTA/IAE

Primary Missions

Microsatellites to LEO

Estimated Launch Cost

$4 million (CTA/IAE)

Spaceport

Launch Sites	Alcântara Launch Center
Location	44.4° W, 2.3° S
Available Inclinations	4–100 deg

Performance Summary

200 km (108 nmi), 5 deg	100 kg (200 lbm)
200 km (108 nmi), 90 deg	67 kg (150 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	67 kg (150 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	18 kg (40 lbm)
GTO: 185×35,786 km (100×19,323 nmi), 5 deg	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

Total Orbital Flights	0

Flight Rate

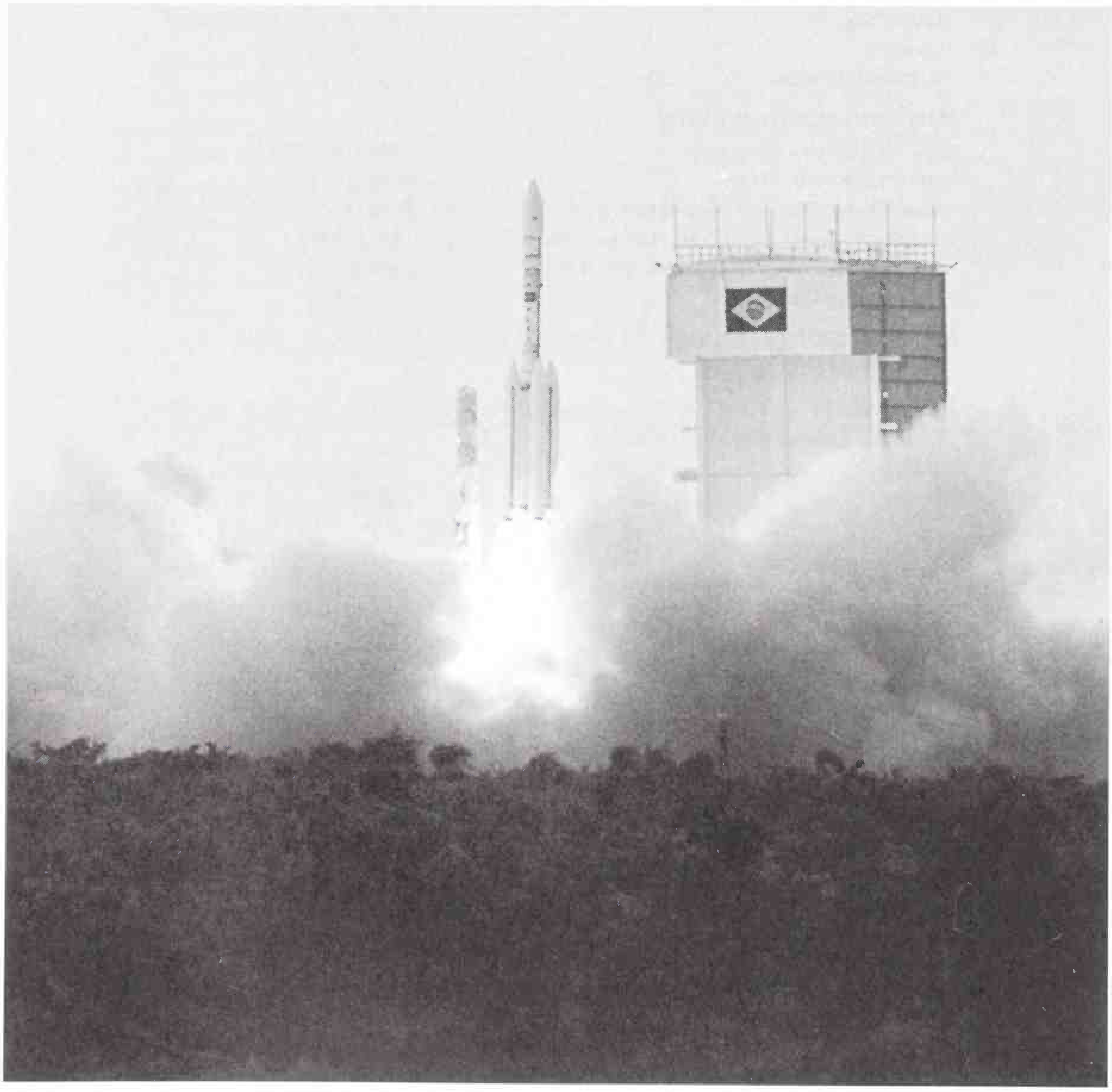
2 per year (planned production capability)

Cost

The estimated cost to National Institute for Space Research (INPE—Instituto Nacional de Pesquisas Espaciais) is $8 million per vehicle for the VLS-1 and $4 million per vehicle for the VLM. This figure does not include annual operating costs. Total development costs of the launch vehicle program are estimated at $250–300 million over nearly 20 years, including investment in launch facilities, laboratories, and other related elements of Brazil's space program.

Availability

The VLS-1 was developed to launch Brazilian national payloads and is not currently being marketed to other customers during its development phase. However, once the vehicle becomes operational it will be available for foreign customers. Within Brazil, users include the Brazilian Space Agency (AEB—Agencia Espacial Brasileira), and the INPE. The first production batch included four vehicles, two of which have flown. Additional vehicles can be produced at a rate of two per year. The VLM program is progressing with ground qualification of the S33 motor used on the fourth stage. Development of VLM will be completed after the VLS-1 is successfully qualified.

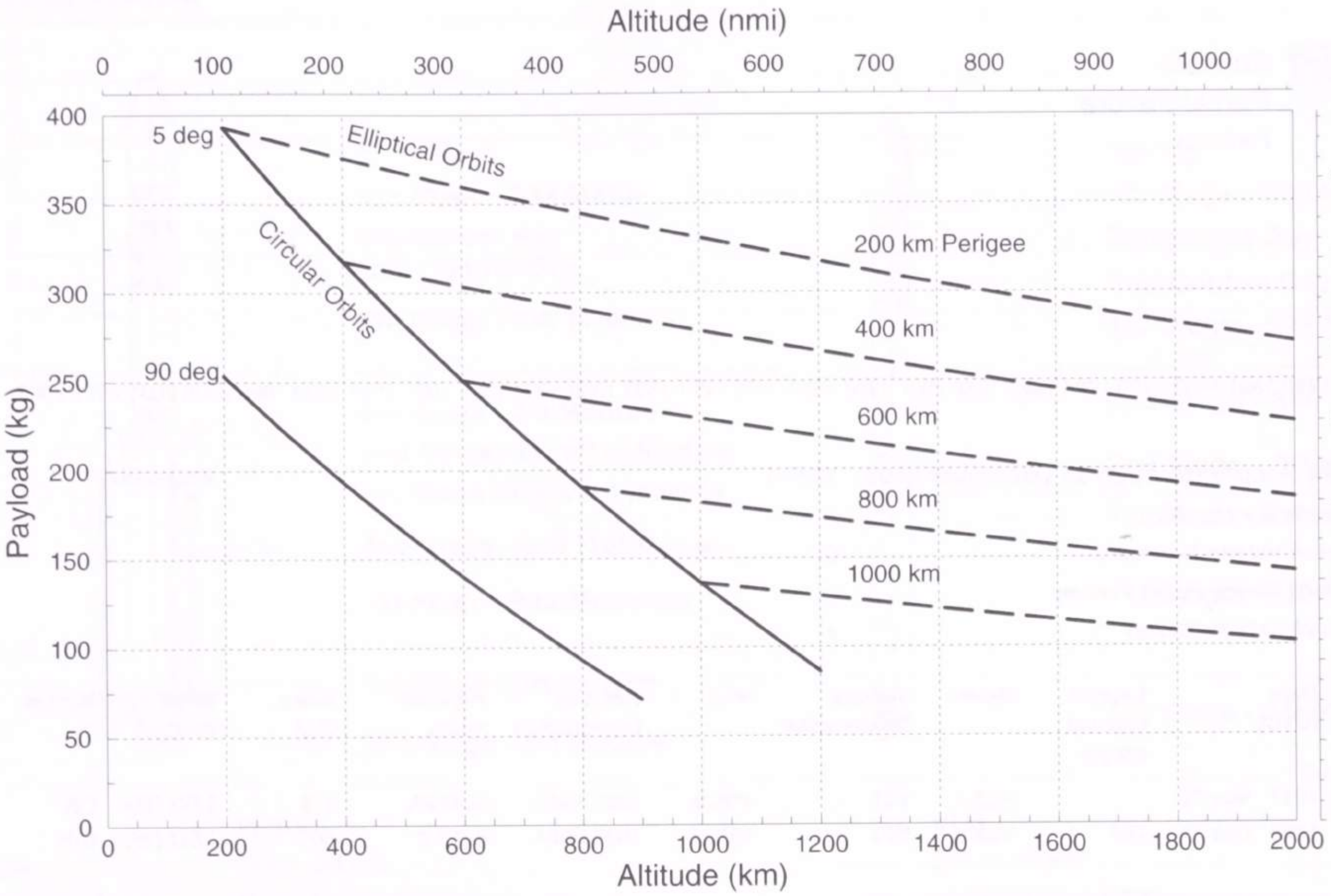


Courtesy CTA/IAE.

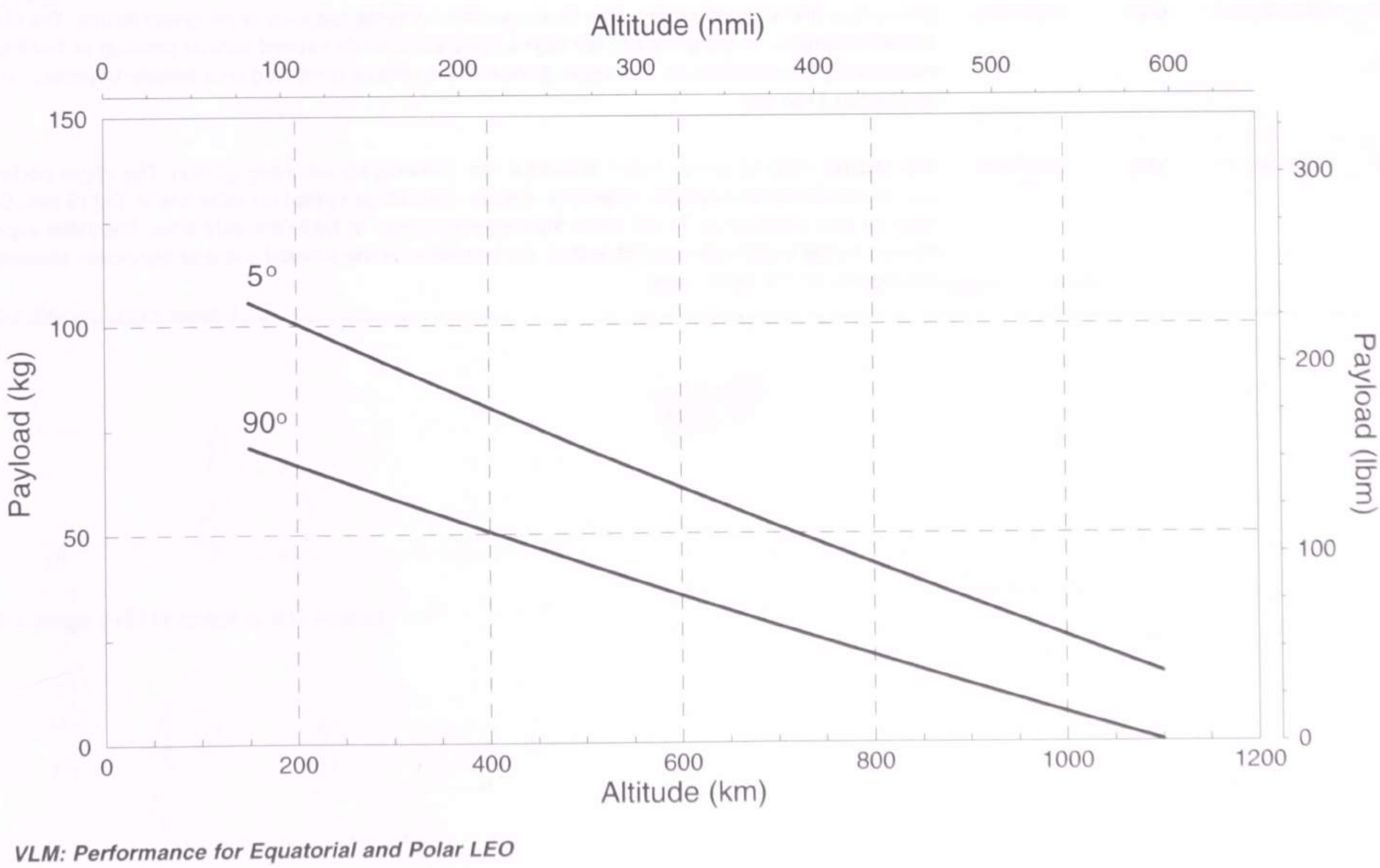
Launch of the first VLS-1, November 1997. Note that SRB on the right side has failed to ignite.

Performance

From the launch complex at Alcântara, rockets can be launched on azimuths from –10 deg to 86 deg, corresponding to inclinations of 4–100 deg.



VLS-1: Performance for Equatorial and Polar LEO



VLM: Performance for Equatorial and Polar LEO

Flight History

Orbital Flights Per Year

Flight Record (through 31 December 2003)	VLS-1
Total Orbital Flights	2
Launch Vehicle Successes	0
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	2

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation	Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
F,T	1	**1997 Nov 02**		**VLS-1**	**V01**	**PVLS**	1997 F03A	SCD 2A	115	LEO (25)	CIV	Brazil
F,T	2	**1999 Dec 11**	769	**VLS 1**	**V02**	**PVLS**	1999 F05A	SACI 2	80	LEO (16)	CIV	Brazil

CLA (Centro de Lançamento de Alcântara) is the Alcântara Launch Complex in northern Brazil

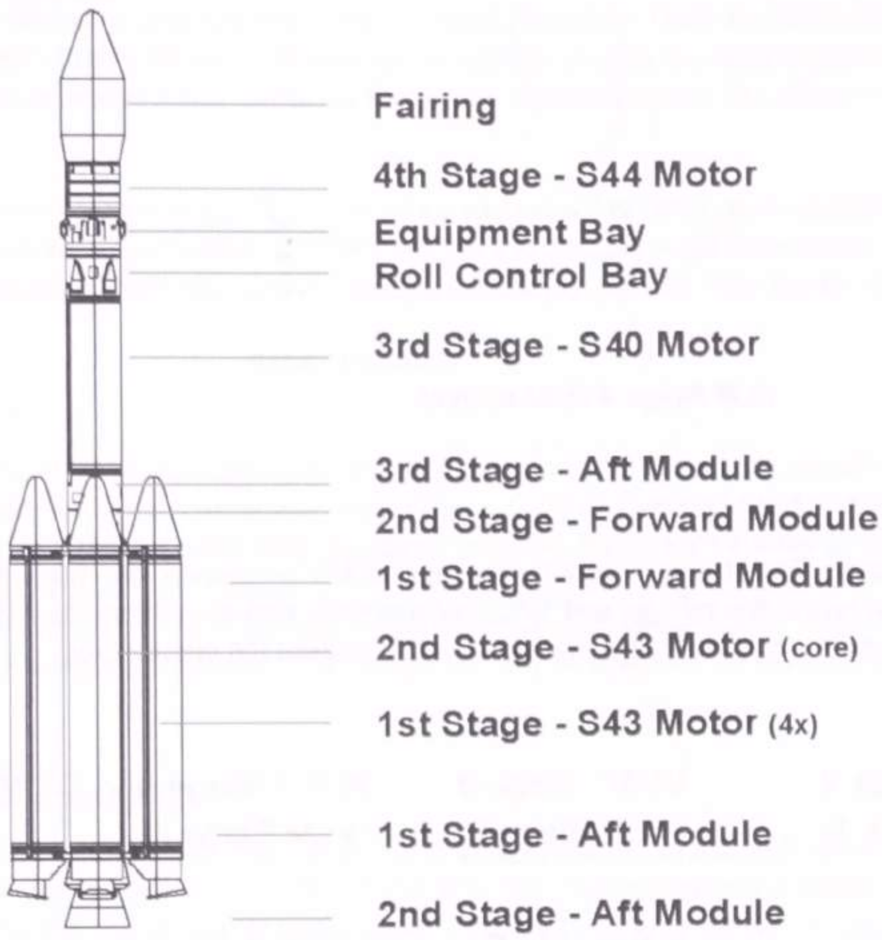
T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

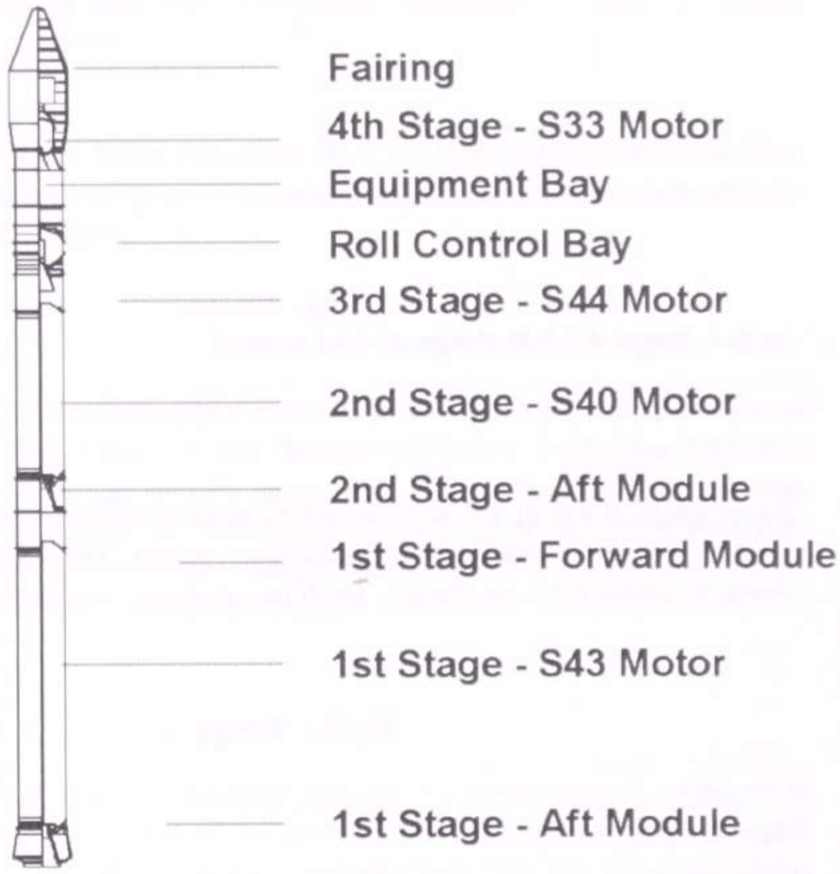
Failure Descriptions:

F	1997 Nov 02	V01	1997 F03	One of four first stage boosters—the D motor—did not ignite because of an igniter failure. The vehicle control system attempted to compensate, but high aerodynamic loads caused vehicle breakup at T+29 followed immediately by autodestruct. The upper portion of the vehicle continued on a ballistic trajectory until command destruct at T+65 sec.
F	1999 Dec 11	V02	1999 F05	The second stage motor exploded at T+55.9 sec, immediately following ignition. The upper portion of the vehicle continued on a ballistic trajectory until the third stage ignited on schedule at T+119 sec. Command destruct was triggered at T+189 when the trajectory began to leave the safe area. The initial explosion was caused by the propellant separating from the insulation at the forward dome of the motor, allowing flame propagation to the motor case.

Vehicle Design

Overall Vehicle





Courtesy CTA/IAE.

	VLS-1	VLM
Height	19.5 m (64.0 ft)	20.0 m (65.6 ft)
Gross Liftoff Mass	49,600 kg (109,400 lbm)	15,900 kg (35,100 lbm)
Thrust at Liftoff	Vacuum: 1198 kN (269 klbf)	Vacuum: 305 kN (68.5 klbf)

Stages

Courtesy CTA/IAE.

VLS-1 Stage 1 (S43 motor)

Courtesy CTA/IAE.

VLS-1 Stage 2 (VLM Stage 1; S43 motor)

Courtesy CTA/IAE.

VLS-1 Stage 3 (VLM Stage 2; S40 motor)

Vehicle Design

Stages

Courtesy CTA/IAE.
VLS-1 Stage 4 (VLM Stage 3; S44 motor)

Courtesy CTA/IAE.
VLM Stage 4 (S33 motor)

Each stage of the VLS-1 is powered by solid propellant motors of the same diameter and basic technology. Brazil developed the experience for these motors with the Sonda series of sounding rockets. Mass and dimensions reflect the motors and TVC systems only, and do not include interstage hardware. In addition to the stages, each rocket has a modular 170-kg (375-lbm) equipment bay, and a 115-kg (250-lbm) roll control bay.

	VLS-1 Stage 1	VLS-1 Stage 2 (VLM Stage 1)	VLS-1 Stage 3 (VLM Stage 2)	VLS-1 Stage 4 (VLM Stage 3)	VLM Stage 4
Dimensions					
Length	7.55 m (24.8 ft)	8.12 m (26.6 ft)	5.09 m (16.7 ft)	1.67 m (5.5 ft)	1.25 m (4.1 ft)
Diameter	1.0 m (3.3 ft)	1.0 m (3.3 ft)	1.0 m (3.3 ft)	1.0 m (3.3 ft)	0.66 m (2.1 ft)
Mass					
Propellant Mass	7180 kg (15,830 lbm)	7180 kg (15,830 lbm)	4430 kg (9770 lbm)	814 kg (1795 lbm)	320 kg (705 lbm)
Inert Mass (motor only)	1245 kg (2744 lbm)	1224 kg (2700 lbm)	929 kg (2050 lbm)	127 kg (280 lbm)	57 kg (125 lbm)
Gross Mass (each)	8425 kg (18,575 lbm)	8404 kg (18,530 lbm)	5359 kg (11,815 lbm)	941 kg (2075 lbm)	377 kg (830 lbm)
Propellant Mass Fraction	0.85	0.85	0.83	0.87	0.85
Structure					
Type	Monocoque	Monocoque	Monocoque	Monocoque	Monocoque
Material	300M Steel	300M Steel	300M Steel	Aramid/Epoxy	Graphite/Epoxy
Propulsion					
Engine Designation	S43	S43	S40	S44	S33
Number of Motors	4	1	1	1	1
Propellant	HTPB	HTPB	HTPB	HTPB	HTPB
Number of Segments	1	1	1	1	1
Average Thrust (vacuum)	305 kN (68.6 klbf)	VLS-1: 327 kN (73.5 klbf) VLM: 305 kN (68.6 klbf)	202 kN (45.4 klbf)	33 kN (7.4 klbf)	20.5 kN (4.6 klbf)
Isp (vacuum)	265 s	VLS-1: 276 s VLM: 265 s	272 s	282 s	275 s
Chamber Pressure	58 bar (840 psi)	58 bar (840 psi)	58 bar (840 psi)	41 bar (595 psi)	38 bar (550 psi)
Nozzle Expansion Ratio	12.9:1	37.2:1	26:1	66.2:1	50.4:1
Attitude Control					
Pitch, Yaw	Nozzle gimbal, 11-deg cant	Nozzle gimbal	Nozzle gimbal	VLS-1: Spin stabilized VLM: Bipropellant RCS	Spin stabilized
Roll	Nozzle gimbal, 11-deg cant	Bipropellant RCS	Bipropellant RCS	VLM: Bipropellant RCS	
Staging					
Nominal Burn Time	60.0 s	60.0 s	58.0 s	66.0 s	43.0 s
Shutdown Process	Burn to depletion	Burn to depletion	Burn to depletion	Burn to depletion	Burn to depletion
Stage Separation	Linear shaped charge	Marman Band	Marman Band	Marman Band	Marman Band

Vehicle Design

Attitude Control System

VLS-1

During first-stage flight, the four gimbaled nozzles of the strap-on boosters provide three-axis attitude control. In the second and third stages, the nozzle performs pitch and yaw control, while a liquid propellant RCS provides roll control. The RCS is contained in the roll control bay on top of the third stage and has roughly 45 kg (100 lbm) of propellant. During the long coast phase between the third- and fourth-stage separation, a cold gas system housed in the equipment bay performs three-axis attitude control. The fourth stage is spin stabilized.

VLM

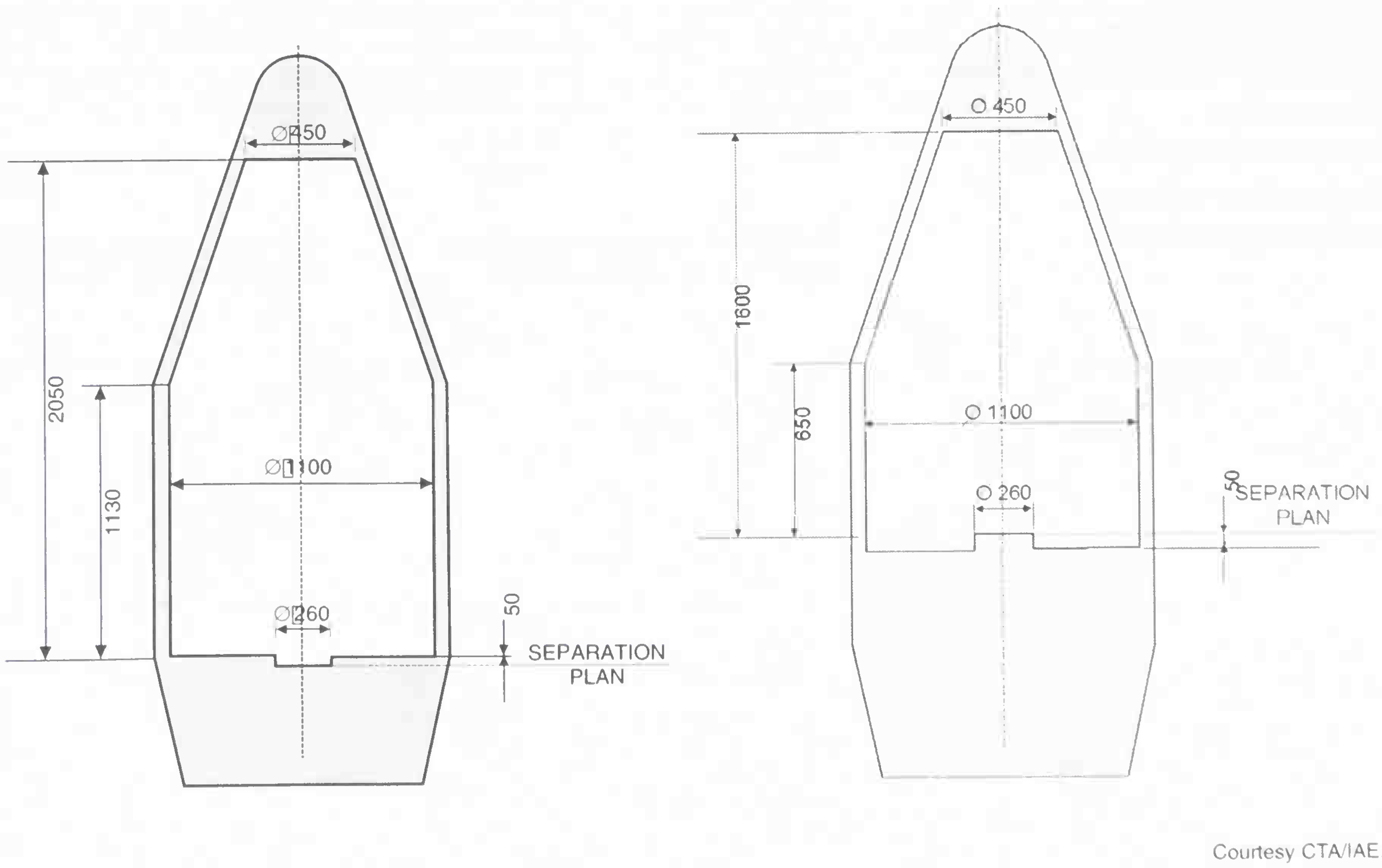
During first- and second-stage flight, the motor nozzle will perform pitch and yaw control, while the liquid propellant RCS will control roll. During the third-stage burn, the liquid-propellant RCS will control the attitude in all three axes. During the long coast phase between the third- and fourth-stage separation, the cold-gas system will perform three-axis attitude control. The fourth stage is spin stabilized.

Avionics

Development of the VLS-1 avionics was one of the most challenging aspects of the development program. Brazil was unable to purchase advanced navigation and control systems from the United States or Europe until it joined the Missile Technology Control Regime (MTCR) in 1995. The IMU used in the VLS and VLM is a three-axis stabilized platform. It provides inertial attitude angles and inertial velocity increments. A separate rate gyro package measures angular rates for stabilization. Navigation processing is performed by the onboard computer. The computer is based on a 68020 processor and performs the functions of guidance, flight control, and sequencing. The equipment bay on top of the third stage houses the IMU, the rate gyro package, and the onboard computer. The equipment bay is released from the vehicle just before the fourth-stage ignition.

Payload Fairing

The same fairing is used for both VLS-1 and VLM missions. In the VLS-1 configuration, it has a usable payload volume of 1.53 m^3 (54 ft^3). In the VLM configuration this is reduced to 1.05 m^3 (37 ft^3). The two halves of the fairing open 3 s after the ignition of the third stage. A two-stage gas-driven piston located at the top of the fairing unlocks a series of clamps located along the bisector seam. Then the piston forces apart the two sections, which rotate on hinges around base pins. After rotating 90 deg, the fairing halves separate from the vehicle. Filtered air conditioning maintains the spacecraft environment at a negotiated temperature and humidity from encapsulation until launch. A 61-pin connector can be provided for monitoring the spacecraft through the T–0 umbilical.



Courtesy CTA/IAE.

	VLS-1	VLM
Length	3.25 m (7.1 ft)	3.25 m (7.1 ft)
Primary Diameter	1.2 m (3.9 ft)	1.2 m (3.9 ft)
Mass	109 kg (240 lbm)	109 kg (240 lbm)
Sections	2	2
Structure	Monocoque	Monocoque
Material	Aluminum	Aluminum

Payload Accommodations

	VLS-1	VLM
Payload Compartment		
Maximum Payload Diameter	1100 mm (43.3 in.)	1100 mm (43.3 in.)
Maximum Cylinder Length	1130 mm (44.5 in.)	650 mm (25.6 in.)
Maximum Cone Length	920 mm (36.2 in.)	920 mm (36.2 in.)
Payload Adapter Interface Diameter	260.4 mm (10.25 in.)	260.4 mm (10.25 in.)
Payload Integration		
Nominal Mission Schedule Begins	T–12 months	T–10 months
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–10 s	T–10 s
On-Pad Storage Capability	Indefinite	Indefinite
Last Access to Payload	T–40 h	T–20 h
Environment		
Maximum Axial Load	9 *g* for 100 kg (220 lbm) payload	12 *g* for 50 kg (110 lbm) payload
Maximum Lateral Load	1 *g* for 100 kg (220 lbm) payload	1 *g* for 50 kg (110 lbm) payload
Minimum Lateral/Longitudinal Payload Frequency	30 Hz	30 Hz
Maximum Acoustic Level	?	?
Overall Sound Pressure Level	140 dB	140 dB
Maximum Flight Shock	?	?
Maximum Dynamic Pressure on Fairing	90 kPa (1900 lbf/ft^2)	70 kPa (1450 lbf/ft^2)
Maximum Aeroheating Rate at Fairing Separation	?	?
Maximum Pressure Change in Fairing	5.8 kPa/s (0.84 psi/s)	?
Cleanliness Level in Fairing	Class 100,000	Class 100,000
Payload Delivery		
Standard Orbit Injection Accuracy (3 sigma)	±50 km (92 nmi), ±0.5 deg	±50 km (92 nmi), ±0.5 deg
Attitude Accuracy (3 sigma)	±2.5 deg	±2.5 deg
Nominal Payload Separation Rate	1 m/s (3.3 ft/s)	1 m/s (3.3 ft/s)
Deployment Rotation Rate Available	180 rpm	180 rpm
Loiter Duration in Orbit	30 s	30 s
Maneuvers (Thermal/Collision Avoidance)	No	No
Comanifest/Auxiliary Payloads		
Comanifest, Auxiliary Payloads	The small payload capacity generally limits the VLS to single payloads.	The small payload capacity generally limits the VLM to single payloads.

Production and Launch Operations

Production

The VLS and VLM are built by the Institute for Aeronautics and Space (IAE), part of the Aerospace Technical Center (CTA) in São José dos Campos. CTA/IAE performs the vehicle design, manufactures most of the electrical systems and solid propellants, does subsystem assembly and vehicle integration and testing, and performs the launch. About 80% of the VLS is produced in by domestic companies in Brazil. The following subcontractors supply components for the rockets.

Subcontractors	VLS/VLM
Altech	composite materials
Abril, Arroyo	structural elements
Cenic	composite motor cases, nozzle parts
Confab	metal motor cases
EDE	hydraulic and pneumatic components
Gespi, Petroflex	propellant ingredients
Neiva	payload fairing, payload adapter
Villares	steel materials

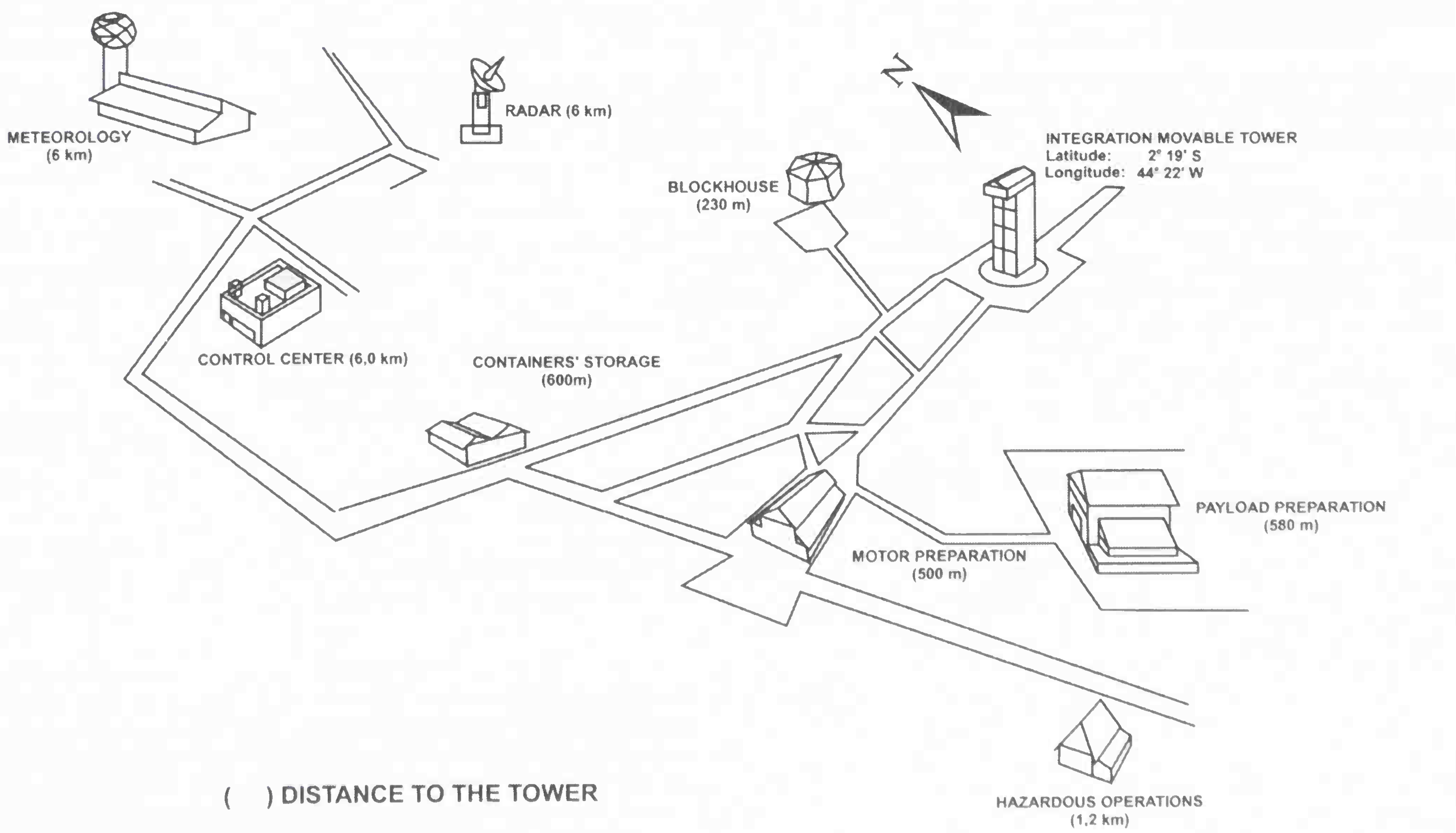
Launch Operations—Alcântara Launch Complex

The Alcântara Launch Center is located on the northern coast of Brazil, covering 620 km^2 (240 mi^2) northwest of the city of São Luis. São Luis has a population of 800,000, with an international airport, hotels, restaurants, and hospitals. Alcântara was built at a cost of roughly $300 million by the Brazilian government and costs approximately $5 million per year to operate.

Facilities at Alcântara are divided into two areas. Rockets are launched from the Setor de Preparação e Lançamento (Preparation and Launching Sector, or SPL). Operations are controlled from an area 6 km away, where the control center, meteorological facilities, and tracking radar are located. The facilties at the SPL include a 560 m2 (6000 ft2) payload processing building called the PPCU, a blockhouse, the LPM rail launcher for sounding rockets, and the primary launch pad, called the Platforma VLS. The VLS pad is made up of a fixed launch mount, an umbilical mast, and a 32.8-m (107-ft) mobile service tower that encloses the vehicle. Launch vehicle buildup begins with the stacking of the solid motors about one month before launch. The RCS is fueled and tested in a separate facility and integrated with the vehicle 20 days before launch. The hydraulic fluid system and nitrogen tanks are loaded and pressurized from the utility room during vehicle assembly. The payload is mated to the vehicle in a class 100,000 clean room at the payload integration level of the mobile tower. An electrical network check is performed after integration of each module, including the satellite. The mobile tower is withdrawn 2.5 h before liftoff.

For several years, Brazil has attempted to commercialize the Alcântara facility to reduce government expenditures. Alcântara is closer to the equator than any other launch facility, at only 2.3° S latitude. Because low latitude launch sites provide easy access to low inclination orbits and more initial velocity, an effect of Earth's rotation, many organizations have considered launching their vehicles from the facility. Current likely candidates include OSC's Pegasus, Israel's Shavit, and Ukraine's Cyclone 4. In April 2000 Brazil signed an accord with the United States that permits U.S. rockets or spacecraft to be launched from Alcântara. In return, safeguards prevent Brazil from gaining access to U.S. technology or using the revenues from U.S. launches to fund missile-related programs. The agreement has not yet been ratified by Brazil's Congress, because of concern over clauses that limit Brazil's rights, for example a ban on customs inspections of spacecraft. As a result of the controversy, a planned Pegasus launch from Alcântara for the C/NOFS satellite has been shifted to the U.S. Army test range at Kwajelein. The United States and Germany have conducted sounding rocket launches from Alcântara.

Launch Facilities



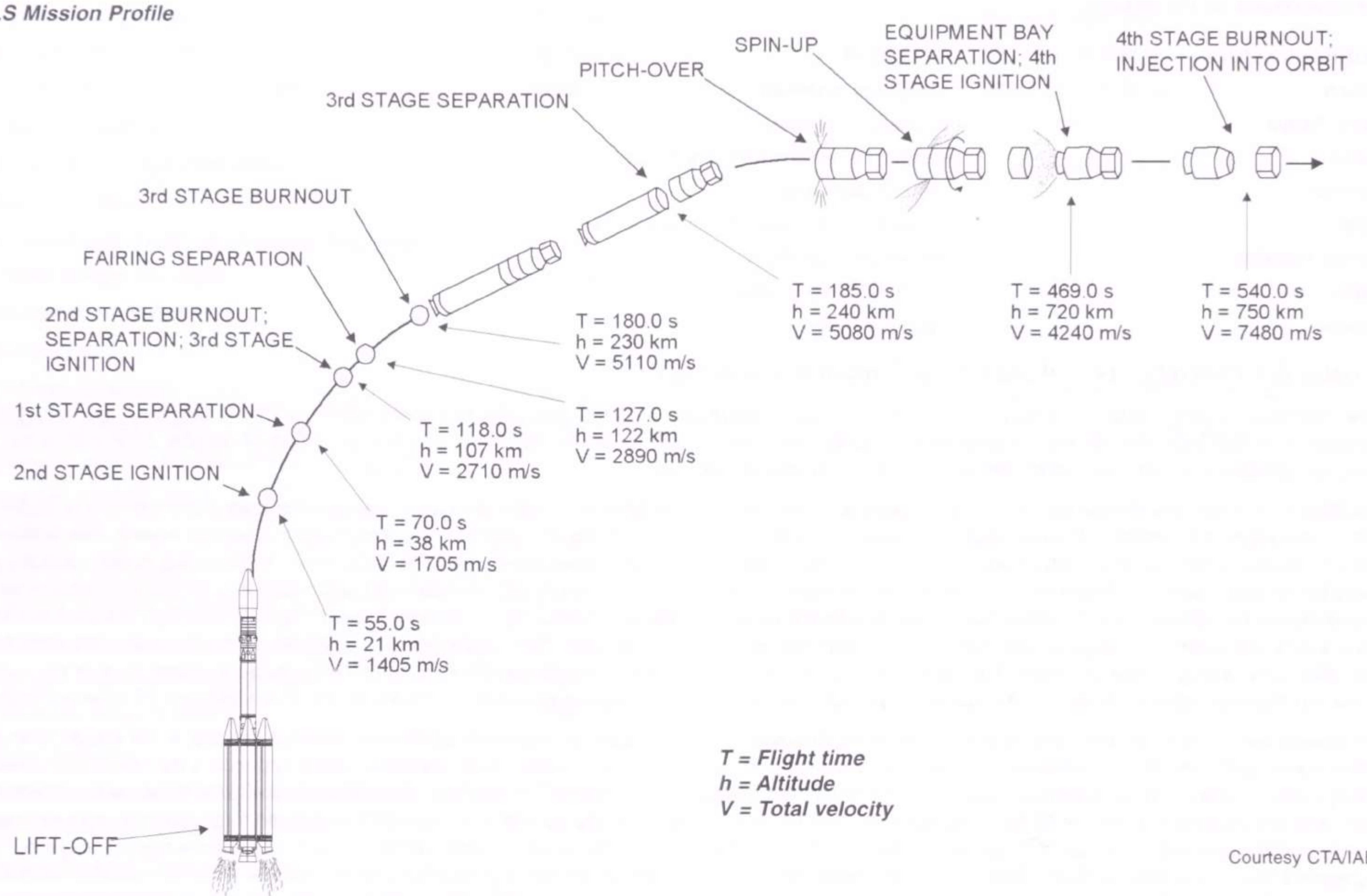
Courtesy CTA/IAE.

Alcântara Launch Center

Production and Launch Operations

Flight Sequence

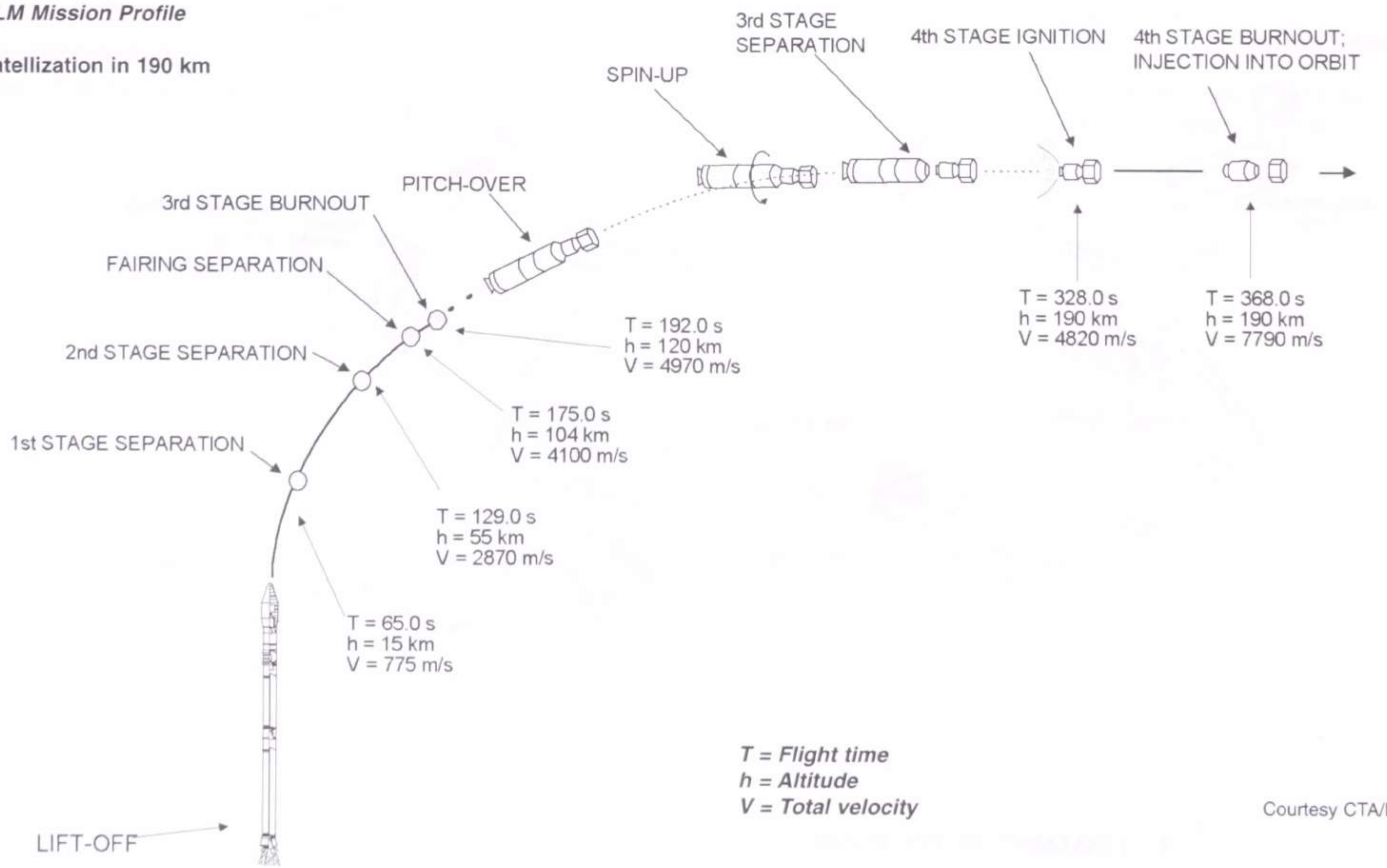
VLS Mission Profile



Courtesy CTA/IAE.

VLM Mission Profile

Satellization in 190 km



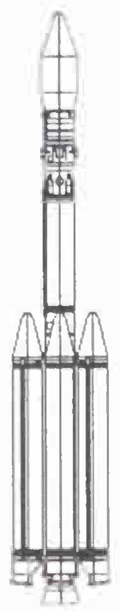
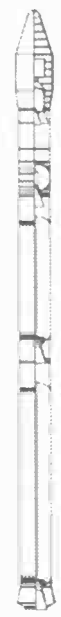
Courtesy CTA/IAE.

Vehicle Upgrade Plans

In addition to the smaller VLM, IAE is developing preliminary designs for a larger vehicle called VLS-2. This vehicle would likely include some liquid propulsion, and would have a payload capacity of up to 2700 kg (5900 lbm) reaching a 200-km (108-nmi), 5-deg circular orbit.

Vehicle History

Vehicle Evolution

	Retired	Operational
Vehicle	**VLS**	**VLM**
		
First Launch	1997	2002
Payload 200 km (108 nmi), 5 deg	380 kg (840 lbm)	100 kg (220 lbm)

Vehicle Description

•VLS-1 Four-stage, all-solid propellant launch vehicle with four S43 motors as the first stage, one S43 motor as the second stage, a S40 motor as the third stage, and a S44 motor as the fourth stage. The first three stages are controlled by nozzle gimbaling whereas the fourth stage is not guided.

•VLM Four-stage, all-solid propellant launch vehicle derived from the core stages of VLS-1. It is composed of a S43 motor as the first stage, S40 motor as the second stage, S44 motor as the third stage,and S33 motor as the fourth. The guidance and control scheme is the same of VLS-1.

Historical Summary

Brazil began development of the Sonda series of sounding rockets in the 1960s. These small rockets were launched from the Barreira do Inferno test range near Natal. The Sonda I, II, III, and IV rockets provided experience building and operating small, multistage solid rockets and testing technologies for future launch vehicles. Sonda IV demonstrated three-axis attitude control, and solid motors large enough for use on a satellite launch vehicle.

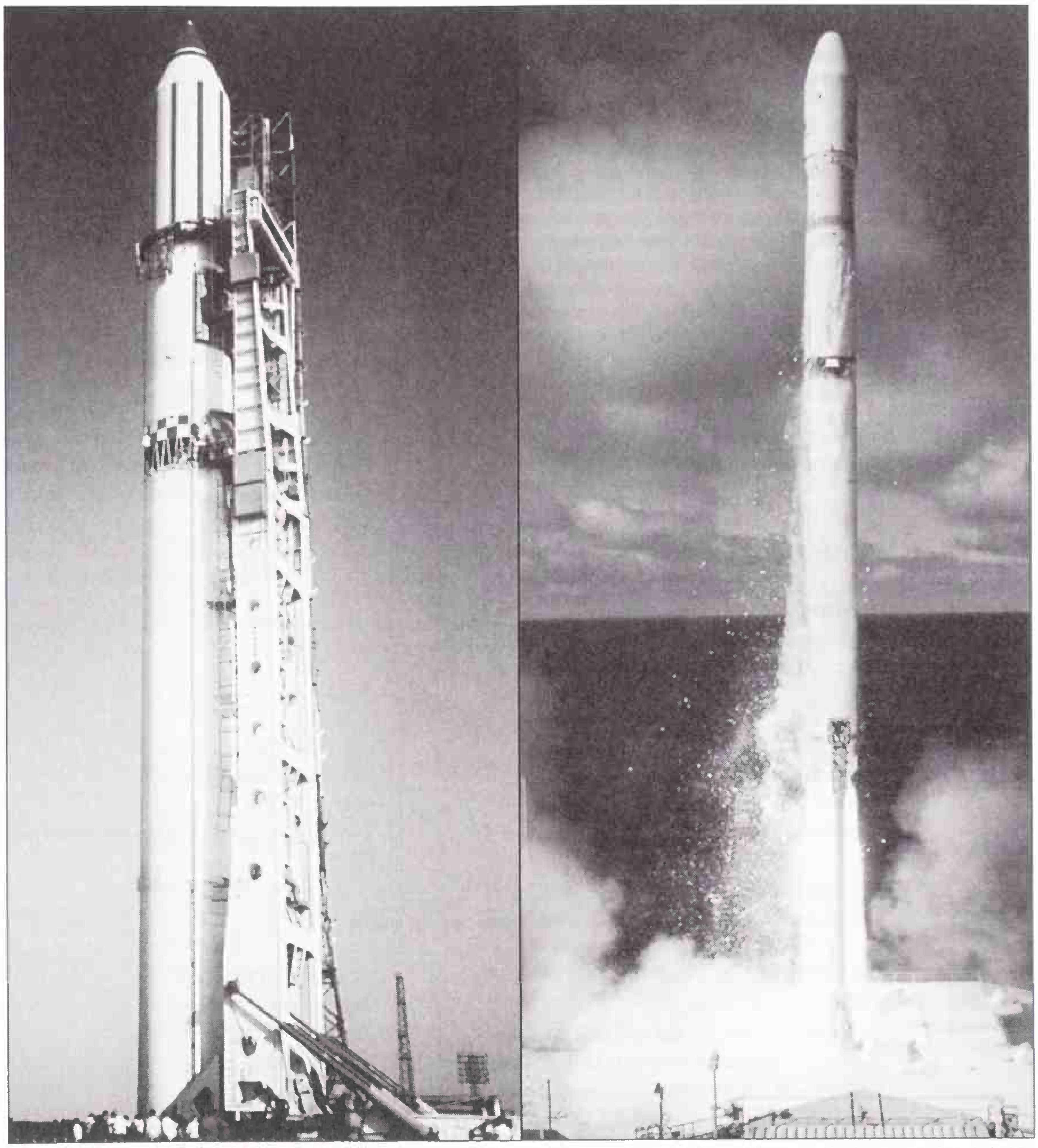
During the 1970s CTA cooperated with the French space agency, CNES, to study a space launch vehicle based on motors from Brazil's Sonda IV and France's Diamant launch vehicle. During 1978 and 1979, the BR2 vehicle was designed, using Ariane's Viking IV engine as a first stage, a Sonda IV motor as the second stage, a new motor for the third stage, and a payload fairing from Britain's Black Arrow. However, becuase these vehicles would be expensive and much of the work would be done outside the country, Brazil rejected French cooperation in favor of a domestic program.

In 1979, the Brazilian Complete Space Mission (MECB—Missão Espacial Completa Brasileira) was drafted by the former Brazilian Commission for Space Activities (COBAE—Comissão Brasileira de Atividades Espaciais) with the purpose of fostering space activities in Brazil. The MECB included three initiatives for the space program: the development of small environmental data-collecting and remote-sensing satellites and their respective ground stations; the development of a small Satellite Launch Vehicle (VLS-1) to launch them; and the construction of a new launch center in Alcântara, Maranhão State. The new site at Alcântara was needed because the proximity of Barreira do Inferno to the growing city of Natal made it unsafe to launch larger rockets there. The Ministry of Aeronautics, represented by the Institute for Aeronautics and Space within the Aerospace Technical Center (CTA/IAE) of São Jose dos Campos, was awarded the development of the VLS-1 launcher and the construction of Alcântara Launch Center. The Ministry of Science and Technology, represented by the National Institute for Space Research (INPE) of São Jose dos Campos, was awarded the construction of satellites and satellite ground stations.

While the VLS-1 project was approved in 1979, budgetary difficulties delayed the effective initiation of the program until 1984 after the Sonda IV sounding rocket's maiden flight. The development program was based on the premise that most of the vehicle should be built with existing domestic technology to save time and cost. Some complex subsystems, such as the IMU and liquid RCS propellant would be procured on the international market to reduce development cost. However, under the Missile Technology Control Regime (MTCR), European and North American countries refused to sell these components to Brazil as a nonsignatory. The combination of export restrictions and domestic budget problems caused long delays in the program.

Progress continued with the May 1989 launch of the VS-40, a new sounding rocket designed to test the S44 motor of the VLS in vacuum conditions. Alcântara became operational in December 1989. In 1993, the first SCD (Satélite de Coleta de Dados) environmental data-gathering satellite was launched on a Pegasus booster. In 1995 Brazil joined the MTCR and became eligible to purchase guidance technology. The first launch of the VLS-1 took place on 2 November 1997, carrying INPE's SCD-2A, another satellite designed to collect data from remote environmental monitoring stations. An ignition failure on one of the first-stage motors caused the flight to be terminated by the automatic destruct system. The second flight, in 1999, also failed. The third VLS-1 vehicle was destroyed in an accident three days before a scheduled launch on 22 August 2003 when a solid motor accidentally ignited. The integration tower was destroyed and 21 technicians were killed.

ZENIT



Photos courtesy KB Yuzhnoye (left) and Sea Launch (right).

The Zenit-2 (left) was the last large launch vehicle developed by the Soviet Union. A two-stage vehicle, it is launched from a highly automated launch facility at Baikonur Cosmodrome to carry intermediate to heavy LEO payloads. The Zenit-3SL (right) is a commercial version of the Zenit operated by the Sea Launch Company, a partnership between U.S., Ukrainian, Russian, and Norwegian companies. The Zenit-3SL incorporates a third stage and is designed to carry heavy spacecraft to GTO or other high-altitude orbits from a floating launch platform on the equator in the Pacific Ocean.

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Web site: www.yuzhnoye.com

Zenit 2/2SLB

General Description

Zenit 2

Summary

The Zenit 2 is a large, two-stage launch vehicle developed in the Soviet Union in the 1970s and early 1980s. Both stages are fueled by LOX and kerosene. Zenit 2 is launched from Baikonur primarily to deliver large military intelligence satellites into LEO. An original purpose of the Zenit-2 was to quickly reconstitute military intelligence satellites in the event of a major conflict within the orbital period of an opponent's surveillance satellite (1.5 h). Accordingly the system was designed with a rapid, automated launch capability that is unique for launch vehicles of this size. Plans to man-rate the Zenit for launches to Mir were later dropped with the demise of the Soviet Union. Although the Zenit 2 is made in Ukraine, the Russian government has continued to use it occasionally. The Zenit 2 has been marketed for commercial missions since the mid 1990s. Since 2003, a slightly modified version known as Zenit-2SLB has been offered for commercial missions from Baikonur. The Zenit-2SLB incorporates avionics and other changes originally made for Sea Launch. Sea Launch is also considering launches of a two-stage Zenit, known as Zenit-2S, from the Sea Launch platform.

Status

Operational. First launch in 1985.

Origin

Ukraine and Russia

Key Organizations

Marketing Organizations	Boeing Launch Services (Commercial payloads) SDO Yuzhnoye (CIS government payloads)
Launch Service Provider	SDO Yuzhnoye
Prime Contractors	SDO Yuzhnoye (Design) PO Yuzhmash (Manufacturing)

Primary Missions

Large satellites or groups of satellites to LEO

Estimated Launch Price

Unknown

Spaceport

Launch Site	Baikonur LC 45L
Location	45.9° N, 63.7° E
Available Inclinations	51.4, 63.9, 88.1, 98.8 deg directly, others require dogleg

Performance Summary

Performance values shown below do not include a payload adapter.

200 km (108 nmi), 51.4 deg	13,920 kg (30,700 lbm)
200 km (108nmi), 90 deg	11,200 kg (24,700 lbm)
Space Station Orbit: 407 km (220 nmi), 51.6 deg	11,900 kg (26,250 lbm)
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	4900 kg (10,800 lbm)
GTO: 200×35,786 km (108×19,323 nmi), 0 deg	No capability
Geostationary Orbit	No capability

Flight Record (through 31 December 2003)

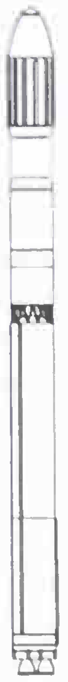
Total Orbital Flights	33
Launch Vehicle Successes	28
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	5

Flight Rate

0–2 per year

Zenit 3SL/3SLB

General Description



Zenit 3SL

Summary

The Zenit 3SL is a three-stage configuration offered for commercial launches by the Sea Launch Company. Sea Launch is an international joint venture led by Boeing. Boeing provides the payload accomodations. SDO Yuzhnoye and PO Yuzhmash of Ukraine provide the two Zenit stages. RSC Energia of Russia provides the Block DM-SL upper stage. The Kvaerner Group of London provided the two vessels. The vehicle is launched from a seagoing launch platform (LP). The launch platform is the most distinctive aspect of the project. Launch control and tracking is performed on the Assembly and Command ship (ACS), which also travels to the launch site. Payload processing and launch vehicle integration are performed at the home port in Long Beach, California. The launch vehicle is assembled on the ACS, then transferred onto the LP, which sails to the equator in the central Pacific where it partially submerges to provide stability. The Zenit 3SL is then raised into vertical position, fueled, and launched using the automated system originally developed for the Zenit 2. The equatorial launch location is well suited for launching commercial satellites into GTO by eliminating plane changes to GEO. Sea Launch has recently begun marketing launches of the "Land Launch" Zenit 3SLB from the existing Zenit launch complex at Baikonur. The Zenit 3SLB uses payload accommodations provided by Russian companies, in place of the Zenit-3SL payload accommodations that are provided by Boeing. This is the main difference between the two configurations. Performance to GTO is reduced by the change in launch site from the equator to Baikonur.

Status

Operational. Zenit 3SL: First launch in 1999.

Origin

Ukraine, Russia, United States, and Norway

Key Organizations

Marketing Organizations	Boeing Commercial Space Company
Launch Service Provider	Sea Launch Company, LDC
Prime Contractors	SDO Yuzhnoye PO Yuzhmash RSC Energia Boeing Commercial Space Company Barber Moss Ship Management AS

Primary Missions

Large payloads to GTO

Estimated Launch Price

Prices negotiable

Spaceport

	Sea Launch	Zenit 3SLB
Launch Site	*Odyssey* launch platform	Baikonur LC 45L
Location	Mobile, typically launched from 0° N, 154° W	45.9° N, 63.7° E
Available Inclinations	All	51.4, 63.9, 88.1, 98.8 deg directly, others require dogleg

Performance Summary

Performance reflects the separated spacecraft mass, assuming a 100-kg payload adapter.

	Sea Launch	Zenit 3SLB
200 km (108 nmi), 51.4 deg	Nonstandard mission, contact Sea Launch	Nonstandard mission, contact Sea Launch
200 km (108 nmi), 90 deg	Nonstandard mission	Nonstandard mission
Space Station Orbit: 407 km (220 nmi), 51.6 deg	Nonstandard mission	Nonstandard mission
Sun-Synchronous Orbit: 800 km (432 nmi), 98.6 deg	Nonstandard mission	Nonstandard mission
GTO: 200 x 35,786 km (108 x 19,323 nmi), 0 deg	6066 kg (13,373 lbm)	—
4,100 x 35,786 km (2,214 x 19,323 nmi), 23.2 deg	—	3500 kg (7716 lbm)
Geostationary Orbit	1700 kg (3750 lbm)	1500 kg (3,307 lmb)

Flight Record (through 31 December 2003)

Total Orbital Flights	11
Launch Vehicle Successes	10
Launch Vehicle Partial Failures	0
Launch Vehicle Failures	1

Flight Rate

1–3 per year

NOMENCLATURE

Zenit

The Zenit was the first launch vehicle to break the old Soviet tradition of naming a launch vehicle after its first payload, although there have been Soviet reconnaissance satellites named Zenit, which used other launch vehicles. Instead, the name Zenit was selected by SDO Yuzhnoye, and translates to zenith, referring to a high point or culmination. The Zenit 2 designation refers to the basic two-stage configuration. (There was not a Zenit 1.) The Zenit 3 is a three-stage vehicle developed specifically for Sea Launch using the Block DM-SL upper stage.

Block DM-SL

The upper stage of the Zenit 3SL is the Block DM-SL. The term "Blok" means stage. Because the stage was originally designed as the fifth stage of the N-1 lunar launch vehicle, it is given the letter D, the fifth letter of the Cyrillic alphabet. The Block DM, which first flew in 1974, is a modernized version of the stage, the primary change being the addition of a toroidal avionics compartment. At least 10 versions of the Block DM exist, eight of which have been used on Proton. The Sea Launch configuration is designated the Block DM-SL to distinguish it from the Proton versions. The Land Launch configuration is designated the Block DM-SLB.

COST

No information is available on the current price or cost of Zenit 2 launches, because there have been no recent launch contracts announced in the market segment addressed by Zenit 2. In 1999 the FAA estimated launch prices were in the range of $35–50 million based on the contract price for one Globalstar launch.

The satellite radio company Sirius disclosed in the late 1990s that it was offered a price of $90 million by Sea Launch for a launch in 2000. Sea Launch officials told *Satellite News* in 1999 that typical prices were between $70–$90 million at that time.

Boeing reports some financial information on its share of the Sea Launch Company in its annual reports to the U.S. Securities and Exchange Commission, which can be used to infer the approximate magnitude of development and operating costs. At the end of 2002 Boeing had a total exposure of $621 million, consisting of $335 million in guarantees to Sea Launch creditors, $33 million in performance guarantees to a Sea Launch customer, and $253 million exposure in accounts receivable and inventory. The Kvarner Group disclosed in 2001 that it had $334 million in exposure in Sea Launch and had already written off its $46 million in equity.

AVAILABILITY

Launch services using the Zenit 3SL are provided commercially by Sea Launch Company. Sea Launch Company has subcontracted with Boeing Launch Service to perform marketing services. Sea Launch Company also has worldwide rights to market commercial launches of the Zenit 2, 2SLB, and 3SLB.

In recent years, the Zenit 2 has performed one launch per year. The maximum available annual launch rate is about 12 flights per year, limited by production capacity. If necessary, two Zenit launch vehicles could be launched from the same launch pad less than 90 min apart, however no more than three launches per month are possible because of limitations of other facilities.

To date the Zenit 3SL has launched no more than three times in one year. The maximum available launch rate is six to seven flights per year. This is limited by the time required to process the spacecraft and launch vehicle in Long Beach, and the 10–12 day one-way travel time required to move the launch platform from Long Beach to the equatorial launch location.

Performance

Zenit 2

The Zenit 2 is launched from the Baikonur Cosmodrome in Kazakhstan. Because Baikonur is a landlocked launch site, specific drop zones approximately 800–1925 km downrange are reserved for the impact of the first stage and payload fairing. Zenit is therefore constrained to fly along launch azimuths that overfly these zones. The bend in the Zenit 2 performance curves reflects the transition between two classes of trajectories. The available launch azimuths and inclinations are provided in the table. Zenit can reach other inclinations using a dogleg maneuver during the second-stage main-engine burn. For example, many Zenit launches have delivered payloads to an inclination of 71 deg, using either the 35-deg or the 178.8-deg initial launch azimuth. Because the dogleg maneuver imposes a performance penalty, it is not valid to interpolate between the inclinations shown. The performance shown below does not account for a mission specific payload adapter.

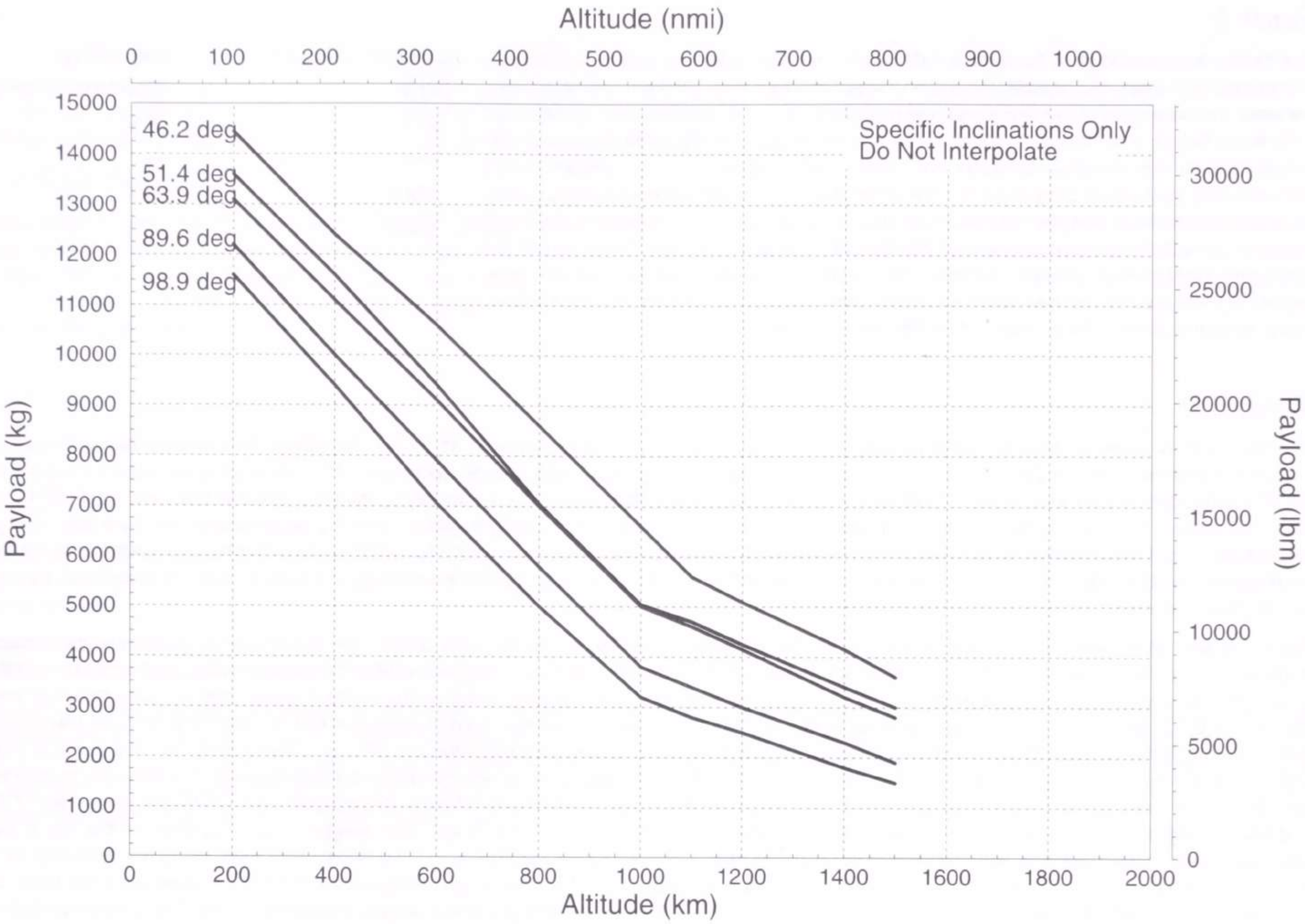
Launch Azimuth (deg)	Resulting Inclination (deg)
35.0	63.9
64.2	51.4
178.8	88.1
194.2	98.8

Zenit 3SL

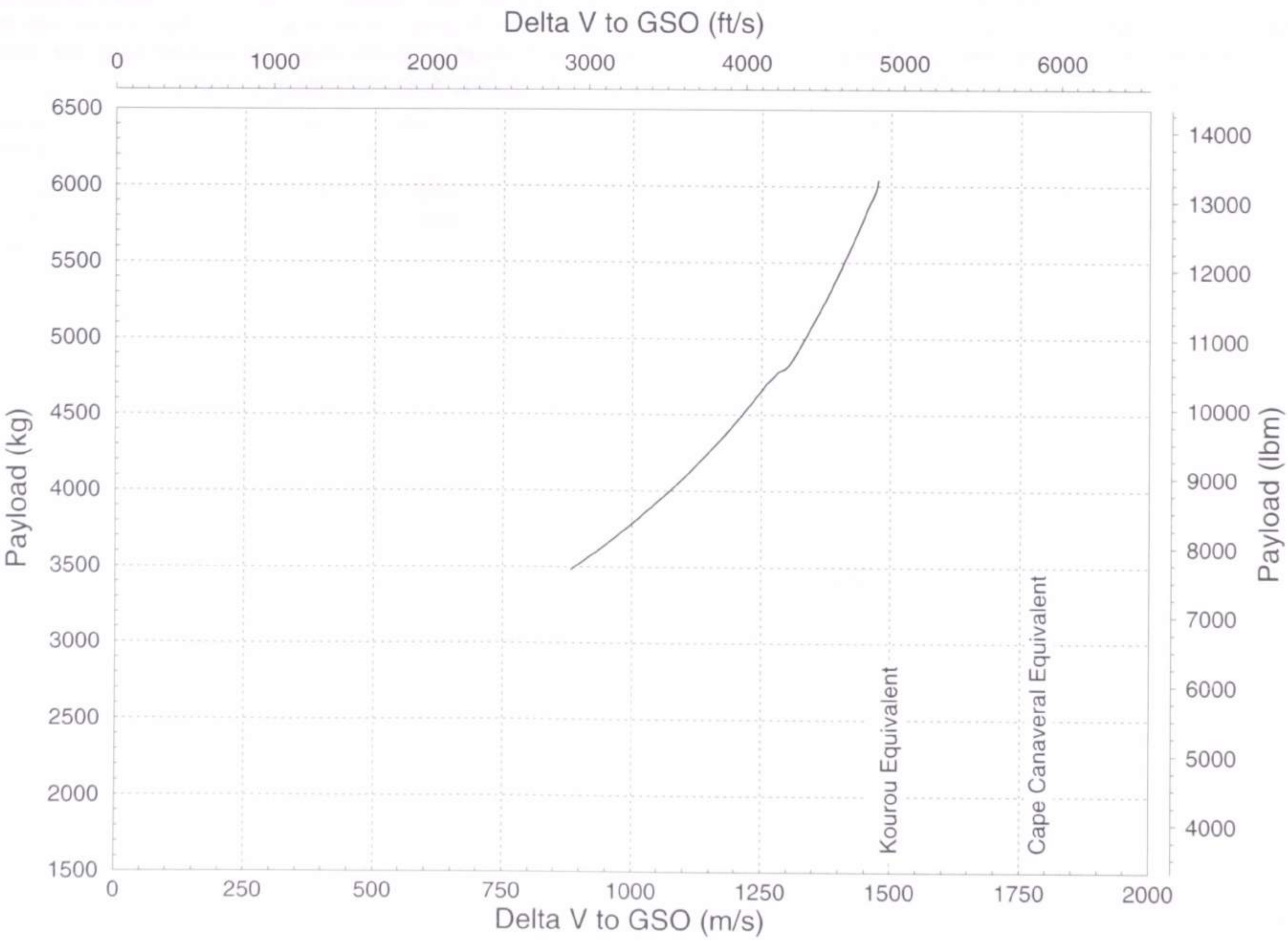
The Zenit 3SL is launched from a mobile floating launch platform at a point on the equator at 154° W longitude. This location has significant performance advantages. It allows Zenit 3SL to deliver spacecraft directly to a transfer orbit at 0-deg inclination, thus eliminating the need for a plane change to GEO and reducing the required spacecraft apogee burn. The lack of a plane change is especially useful for spacecraft that raise orbit with ion propulsion. The launch site location also nearly eliminates range safety concerns and resulting trajectory shaping requirements, as there are no populated areas to constrain the selection of downrange impact zones. The standard launch location is preferred because of the benign weather and waves even for missions to inclined orbits that do not need to be launched from the equator, even though theoretically the launch could be conducted from a different location if needed.

The two-stage Zenit retains its own avionics and flies autonomously of the Block DM-SL upper stage. The Block DM-SL takes over the mission upon separation from Zenit. The Block DM-SL upper stage is capable of multiple restarts and complex orbital maneuvers including a continuous roll during coast periods (unlike other versions of the Block DM). However, the convenient starting point on the equator means that no more than two starts are needed for GTO missions and short coast periods of roughly 30 min are typical. A single burn of the Block DM-SL is optimal for GTO payloads above approximately 5980 kg (13,180 lbm) and can deliver up to 6066 kg (13,375 lbm) to a synchronous GTO at 0 deg inclination. Two burns of the Block DM-SL are normally used for payloads less than 6000 kg (13,225 lbm), to raise perigee and maximize satellite lifetime. For two-burn missions, there are two types of injection schemes –"perigee inject" and "post-perigee inject." These have different intermediate park orbits and placement of the second burn. Perigee inject is optimal for the lower perigees associated with heavier payloads and post-perigee inject is optimal for the higher perigees associated with lighter payloads, with the crossover occurring near a perigee altitude of 2000 km (1250 miles). The Zenit 3SL is not optimized for heavy-lift LEO missions because payload mass is limited by the structural capability of the three-stage configuration. A Zenit 2S (eliminating the Block DM-SL) could deliver payloads in excess of 14,000 kg (30,850 lbm) to LEO, but modifications to current launch equipment on the Sea Launch platform would be required. Customers interested in this option should contact Boeing Launch Services for more information. Zenit 3SL performance has increased roughly 20% since the first flight. from 5000 kg (11,023 lbm) to over 6000 kg (13,225 lbm). The full 6000-kg equivalent performance was successfully flown in June 2003. The performance enhancement was implemented by various mass reductions concentrated in the third stage and payload accommodations, upgrading the second stage engine, increasing the propellant load and extending the nozzle on the third stage, and by reducing performance margins based on flight experience. The performance shown is based on two burns of the Block DM-SL, accounts for a standard 100-kg (220-lbm) payload adapter, and accounts for sufficient flight performance reserve to provide 99% probability of command shut-down.

Performance

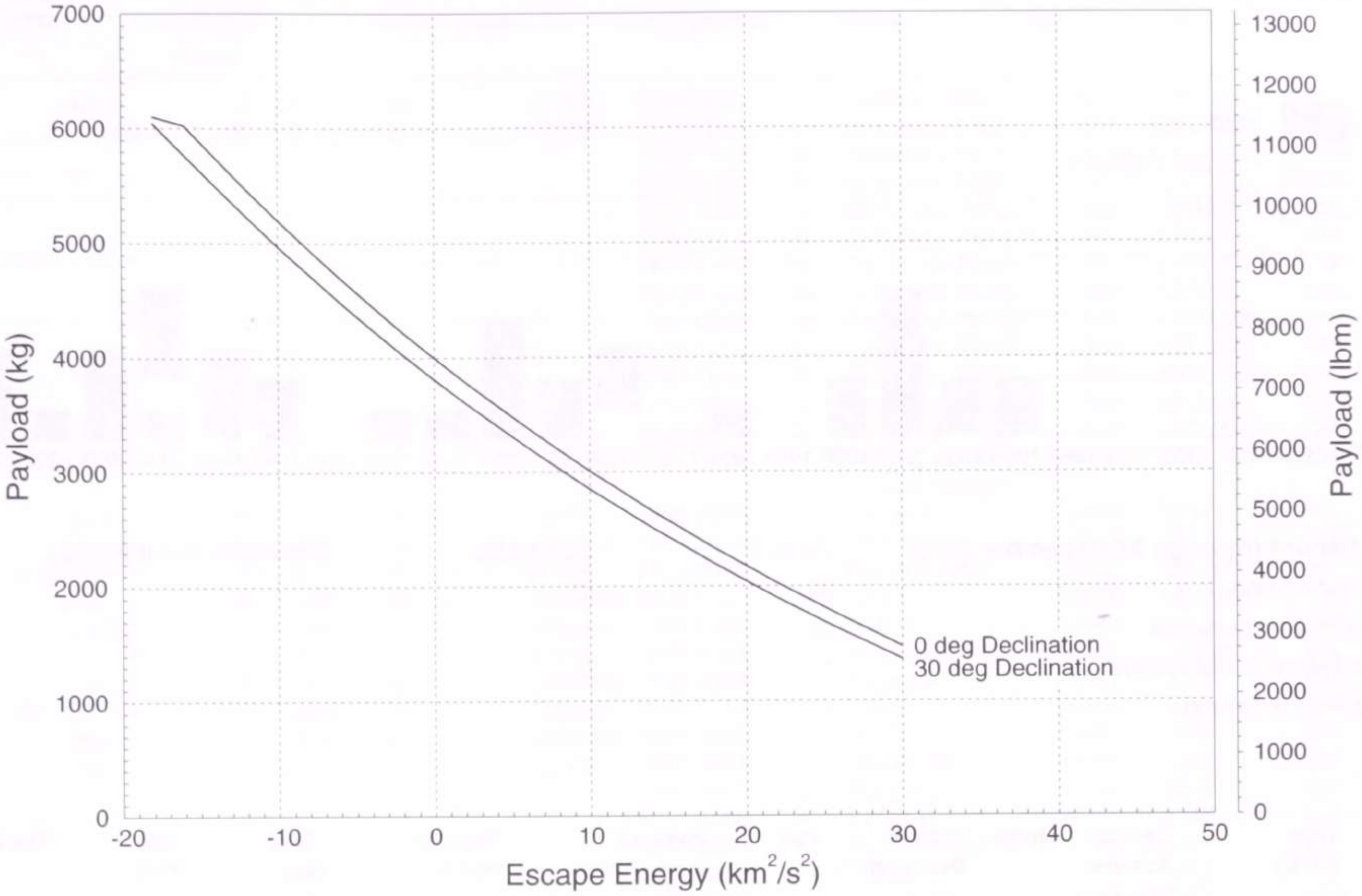


Zenit 2: Performance for Circular LEO Orbits Launched from Baikonur



Zenit 3SL: GTO Performance Capability

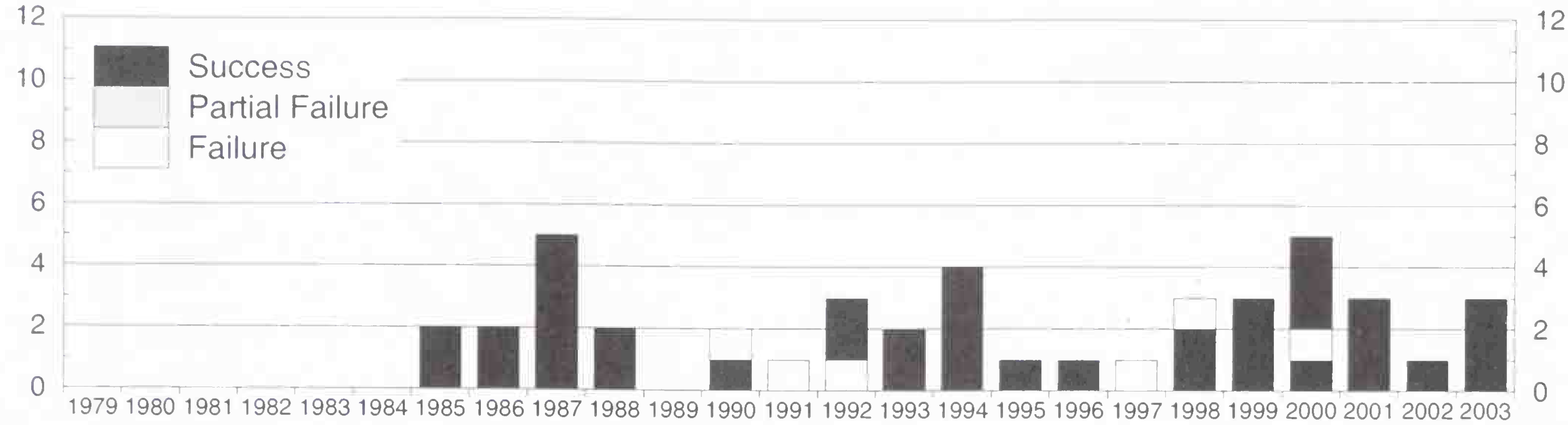
Performance



Zenit 3SL: Earth Escape Energy Capability

Flight History

Orbital Flights Per Year



Flight Record (through 31 December 2003)	Zenit 2	Zenit 3SL	Combined Zenit Family
Total Orbital Flights	33	11	44
Launch Vehicle Successes	28	10	38
Launch Vehicle Partial Failures	0	0	0
Launch Vehicle Failures	5	1	6

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
T	—	1985 Apr 13	—	2	1	LC 45	1985 F01A		ballast	3200	sub-orbital	MIL	USSR
T	—	Jun 21	69	2	2	LC 45	1985 053A		ballast	3200	sub-orbital	MIL	USSR
T	1	Oct 22	123	2	3	LC 45	1985 097A		Kosmos 1697	3200	LEO (71)	MIL	USSR
T	2	Dec 28	67	2	4	LC 45	1985 121A		Kosmos 1714	3200	LEO (71)	MIL	USSR
T	3	1986 Jul 30	214	2	5	LC 45	1986 056A		Kosmos 1767	10000	LEO (64.8)	MIL	USSR
T	4	Oct 22	84	2	6	LC 45	1986 080A		Kosmos 1786	3200	LEO (64.8)	MIL	USSR
T	5	1987 Feb 14	115	2	7	LC 45	1987 016A		Kosmos 1820	10000	LEO (64.9)	MIL	USSR
T	6	Mar 18	32	2	8	LC 45	1987 027A		Kosmos 1833	3200	LEO (71)	MIL	USSR
T	7	May 13	56	2	9	LC 45	1987 041A		Kosmos 1844	3200	LEO (71)	MIL	USSR
T	8	Aug 01	80	2	10	LC 45	1987 065A		Kosmos 1871	10000	LEO (97)	MIL	USSR
T	9	Aug 28	27	2	11	LC 45	1987 071A		Kosmos 1873	11000	LEO (64.8)	MIL	USSR
	10	1988 May 15	261	2	12	LC 45	1988 039A		Kosmos 1943	3200	LEO (71)	MIL	USSR
	11	Nov 23	192	2	13	LC 45	1988 102A		Kosmos 1980	3200	LEO (71)	MIL	USSR
	12	1990 May 22	545	2	14	LC 45	1990 046A		Kosmos 2082	3200	LEO (71)	MIL	USSR
F	**13**	**Oct 04**	**135**	**2**	**15**	**LC 45P**	1990 F05A		Kosmos	3250	LEO	MIL	USSR
F	**14**	**1991 Aug 30**	**330**	**2**	**16**	**LC 45L**	1991 F03A		Kosmos	3250	LEO	MIL	USSR
F	**15**	**1992 Feb 05**	**159**	**2**	**17**	**LC 45L**	1992 F01A		Kosmos		LEO	MIL	Russia
	16	Nov 17	286	2	18	LC 45L	1992 076A		Kosmos 2219	3200	LEO (71)	MIL	Russia
	17	Dec 25	38	2	19	LC 45L	1992 093A		Kosmos 2227	3200	LEO (71)	MIL	Russia
	18	1993 Mar 26	91	2	20	LC 45L	1993 016A		Kosmos 2237	3200	LEO (71)	MIL	Russia
	19	Sep 16	174	2	21	LC 45L	1993 059A		Kosmos 2263	3200	LEO (71)	MIL	Russia
	20	1994 Apr 23	219	2	22	LC 45L	1994 023A		Kosmos 2278	3200	LEO (71)	MIL	Russia
	21	Aug 26	125	2	23	LC 45L	1994 053A		Kosmos 2290	10600	LEO (64.8)	MIL	Russia
	22	Nov 04	70	2	24	LC 45L	1994 074A		Resurs O1	2900	SSO	CIV	Russia
	23	Nov 24	20	2	25	LC 45L	1994 077A		Kosmos 2297	3200	LEO (71)	MIL	Russia
	24	1995 Oct 31	341	2	26	LC 45L	1995 058A		Kosmos 2322	3250	LEO (71)	MIL	Russia
	25	1996 Sep 04	309	2	27	LC 45L	1996 051A		Kosmos 2333	3250	LEO (71)	MIL	Russia
F	**26**	**1997 May 20**	**258**	**2**	**28**	**LC 45L**	1997 F02A		Kosmos	3250	LEO	MIL	Russia
	27	1998 Jul 10	416	2	29	LC 45L	1998 043A		Resurs-01 4	2900	SSO	CIV	Russia
							1998 043A	A	IRIS (LLMS)		SSO	CML	Belgium
							1998 043B	A	Fasat-Bravo	50	SSO	MIL	Chile
							1998 043C	A	TM-SAT	49	SSO	CIV	Thailand
							1998 044D	A	Gurwin Techsat 1B	48	SSO	CML	Israel
							1998 044E	A	WESTPAC	24	SSO	CML	Australia
							1998 044F	A	SAFIR 2	55	SSO	CML	Germany

Launch Complex 45 is at Baikonur. *Odyssey* is the mobile launch platform for Sea Launch.
T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly
Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

Flight History

		Date (UTC)	Launch Interval (days)	Model	Vehicle Designation	Pad	Payload Designation		Payload Name	Mass (kg)	Orbit (Incl)	Market	Country
	28	Jul 28	18	2	30	LC 45L	1998 045A		Kosmos 2360	3250	LEO (71)	MIL	Russia
F	**29**	**Sep 09**	**43**	**2**	**31**	**LC 45L**	1998 F05A	M	Globalstar FM5	450	LEO (52)	CML	USA
							1998 F05B	M	Globalstar FM7	450	LEO (52)	CML	USA
							1998 F05C	M	Globalstar FM9	450	LEO (52)	CML	USA
							1998 F05D	M	Globalstar FM10	450	LEO (52)	CML	USA
							1998 F05E	M	Globalstar FM11	450	LEO (52)	CML	USA
							1998 F05F	M	Globalstar FM12	450	LEO (52)	CML	USA
							1998 F05G	M	Globalstar FM13	450	LEO (52)	CML	USA
							1998 F05H	M	Globalstar FM16	450	LEO (52)	CML	USA
							1998 F05I	M	Globalstar FM17	450	LEO (52)	CML	USA
							1998 F05J	M	Globalstar FM18	450	LEO (52)	CML	USA
							1998 F05K	M	Globalstar FM20	450	LEO (52)	CML	USA
							1998 F05L	M	Globalstar FM21	450	LEO (52)	CML	USA
T	30	1999 Mar 28	200	3SL		*Odyssey*	1999 014A		HS 702 Mass Demonstrator	4500	GTO	CML	USA
	31	Jul 17	111	2	33	LC45L	1999 039A		Okean O	6150	SSO	MIL	Russia
	32	Oct 10	84	3SL	34	*Odyssey*	1999 056A		DirecTV-1R	3548	GTO	CML	USA
	33	2000 Feb 03	117	2	35	LC45L	2000 006A		Kosmos 2369	3100	LEO (71)	MIL	Russia
F	**34**	**Mar 12**	**38**	**3SL**	**36**	***Odyssey***	2000 F02A		ICO-1	2079	MEO (45)	CML	USA
	35	Jul 28	138	3SL	37	*Odyssey*	2000 043A		PAS-9	3748	GTO	CML	USA
	36	Sep 25	59	2	38	LC45L	2000 056A		Kosmos 2372	10400	LEO (64.8)	MIL	Russia
	37	Oct 21	25	3SL	39	*Odyssey*	2000 066A		Thuraya-1	5107	GTO	CML	UAE
	38	2001 Mar 18	149	3SL	40	*Odyssey*	2001 012A		XM-2	4695	GTO	CML	USA
	39	May 08	51	3SL	41	*Odyssey*	2001 018A		XM-1	4682	GTO	CML	USA
	40	Dec 10	216	2	42	LC45L	2001 056A		Meteor 3M	3100	SSO	CIV	Russia/USA
							2001 056B	A	Kompass	80	SSO	CIV	Russia
							2001 056C	A	BADR-B	70	SSO	CIV	Pakistan
							2001 056D	A	Maroc Tubsat	45	SSO	CIV	Morocco
							2001 056E	A	Reflector	8	SSO	CIV	Russia/USA
	41	2002 Jun 15	187	3SL	43	*Odyssey*	2002 030A		Galaxy IIIC	4850	GTO	CML	USA
	42	2003 Jun 10	360	3SL	44	Odyssey	2003 026A		Thuraya 2	5177	GTO	CML	UAE
	43	Aug 08	59	3SL	45	Odyssey	2003 034A		Echostar 9	4737	GTO	CML	USA
	44	Oct 01	54	3SL	46	Odyssey	2003 044A		Galaxy 13	4090	GTO	CML	USA/Japan

Launch Complex 45 is at Baikonur. *Odyssey* is the mobile launch platform for Sea Launch.

T = Test Flight; **F** = Launch Vehicle Failure; **P** = Launch Vehicle Partial Failure; **S** = Spacecraft or Upper-Stage Anomaly

Payload Types: C = Comanifest; M = Multiple Manifest; A = Auxiliary Payload

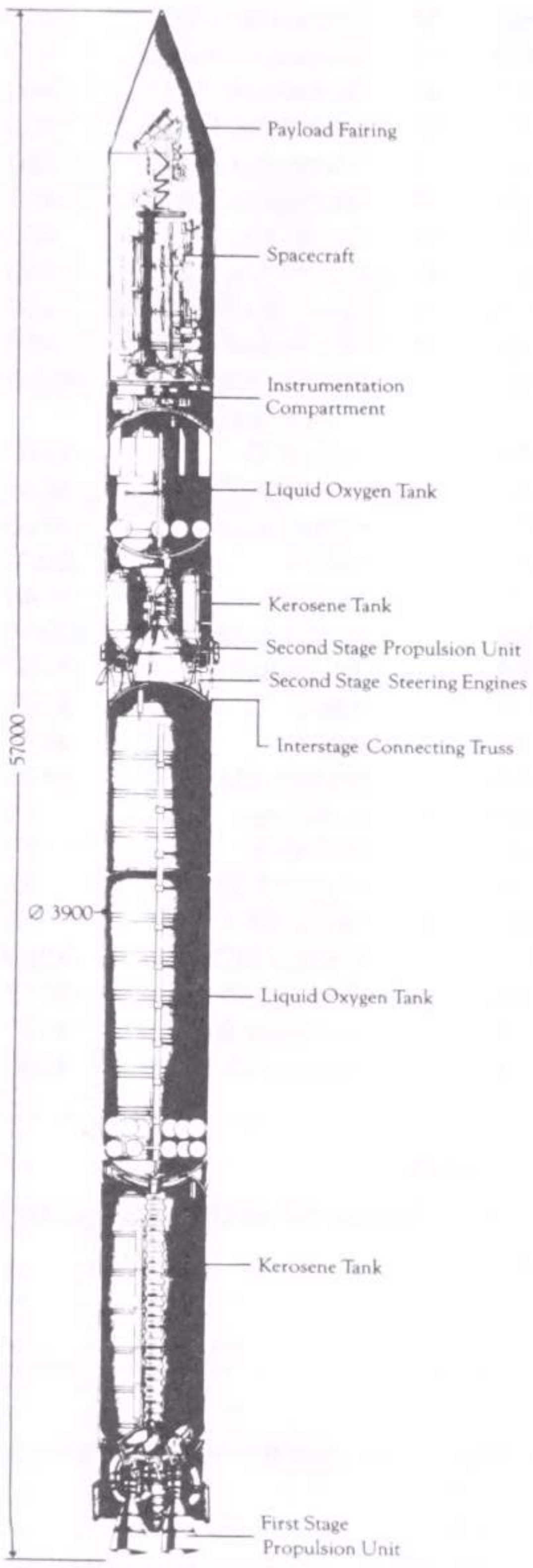
Failure Descriptions:

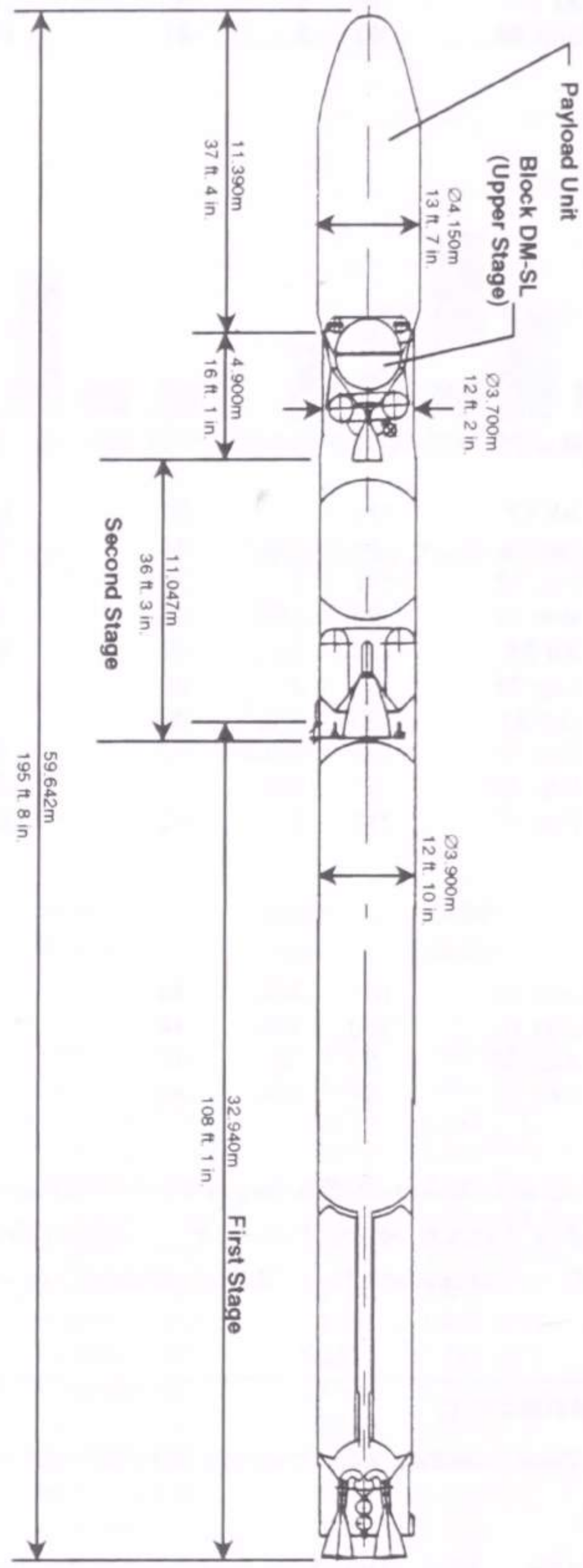
Several early Zenit launches were misinterpreted by outside observers as failures. Please see the Vehicle History section for more information on these flights.

F	1990 Oct 04	15	1990 F05	At T+3 s first-stage RD-171 engine caught fire, destroying the launch vehicle and heavily damaging the launch pad, which has not been repaired since; failure attributed to contamination by traces of lubricating oil in the oxygen manifold after engine test firing.
F	1991 Aug 30	16	1991 F03	Second-stage RD-120 engine oxidizer turbopump caught fire, causing failure to reach orbit.
F	1992 Feb 05	17	1992 F01	Same as previous flight; second-stage RD-120 engine oxidizer turbopump caught fire, causing failure to reach orbit. Failures traced to material change in oxidizer turbopump component, which caused sparks during engine start. Problem had not been detected following previous flight because normal engine tests are performed horizontally, but failure only occured in the vertical position as in flight.
F	1997 May 20	28	1997 F02	Structural failure of strut in first-stage RD-171 engine occured at T+48 s causing failure of engine and resulting loss of vehicle. Failure traced to undetected damage suffered due to high vibration during engine acceptance test.
F	1998 Sep 09	31	1998 F05	Failure at T+272 s, after two out of three redundant guidance computer channels became unsynchronized and were voted out. As designed, remaining channel ordered thrust termination for flight safety reasons, causing vehicle to impact in Siberia.
F	2000 Mar 12	34	2000 F02	A ground software error resulted from mission-specific changes for this launch that required a 1 s launch window. Software failed to command a valve in the second-stage pneumatic system to close before liftoff. The pneumatic system is used for several functions, including operation and actuation of the RD-8 steering engine. The system lost more than 60% of its pressure in flight, reducing the control capability of the engine, and ultimately causing loss of attitude control. The automatic flight temination system shutdown the vehicle 8 min after liftoff.

Vehicle Design

Overall Vehicle





Courtesy Sea Launch.

	Zenit 2	Zenit 3SL
Height	57.0 m (187 ft)	59.6 m (195.5 ft)
Gross Liftoff Mass	459 t (1012 klbm)	471 t (1038 klbm)
Thrust at Liftoff	7269 kN (1630 klbf)	7269 kN (1630 klbf)

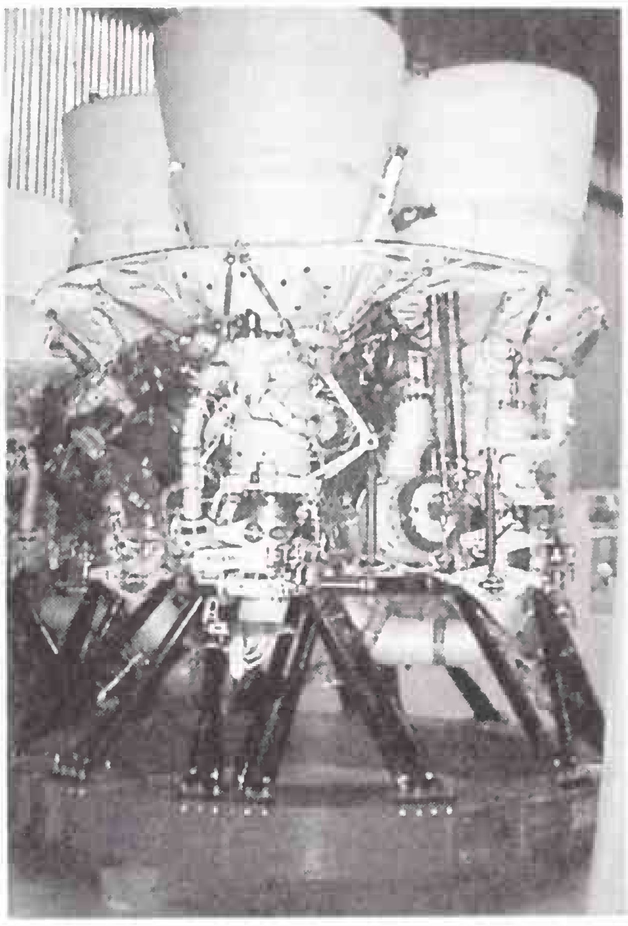
Vehicle Design

Zenit Stages

The Zenit was developed from the 1970s to the mid 1980s and is the only large expendable rocket developed by the former Soviet Union since the ill-fated N-1 lunar rocket of the late 1960s. Therefore, its general design and many of its systems are more modern than other Soviet and Russian launch vehicles. Zenit uses two or three in-line stages, each fueled by LOX/kerosene.

The first stage was developed in parallel with the Energia strap-on boosters, and it shares the same basic structure and propulsion systems. The first-stage RD-171 engine is the most powerful liquid engine ever flown, and its high-pressure staged-combustion turbopumps provide higher sea-level specific impulse than any other LOX/kerosene engine. The primary difference between Zenit's RD-171 and the RD-170 used on Energia is that the Energia RD-170 thrust chambers gimbaled in two axes, while the Zenit RD-171 thrust chambers can gimbal in only one axis. The RD-170 also accommodated a parachute recovery system, and the engine was qualified for use in up to 10 flights as part of the Energia program. The upper LOX tank fits in a concave depression at the top of the kerosene tank, and the LOX feed line runs through the middle of the lower tank. Separation is achieved with four solid retro-rockets located at the base of the stage. The Zenit 3SL first stage is very similar to the Zenit 2 first stage, with strengthened structures for sea-based operations, and additional LOX loading lines for the Block DM-SL upper stage.

The second stage is connected to the first by an open truss structure. Propulsion is provided by a single RD-120 engine, with steering provided by an RD-8 vernier engine with four thrust chambers spaced around the outside of the RD-120. A toroidal kerosene tank surrounds the main engine, and oxygen is stored in a standard cylindrical tank. The second stage is topped by an instrumentation compartment containing the avionics. There are several differences between the Zenit 3SL second stage and the Zenit 2 second stage. The most significant is the incorporation of a different suite of guidance, navigation, and flight control avionics drawing in part on systems used on Soviet submarine ballistic missiles and thus suitable for launch from a floating platform. Structural modifications were also made to adapt for carrying the Block DM-SL upper stage instead of a fairing, to provide local stiffening associated with ground support interfaces, and to add LOX loading lines for the Block DM-SL. In 2002 Energomash completed an upgrade of the RD-120 engine that increased thrust by 10%.

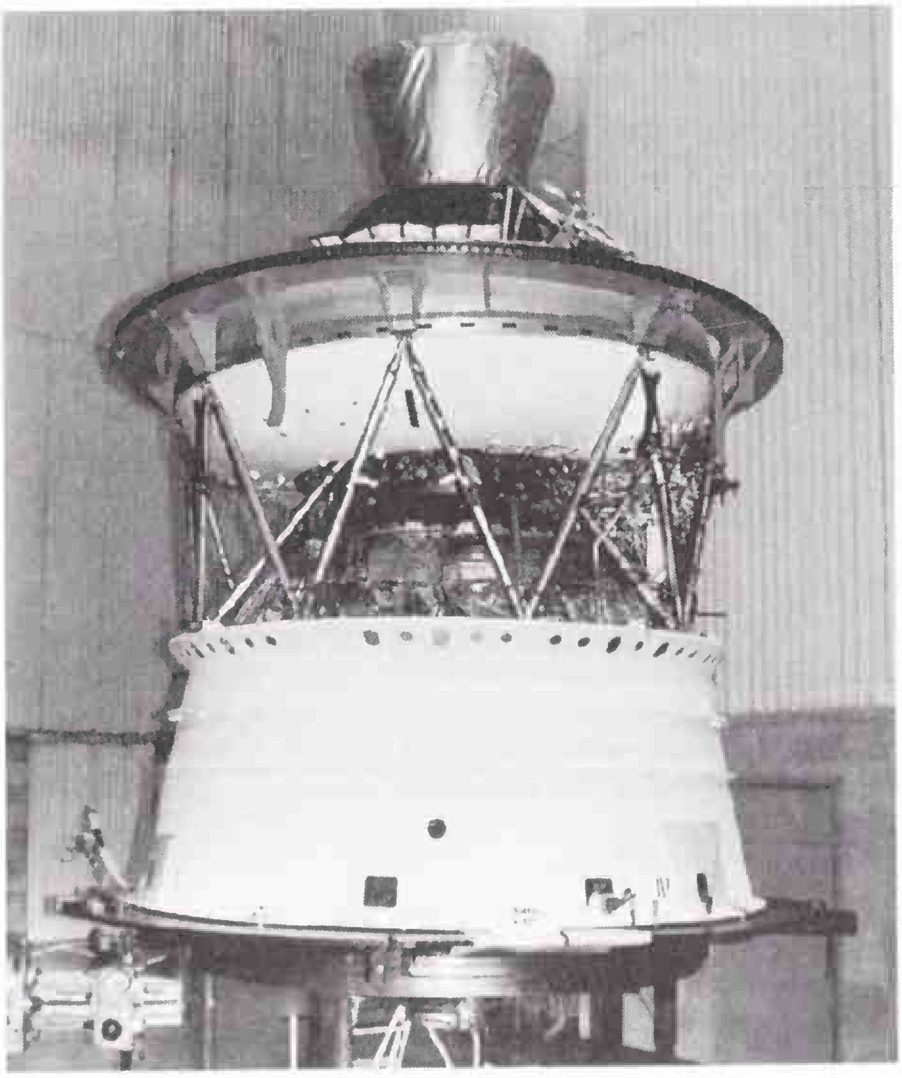


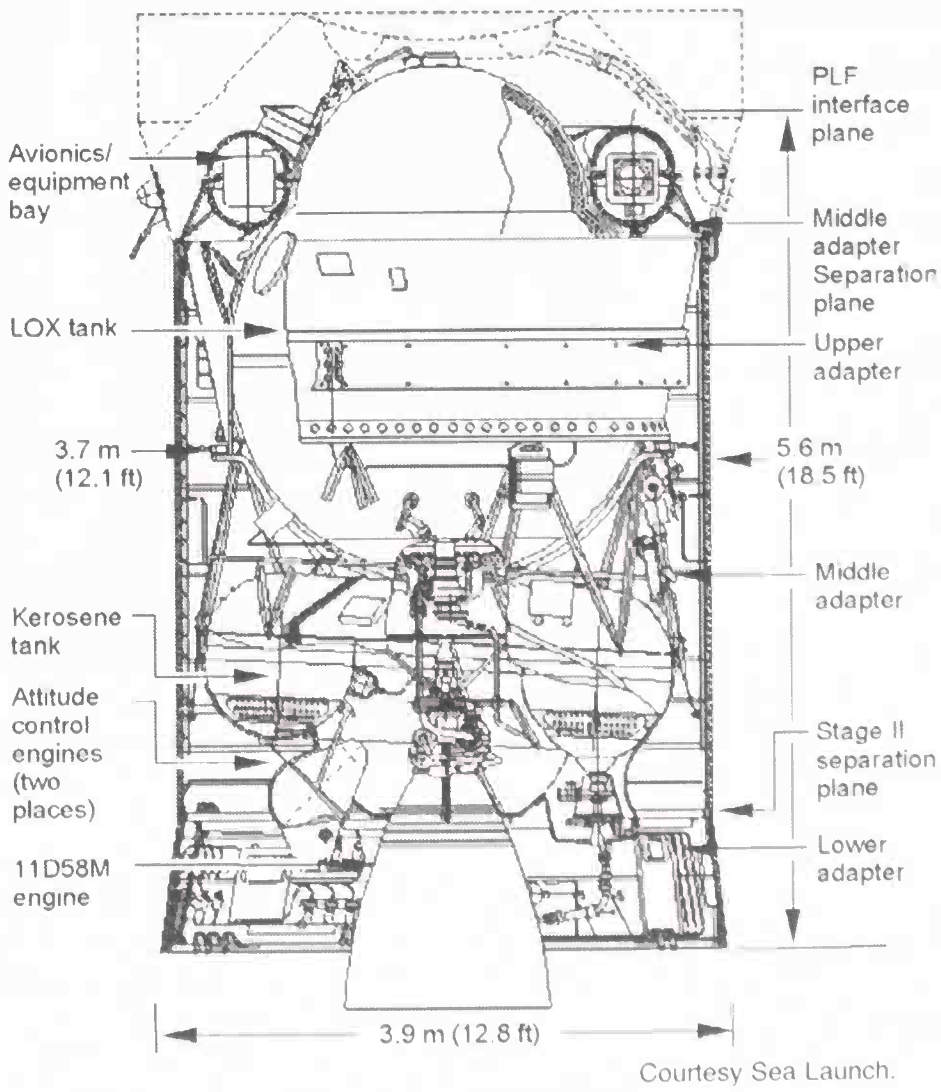
Courtesy Sea Launch.

The RD-171 engine used on the first stage is the largest liquid engine ever flown.

Block DM-SL Upper Stage

The third stage for the Zenit 3SL is a derivative of the Block DM stage used on Proton. The Block DM-SL includes a spherical LOX tank, a toroidal kerosene tank around the single 11D58M main engine, and a pressurized instrumentation compartment. These elements are connected with a truss structure. When integrated on the launch vehicle, the Block DM-SL is contained inside a cylindrical skin-stringer interstage that is jettisoned before Block DM-SL ignition. The interstage is divided into two sections, referred to as the lower adapter and middle adapter. The mass of these two components is 526 kg and 715 kg (1160 lbm and 1575 lbm), respectively. The stage can function for 24 h or more on orbit, and is capable of four restarts, allowing complex orbital maneuvers. On Sea Launch no more than two starts are typically necessary and mission duration is unlikely to exceed 2 h. The Block DM-SL is capped by an aluminum interface skirt. This flared structure serves as the transition from the 3.7-m (12.2-ft) diam stage to the 4.15-m (13.6-ft) diam fairing, and also carries the conical payload support structure. Other modifications from the Proton version include updated avionics (including a different flight guidance unit, which among other things allows continuous roll maneuvers), modified interfaces to the lower stages and the payload section, and modified prelaunch support interfaces. For example, on Proton the fueling and umbilical connections are provided by an umbilical tower. On Zenit, these connections are routed up through the lower stages. The instrument compartment of the Block DM-SL is also mounted lower than in previous versions of the Block DM in order to provide clearance for the payload adapter. Enhancements to the Block DM-SL since its introduction include a longer nozzle on the main engine and a 9% increase in propellant.





Courtesy Sea Launch.

The Block DM-SL upper stage (shown upside down on the left) is a modified version of the Proton upper stage.

Vehicle Design

	Stage 1 (Block A)	Stage 2 (Block B)	Block DM-SL
Dimensions			
Length	32.9 m (108 ft)	10.4 m (34 ft)	4.9 m (16.1 ft) stacked 5.6 m (18.5 ft) alone
Diameter	3.9 m (12.8 ft)	3.9 m (12.8 ft)	3.7 m (12.1 ft)
Mass			
Propellant Mass	322.3 t (710.5 klbm)	81.7 t (180.2 klbm)	15.9 t (34.9 klbm)
Inert Mass	32.3 t (71.2 klbm)	9 t (19.8 klbm)	2620 kg (5775 lbm) excluding jettisoned adapters
Gross Mass	354.6 t (781.7 klbm)	90.8 t (200.1 klbm)	18.5 t (40.8 klbm)
Propellant Mass Fraction	0.91	0.90	0.86
Structure			
Type	Integrally machined stiffeners	Integrally machined stiffeners	Monocoque and truss
Material	Aluminum	Aluminum	Aluminum
Propulsion			
Engine Designation	RD-171 (11D520)	RD-120 (11D123) + RD 8 vernier	11D58M
Number of Engines	1 (4 thrust chambers)	1 main engine and 1 vernier with 4 chambers	1 main engine and 2 attitude control/ullage engines
Propellant	LOX/kerosene	LOX/kerosene (main and vernier engines)	LOX/kerosene (main engine) N_2O_4/UDMH (ullage engines)
Average Thrust	Sea level: 7269 kN (1630 klbf) Vacuum: 7915 kN (1780 klbf)	Main engine, vacuum: 992 kN (223 klbf) Vernier, vacuum: 78.4 kN (17.6 klbf) total	Vacuum: 84 kN (18.7 klbf)
Isp	Sea level: 309 s Vacuum: 337 s	Main engine, vacuum: 350 s Vernier, vacuum: 342 s	Vacuum: 356 s
Chamber Pressure	245 bar (3556 psi)	Main engine: 163 bar (2364 psi) Vernier: 77 bar (1117 psi)	77.4 bar (1123 psi)
Nozzle Expansion Ratio	37:1	Main engine: 106:1 Vernier: 104:1	280:1
Propellant Feed System	Staged-combustion turbopump	Main engine: Staged-combustion turbopump Vernier: turbopump	Staged-combustion turbopump
Mixture Ratio (O/F)	2.63:1	2.58:1	2.48:1
Throttling Capability	49–100%	Main engine: 83–100% Vernier: 100% only	100% only
Restart Capability	No	No	Up to 4 restarts
Tank Pressurization	Pressurized nitrogen	Pressurized nitrogen	Stored-pressurized gas
Attitude Control			
Pitch, Yaw, Roll	Nozzle gimbal ±6.3 deg	Vernier engine nozzle gimbal ±33	Main engine gimbaling and ACS
Staging			
Nominal Burn Time	140–150 s	Zenit 2 main engine: 200–315 s Zenit 3SL main engine: 300 s Vernier: 300–1100 s	712 s (single-burn mission) 270 s + 425 s (two-burn mission)
Shutdown Process	Command shutdown or burn to depletion	Command shutdown or burn to depletion	Predetermined velocity or burn to depletion
Stage Separation	4 retro-rockets	4 retro-rockets	ACS/Spring ejection

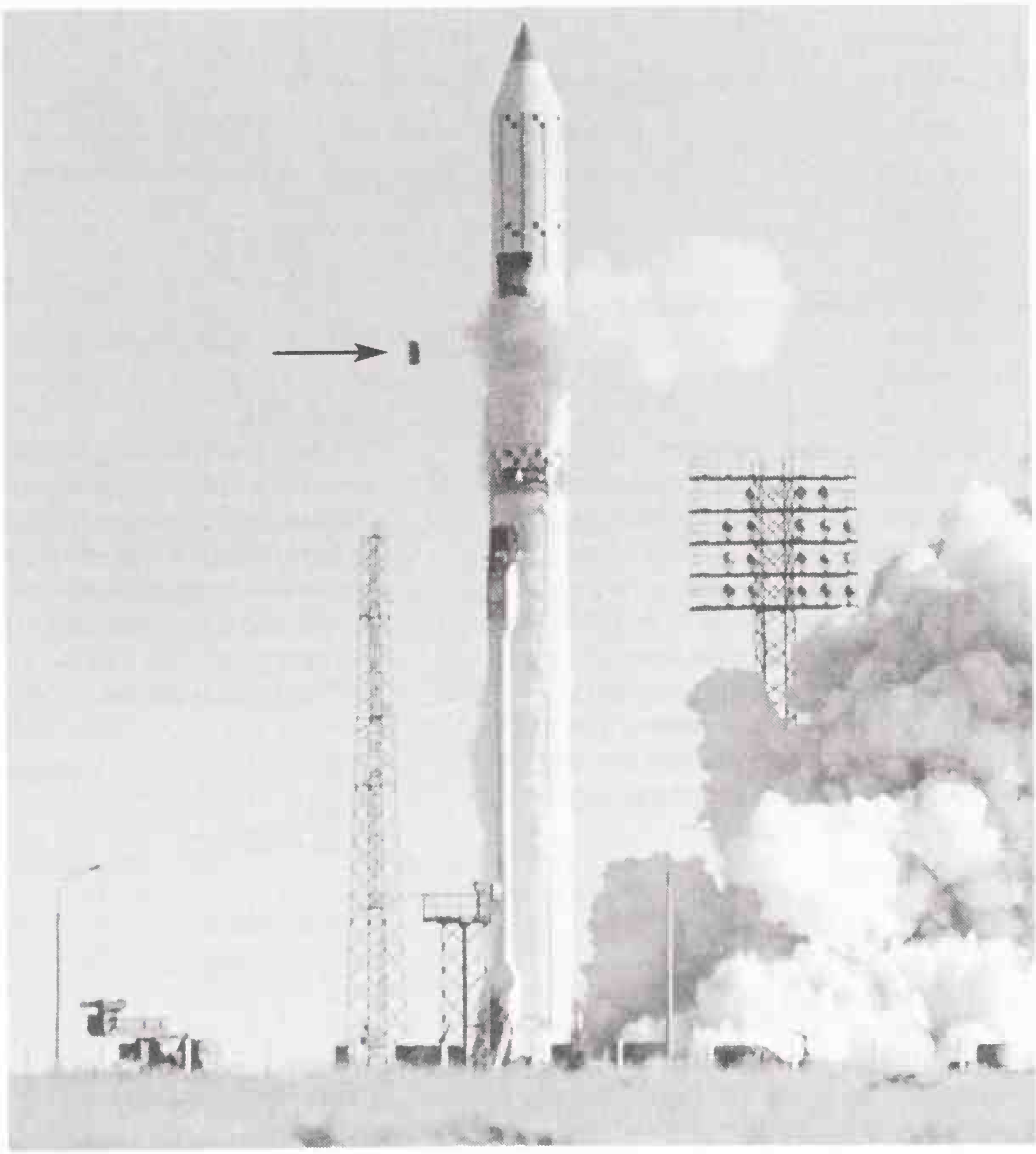
Vehicle Design

Attitude Control System

The four thrust chambers of the RD-171 engine can be individually gimbaled ±5 deg in one axis to provide three-axis control. One of the few differences between Zenit's RD-171 and the RD-170 used in the Energia strap-on booster is that the RD-170 engine supported two-axis gimbaling of each thrust chamber. Attitude control during second-stage flight is provided by the RD-8 vernier engine, which uses one turbopump to feed four thrust chambers, each of which can be gimbaled. The main engine is fixed and does not contribute to attitude control. The RD-8 draws LOX/kerosene propellant from the same tanks as the main engine and operates from 10 s before second-stage ignition to 75 s after main-engine shutdown. During third-stage burns, the main-engine gimbal provides pitch and yaw control, and turbopump bleed gas is used for roll control. The Block DM-SL upper stage also has two sets of independent ACS thruster units for three-axis attitude control during coast phases and payload deployment, and for propellant settling before main-engine ignition. These are fueled by small tanks of N_2O_4/UDMH.

Avionics

The Zenit 2 vehicle uses a digital flight control system. The navigation system is aligned using an optical update unit that is jettisoned on the pad immediately before launch. The Biser-2 flight computer has three redundant channels, with majority voting to detect a failure in any channel. During Flight #31, two channels malfunctioned and were interpreted as failures by the remaining channel. As programmed, the remaining channel commanded flight termination rather than continue on a single channel, since there was no way to confirm that the remaining channel was not also faulty. The Zenit 3SL uses the upgraded Biser-3 computer, which has been modified to prevent this problem. As is typical with Russian and Ukrainian launch vehicles, flight termination is performed by automatic shutdown of the engines, which brings the vehicle down largely intact, rather than by triggering explosive systems to destroy the vehicle as is typically done in the West. However, the Zenit 2 has a much more extensive flight safeguard system than most vehicles, probably stemming from early man-rating requirements for proposed spaceplane programs. The flight safeguard system can transmit a launch vehicle emergency signal to the spacecraft resulting from either an onboard computer failure or a loss of stability. Similarly, it can receive a spacecraft emergency signal from the spacecraft. If an emergency occurs after a predetermined time late enough in flight that the spacecraft can still complete its mission or perform an abort, an emergency main-engine shutdown is commanded, the payload fairing is jettisoned, and the payload is separated. If a launch vehicle emergency occurs in the first 20 s of flight, instead of commanding engine shutdown, the safety system commands a 6-deg/s pitchover maneuver to move the vehicle downrange as rapidly as possible so that the launch complex will not be destroyed.



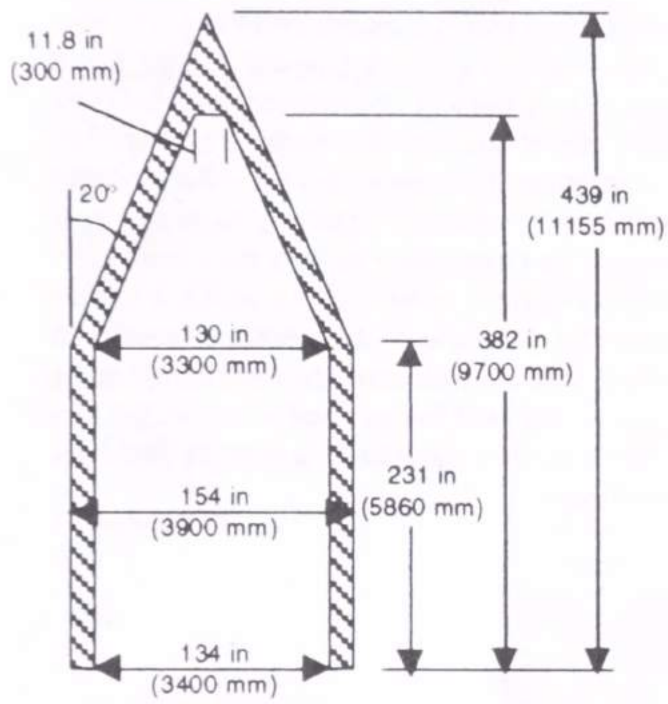
Courtesy KB Yuzhnoye.

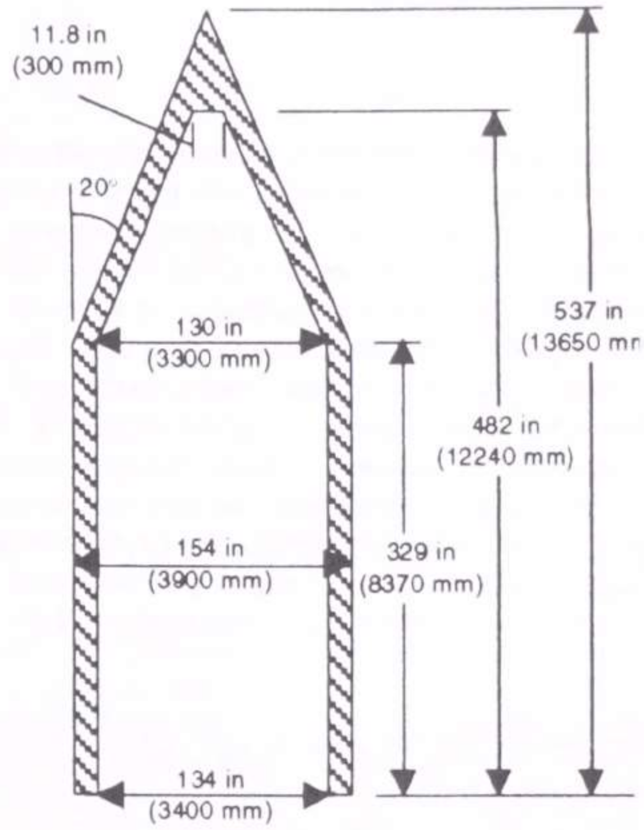
The optical alignment unit (see arrow) is shown being jettisoned from the second stage immediately before liftoff.

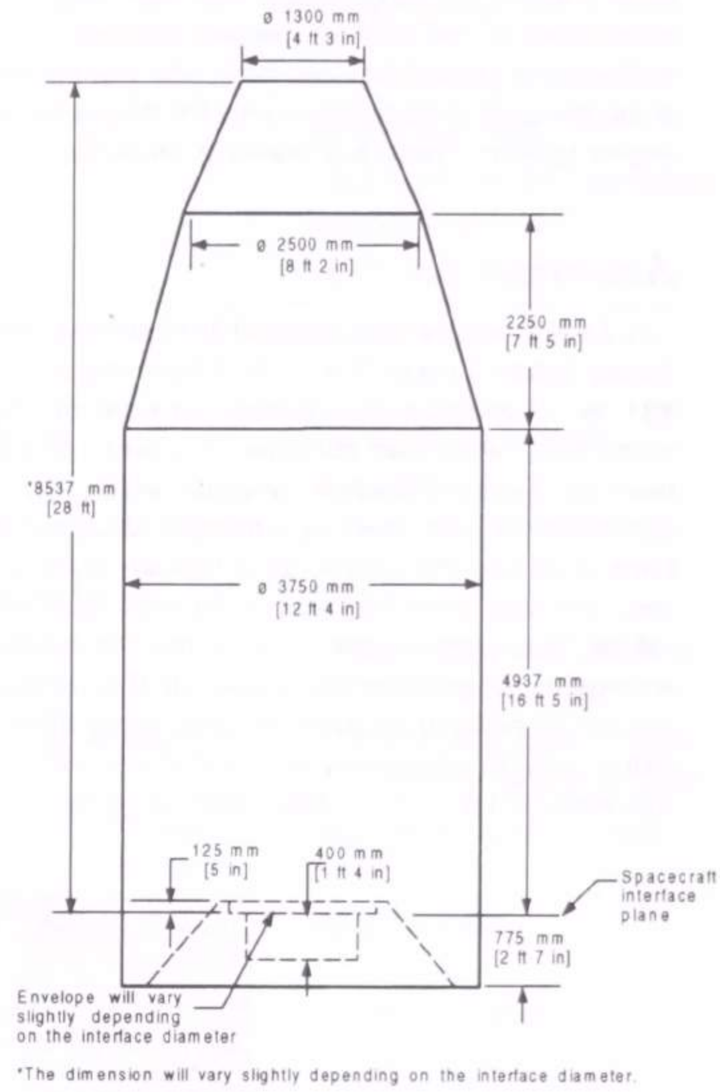
When used in the three-stage configuration, the two lower stages of Zenit are still controlled by their own avionics, with separate systems on the Block DM-SL upper stage. The avionics for the lower stages are similar to Zenit 2, but with a new, more-redundant guidance computer that has been flight proven on other Soviet launch systems. Block DM-SL avionics are installed in a toroidal pressurized compartment at the top of the stage. The Block DM-SL has an updated flight computer, and a new inertial navigation unit that is better able to operate from a floating launch platform. The Zenit 3SL telemetry system can transmit to either the tracking antennas on the assembly and command ship, or to TDRSS communications relay satellites.

Vehicle Design

Payload Fairing







Courtesy Sea Launch.

Note: Dimensions for Zenit 3SL are for the payload envelope, not the fairing.

	Zenit 2	Zenit 3SL
Remarks	The Zenit 2 fairing is a riveted aluminum skin-stringer structure, consisting of a cylindrical section and a 20-deg conical nose. The fairing is available in two lengths—a standard long fairing and a shorter version made by removing the lower third of the cylindrical section. A third 17-m (56-ft)-long configuration was designed but never flown. Access doors and radio transparent windows are standard. The payload fairing is separated by releasing mechanical latches and firing 12 explosive bolts, then pushing the two sections apart with a helium pneumatic system.	The Sea Launch fairing is built by Boeing with a large diameter to carry large communications satellites. The structure is composite material over aluminum honeycomb with external insulation. The fairing is designed for offline encapsulation at Sea Launch home port before integration with the launch vehicle. Up to two payload access doors 610 mm (24 in.) in diameter are standard, one per fairing half. The location of the doors can be varied based on Sea Launch defined constraints.
Length	Short: 11.16 m (36.6 ft) Long: 13.65 m (44.8 ft)	11.39 m (37.4 ft)
Primary Diameter	3.9 m (12.8 ft)	4.15 m (13.6 ft)
Mass	Short: 2100 kg (4630 lbm) Long: 2300 kg (5070 lbm)	1700 kg (3750 lbm)
Sections	2	2
Structure	Skin-stringer	Honeycomb sandwich
Material	Aluminum	Graphite–epoxy face sheets over aluminum honeycomb

Payload Accommodations

	Zenit 2	Zenit 3SL
Payload Compartment		
Maximum Payload Diameter	3300–3400 mm (130–134 in.)	3750 mm (147.6 in.)
Maximum Cylinder Length	Short: 5860 mm (230.7 in.) Long: 8370 mm (329.5 in.)	4937 mm (194.4 in.)
Maximum Cone Length	3870 mm (152.4 in.)	3600 mm (141.7 in.)
Payload Adapter Interface Diameter	3620 mm (142.5 in.)	1666 mm (65.6 in.) 1664 mm (65.5 in.) 1194 mm (47.0 in.) 937 mm (36.9 in.) (nonstandard)
Payload Integration		
Nominal Mission Schedule Begins	T–18 months	T–18 months for new mission, T–12 months for reflight of similar payload
Launch Window		
Last Countdown Hold Not Requiring Recycling	T–15 min (Next available launch opportunity is 2 h later)	T–30 minutes
On-Pad Storage Capability	4 h for a fueled vehicle 9 days for unfueled vehicle	4 h for fueled vehicle Unfueled vehicles would be stored horizontally in a nearby hangar
Last Access to Payload	T–30 h or T–3 h through access doors	T–19 h through access doors
Environment		
Maximum Axial Load	4.5 *g*	4.5 *g*
Maximum Lateral Load	2 *g* lateral	2 *g*
Minimum Lateral/Longitudinal Payload Frequency	10 Hz/20 Hz	8 Hz/20 Hz
Maximum Acoustic Level	131 dB at 200 Hz	134 db at 250 Hz
Overall Sound Pressure Level	140 dB from 20–20,000 Hz	142 dB
Maximum Flight Shock	500 *g* from 100–5000 Hz	4000 *g* at 10 kHz (1666-mm interface) 6000 *g* at 2.5 kHz (1664-mm interface) 2000 *g* at 2 kHz (1194-mm interface)
Maximum Dynamic Pressure on Fairing	?	53 kPa (1106 lb/ft^2)
Maximum Aeroheating Rate at Fairing Separation	1135 W/m^2 (0.1 $BTU/ft^2/s$)	1135 W/m^2 (0.1 $BTU/ft^2/s$)
Maximum Pressure Change in Fairing	5 kPa/s (0.7 psi/s)	2.2 kPa/s (0.3 psi/s)
Cleanliness Level in Fairing	Class 10,000	Class 5000
Payload Delivery		
Standard Orbit Injection Accuracy (2.3 sigma)	200 km (108 nmi) circular orbit: Altitude: ±7 km (3.8 nmi) Inclination: ±0.07 deg Period: ±2.5 s	Standard GTO: Perigee: ±10 km (5.4 nmi) Apogee: ±80 km (43 nmi) Inclination: ±0.25 deg
Attitude Accuracy (2.3 sigma)	±2 deg, ±2.5 deg/s	±3.0 deg, ±0.2 deg/s
Nominal Payload Separation Rate	2.8 m/s (9.2 ft/s)	0.3 m/s (1 ft/s)
Deployment Rotation Rate Available	0.7 rpm	Contact Sea Launch
Loiter Duration in Orbit	0	3 h
Maneuvers (Thermal/Collision Avoidance)	Thermal: No Collision avoidance: Yes	Yes
Multiple/Auxiliary Payloads		
Multiple or Comanifest	Zenit can carry multiple payloads on mission unique dispensing structures. For example, 12 Globalstar spacecraft were carried on one flight. Comanifested primary payloads are not standard.	Not currently available, but a possible future option.
Auxiliary Payloads	Zenit has carried up to five auxiliary payloads on one flight. However, no standard auxiliary payload interface or accommodations have been documented.	Not currently available, but a possible future option.

Production and Launch Operations

Production

The Zenit was designed by SDO Yuzhnoye of Dneipropetrovsk, Ukraine. Yuzhnoye has been designing and building strategic missiles and related space systems since 1954. It has produced about 1400 ICBMs, roughly half of the former Soviet land-based missiles. In 1991, SDO Yuzhnoye split into two distinct organizations. SDO Yuzhnoye is the design bureau, which employs about 3500 technical staff. PO Yuzhmash (Yuzhnoye Mashinostroeniya—Southern Machine Building Plant) is the production organization, responsible for manufacturing products designed at the design bureau, such as the Zenit and Tsiklon launch vehicles. About 28,000 people are employed at the PO Yuzhmash factory. This is down from a combined total of 40,000–50,000 employees in 1991.

Courtesy Sea Launch.

Zenit first stages in storage at PO Yuzhmash.

NPO Energomash of Khimki, Russia designed and produced both test and initial flight units for both the RD 170/171 first-stage and the RD-120 second-stage engines. Soviet practice was to transition serial engine production from the design bureau to dedicated factories once the designs were proven. This was done for the RD-120, which has been made since the 1980s at PO Yuzhmash in Dneipropetrovsk, Ukraine. In the case of the RD 170/171, however, the transition (to PO Polyot in Omsk, Russia) was started but never completed because of the cancellation of the Energia program and the lower than planned Zenit flight rates. As a result, Energomash has retained the RD-171 production capability and manufactures them today in the same location with the very similar RD-180 engine (essentially an RD-171 with only two rather than four thrust chambers) which has been adopted for the Atlas III and V. All told approximately 300 RD-170/171 engines have been built, of which 243 were used for test firing that cumulatively exceeded 100,000 s. As of 2003, 46 RD-171 and eight RD-170 engines have flown on Zenit and Energia vehicles respectively. Forty-six RD-120 engines have flown on Zenit, but three were not operated. All first- and second-stage engines undergo acceptance test firing.

Zenit-2 avionics are provided by NPO Auto Pribor and NPTs AP. The advanced automated launch pads at Baikonur and on the Sea Launch Odyssey platform were produced and are maintained by KB Transportnogo Mashinostroeniya (Design Bureau for Transportation Machine Building).

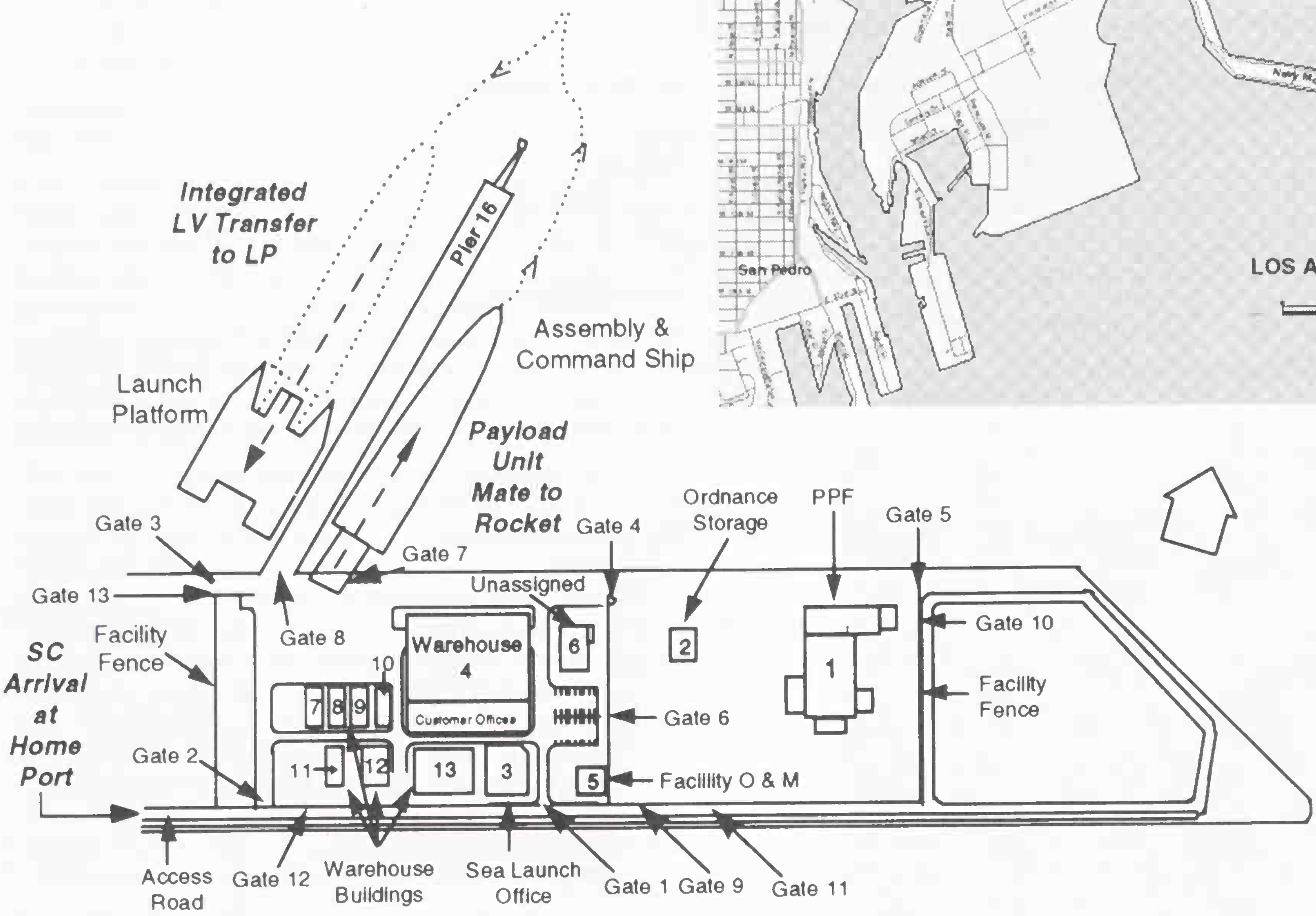
Several additional systems are added for Sea Launch's Zenit-3SL configuration. RSC Energia provides the Block DM-SL upper stage. RSC Energia produces the 11D58M engine at the Voronezh plant. Boeing Commercial Space Company of Seattle, Washington, produces the interface skirt/payload structure and payload fairing. The assembly and command ship and the launch platform were produced by Kvarner Maritime a.s. (today known as Kvaerner Group) in Glasgow, Scotland, and in Stavanger, Norway, respectively. The assembly and command ship was built from the keel up to specific Sea Launch requirements, based on a standard roll-on/roll-off container ship design. A former oil rig was refurbished and modified to create the launch platform. Launch equipment specific to the Zenit 3SL was added at shipyards near St. Petersburg, Russia.

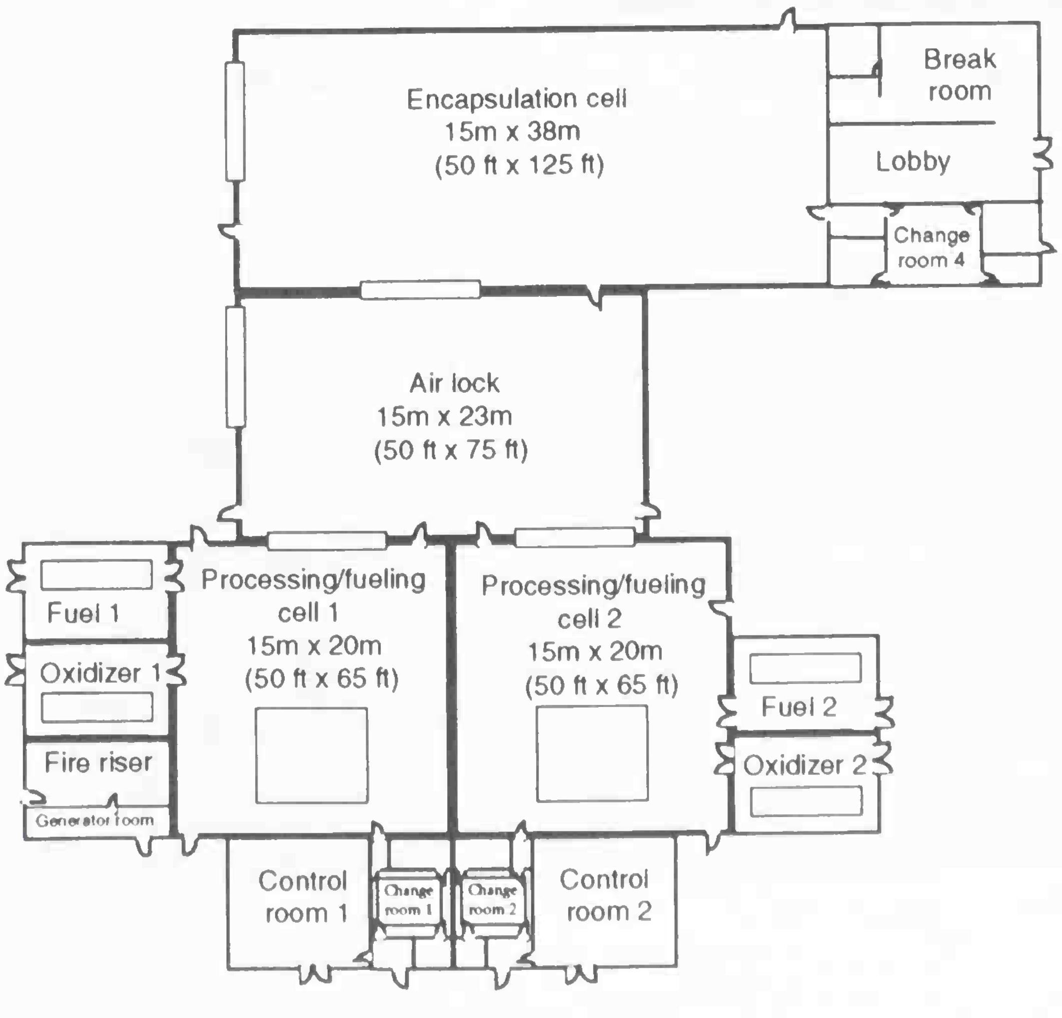
Production and Launch Operations

Launch Facilities

Map of Los Angeles Sea Launch Home Port (right) and Home Port Layout (below)

Courtesy Sea Launch.

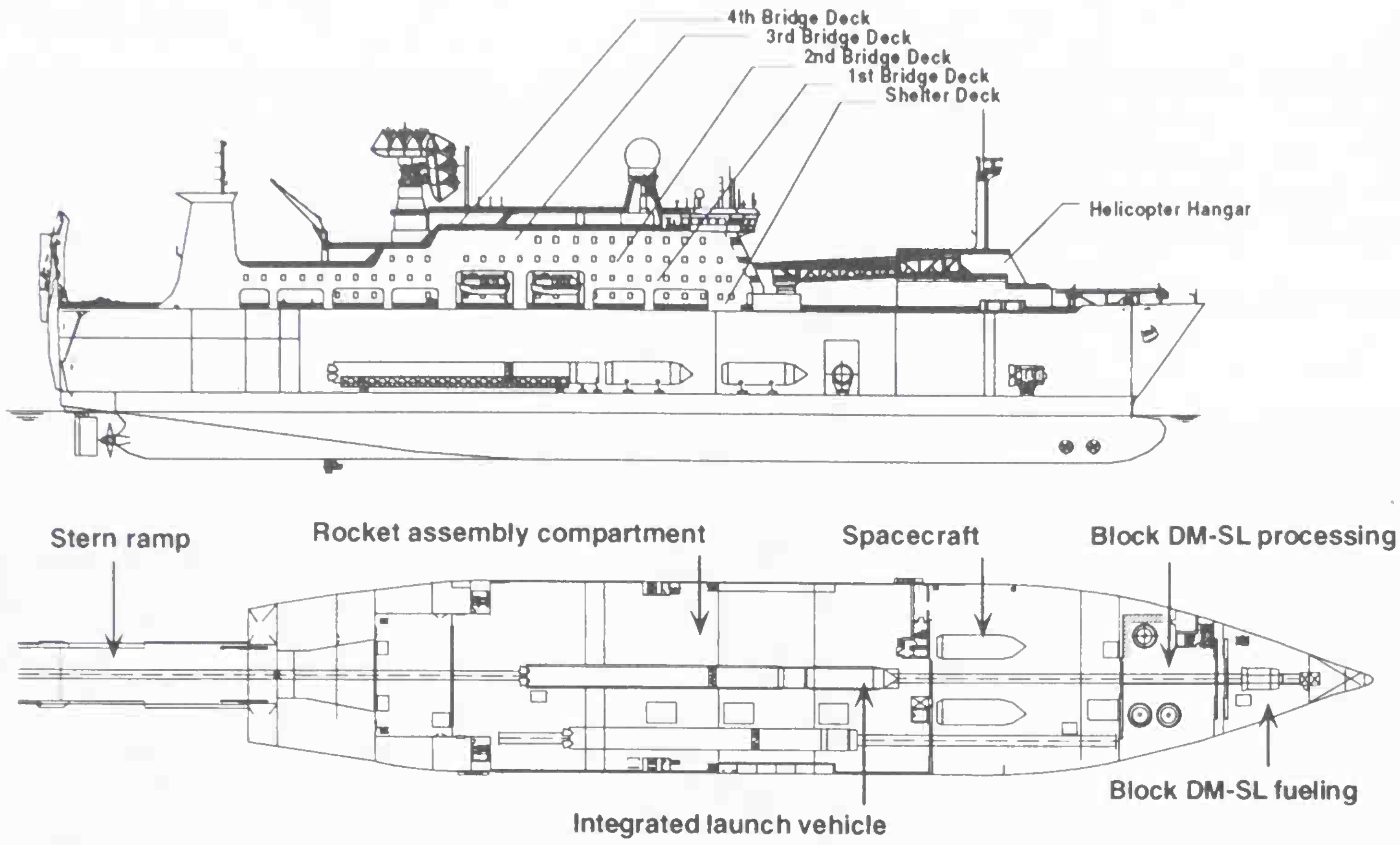




Courtesy Sea Launch.

Layout of Sea Launch Building 1, Payload Processing Facility

Production and Launch Operations



Courtesy Sea Launch.

Sea Launch Assembly and Command Ship Layout

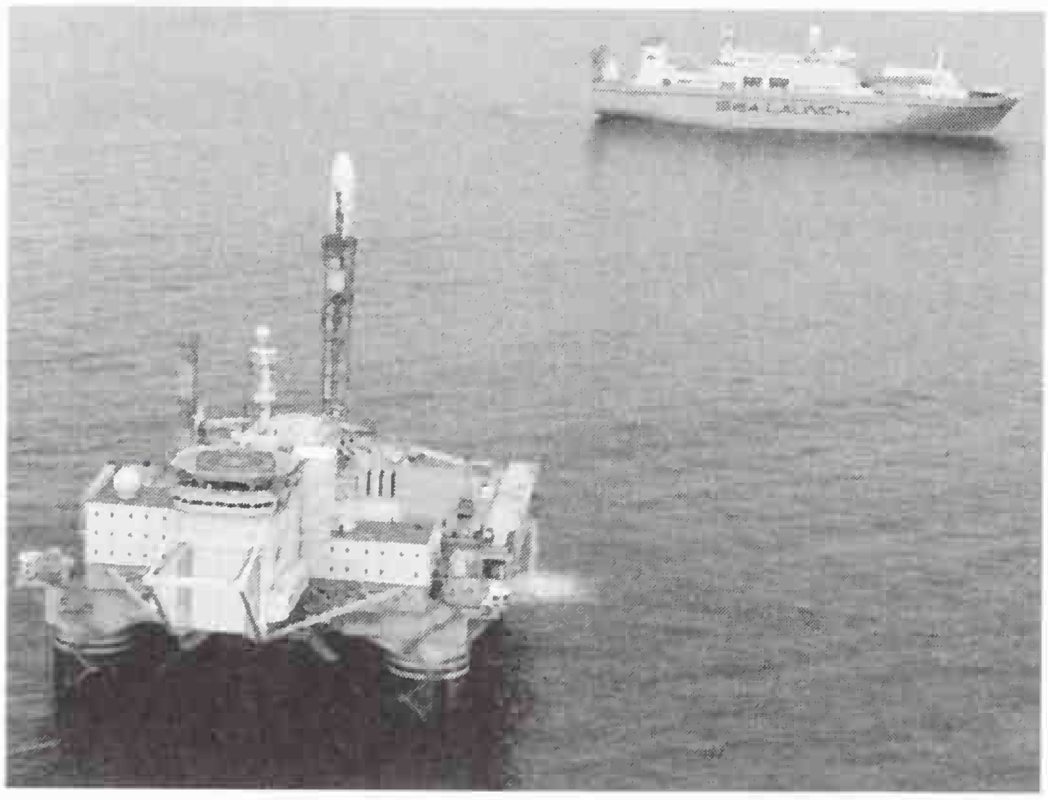
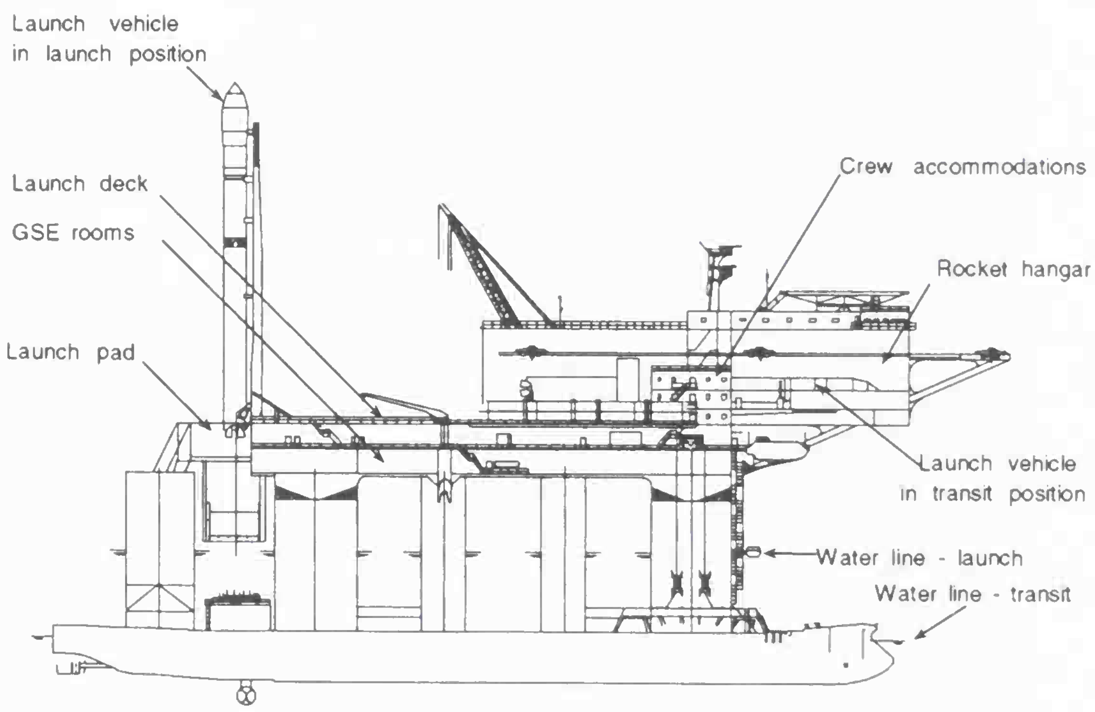


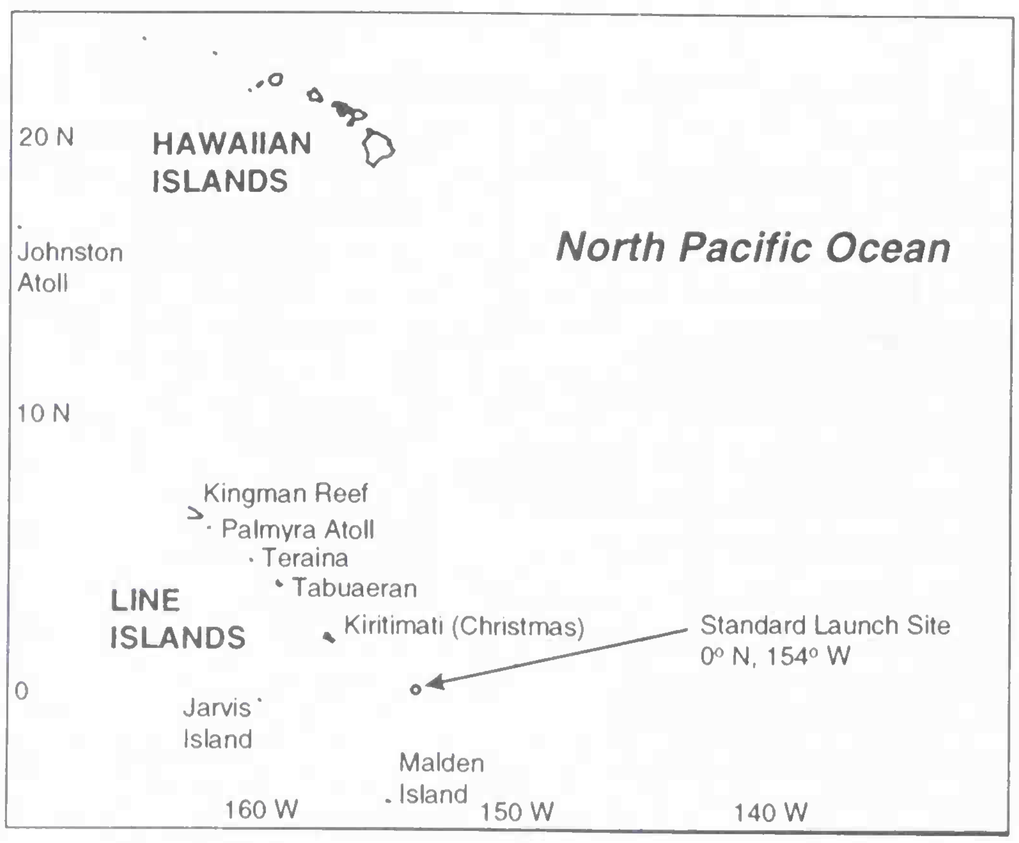
Photo courtesy Sea Launch.

Sea Launch Assembly and Command Ship and Launch Platform



Courtesy Sea Launch.

Sea Launch Launch Platform



Courtesy Sea Launch.

Production and Launch Operations

Launch Operations—Baikonur

Zenit 2 is launched from Baikonur. For background on the Baikonur Cosmodrome, please refer to the Spaceports chapter.

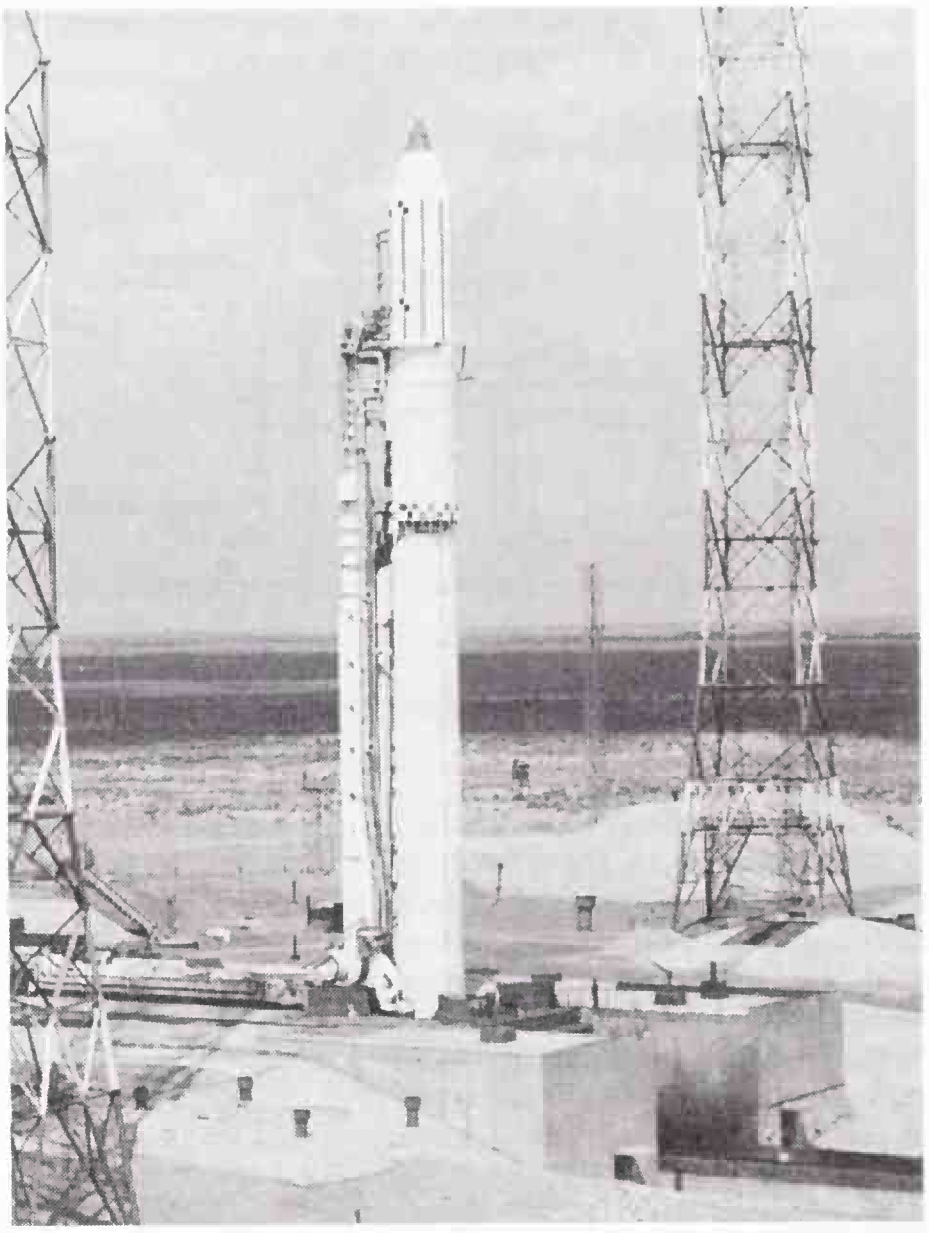
Rocket stages arriving from Ukraine on railroad cars can be put in storage or delivered directly to the Zenit Assembly and Test Building at Area 42. Three rail cars are used, one each for the first stage, second stage, and fairing. All vehicle processing takes place in a horizontal orientation. Using overhead bridge cranes, the elements of the launch vehicle are removed from the rail shipping cars and transferred to special mating trolleys that are capable of transitioning or rotating the stages to ensure proper alignment for mating. Any additional required flight hardware is integrated onto the stages, the stages are mated together, and the vehicle undergoes electrical and other tests. Only 76 work hours are needed to prepare and integrate the arriving stages to the point where the spacecraft can be integrated. Following vehicle preparation, the spacecraft is integrated onto the vehicle, electrical connections are checked, and then the spacecraft is covered by the payload fairing. The integrated launch vehicle can then be transferred to the transporter/erector rail cars for transport to the launch pad. Spacecraft integration onto the launch vehicle requires 21 work hours (not counting separate spacecraft preparation activities), for a total of 91 h from arrival of the vehicle stages at the cosmodrome to the completion of vehicle integration. Launch typically occurs within 28 h of the fully integrated vehicle leaving Area 42 on the transporter/erector for the pad.

Spacecraft processing may be performed in Area 31, Area 254, or at other facilities used for commercial launch services of other launch vehicles at Baikonur.

Zenit 2 launches are conducted from Launch Complex 45, which includes two launch pads. However, Pad 45 Right suffered heavy damage when Zenit Flight 15 exploded after liftoff. It has not been repaired since, so all subsequent launches have used Pad 45 Left. Each pad is a large, flat, concrete structure including an exhaust duct with a water noise suppression system. The launch mount is a metal structure with four hold-down clamps, and fittings for automatic propellant filling and, if necessary, draining. The launch complex area includes storage facilities for LOX and kerosene, as well as pressurized helium, nitrogen, and air. Tall lightning towers protect the erected vehicle from lightning strikes.

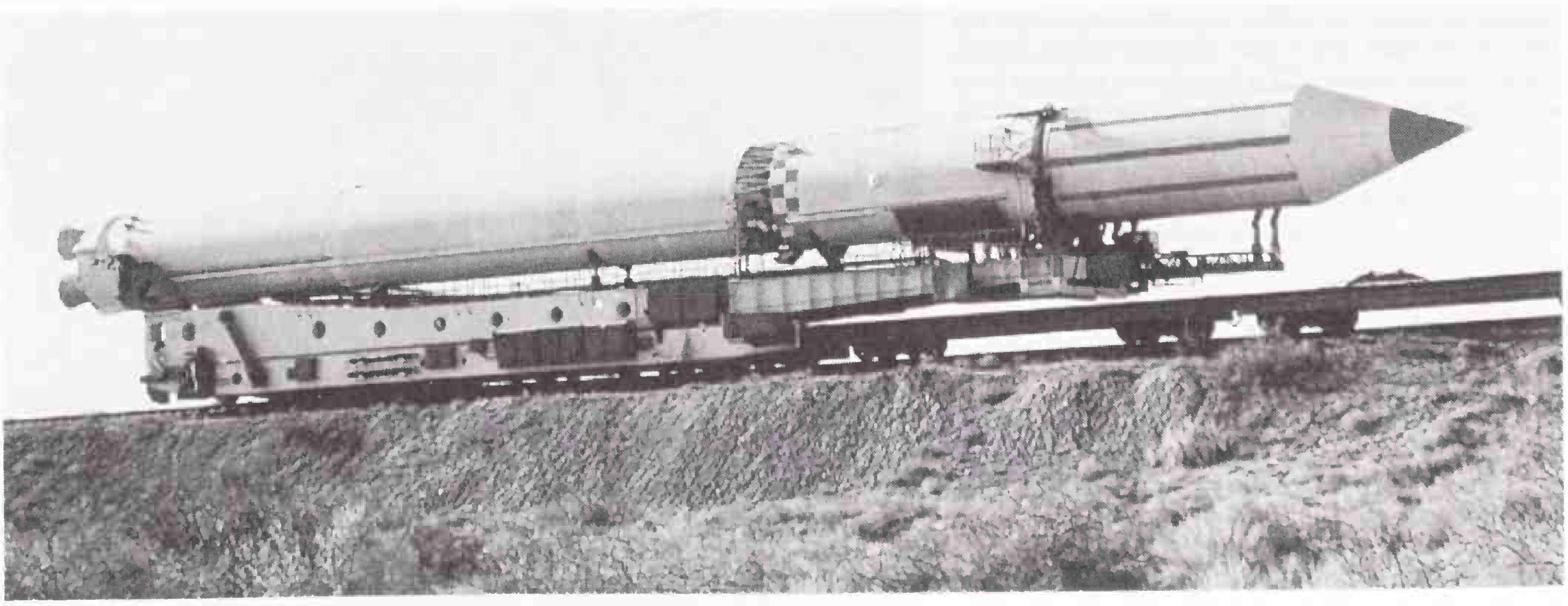
The launch complex does not include a traditional service tower or umbilical tower. Instead, the Zenit arrives in a horizontal position on a transporter/erector system mounted on two railcars. The transporter/erector automatically erects the vehicle into a vertical position using large hydraulic pistons. The vehicle is supported by a simple erector structure during this process. During and after the erection process the electrical, pneumatic, hydraulic, cooling, and propellant loading connections automatically mate with the appropriate interfaces at the base of the launch vehicle and at the top of the first stage. Once the vehicle is fully integrated, the erector is retracted and moved away from the pad, leaving only a small, simple tubular cable mast that carries the electrical and pneumatic connections between the vehicle and the launch mount. There is also a movable service tower that is capable of enclosing the upper portion of the launch vehicle. However, this was built to support planned manned missions, and while it is available for conventional spacecraft, it is not generally needed.

Upon liftoff the vehicle supports and the autocouplers retract into fortified recesses inside the pad where they are protected from damage, and theoretically could be used to perform a subsequent launch in as little as 90 min. Fortified bunkers were built near the pad for storing integrated launch vehicles in support of such a salvo scenario.



Courtesy KB Yuzhnoye.

Zenit 2 Launch Pad at Baikonur



Courtesy KB Yuzhnoye.

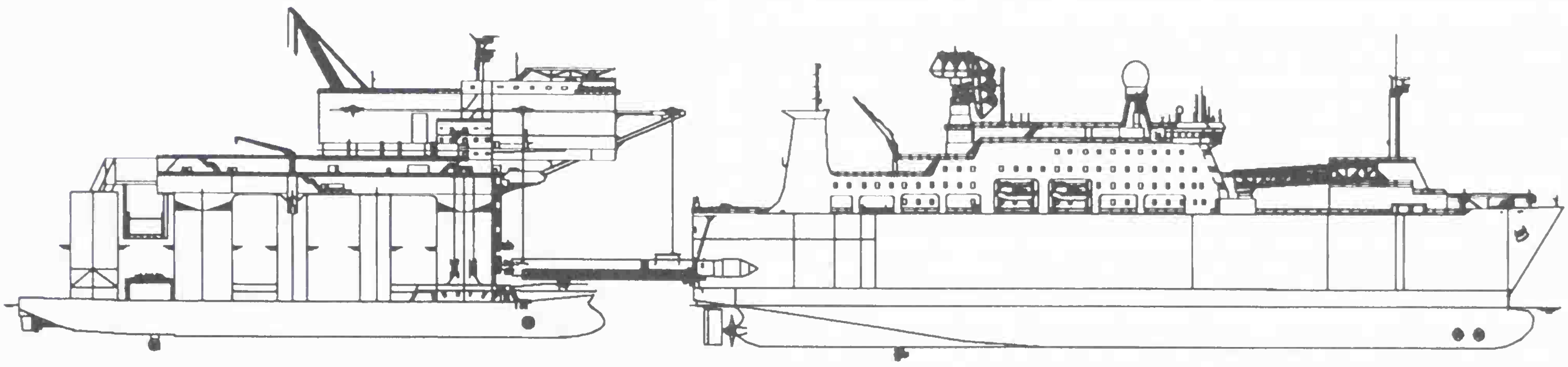
Zenit vehicles arrive at the launch complex in a horizontal position on a transporter/erector system mounted on two railcars.

Production and Launch Operations

Sea Launch

The Sea Launch Zenit 3SL is launched from a floating launch platform in the Pacific Ocean. The launch platform and asssembly and command ship are based at the Sea Launch Home Port, in Long Beach, California. Spacecraft processing is conducted in the payload processing facility at the Sea Launch home port. Two spacecraft processing and fueling cells are available. Payload encapsulation into the fairing is performed horizontally in an encapsulation cell before leaving the payload processing facility.

Launch vehicle assembly and processing is conducted aboard the assembly and command ship (ACS) while it is docked at home port. The ACS is a 200-m (660-ft) long, 30,800-t (34,000-U.S. tons) vessel specifically designed for Sea Launch, based on the basic structure of a roll-on/roll-off cargo ship. Zenit 3SL stages are loaded onto the ACS through a large stern ramp. The rocket assembly compartment takes up most of the main deck. Here the Zenit stages are integrated and tested horizontally. The Block DM-SL is processed and fueled in a special compartment in the bow. The Block DM-SL and the encapsulated payload in its payload fairing are then mated to the rocket. The ACS has room to store three integrated launch vehicles on the main deck. Once the launch vehicle is integrated, it is lifted by crane into a hangar on the launch platform. The *Odyssey* launch platform (LP) is a refurbished oil platform. It is self propelled, with directional thrusters for maintaining its position using input from a GPS navigation system. During transit it has a displacement of 27,400 t (30,100 U.S. tons).



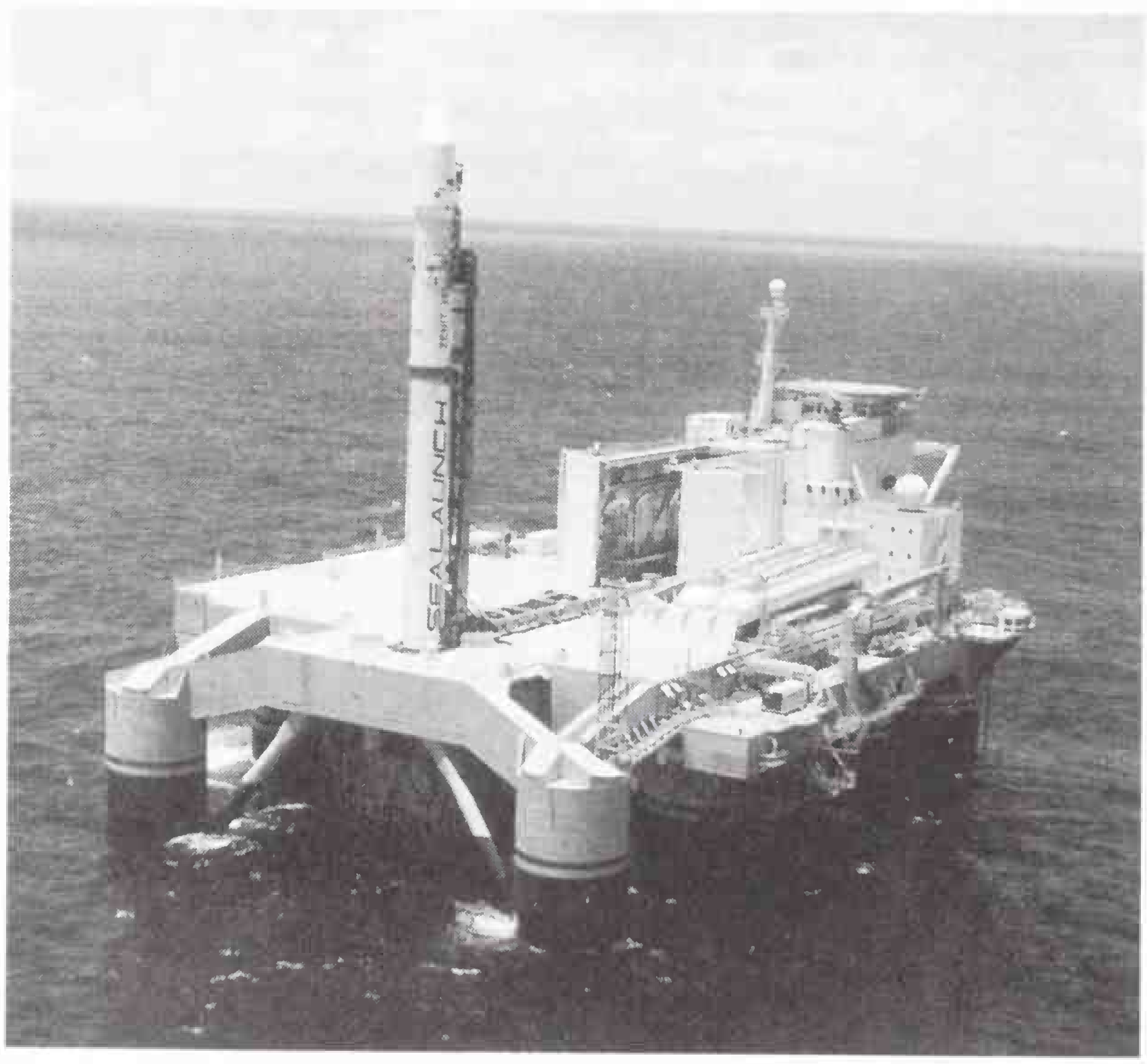
Courtesy Sea Launch.

Vehicle Transfer from Assembly and Command Ship to Launch Platform

Once the launch vehicle has been loaded on board the launch platform, both the LP and ACS sail to a location in the Pacific Ocean on the equator, approximately 1930 km (1200 miles) south of Hawaii. The voyage of the LP takes 10–12 days, and launch occurs on the third day after reaching the launch site. The ACS leaves port several days after the LP and catches up with the LP en route. Like the LP, the ACS has accommodations, conference rooms, and a cafeteria for marine and launch crews, and for customer representatives. During transit launch rehearsals are conducted, simulating prelaunch and postlaunch operations, and testing communications links to the customer's spacecraft control center. When the vessels reach the planned launch point, the LP is partially submerged by taking on water into tanks in the pontoons. This increases its stability for launch operations, holding itself level to within 1 deg. The ACS and LP moor alongside each other. A connecting bridge is extended from the ACS to the LP to allow prelaunch processing personnel access to the LP. Final spacecraft hands-on operations such as ordnance arming are accomplished and payload fairing doors are closed. A launch readiness review is conducted before initiation of launch operations. The launch is controlled from the launch control center, housed in an upper deck of the ACS.

When cleared for launch, the hangar doors are opened, and the Zenit 3SL is automatically rolled in a horizontal position to the launch pad at the opposite end of the LP deck. As it moves to the pad and is raised to the vertical position, electrical, propellant, hydraulic, and pneumatic lines connect automatically using the system originally designed for the Zenit 2. At this point, all but essential launch vehicle personnel leave the LP, and the ACS is repositioned 6.5 km (4.0 mi) away. Once final prelaunch checkouts are complete, the remaining personnel are transferred to the ACS by helicopter. From this point on, the LP and launch equipment are controlled remotely from the ACS. Propellant loading begins at T–2 h 20 min. At T–17 min, the transporter/erector is lowered from the launch vehicle and returns to the hangar where the door closes behind it. If the launch is scrubbed, the vehicle must be drained of fuel, returned to the hangar, and reinitialized, which can take up to four days. The LP carries enough consumables for three launch attempts.

Following ignition, once the computer determines that the first stage engine is operating properly, hold-down clamps are released and the vehicle lifts off. As on the Zenit 2 the launch supports and autocouplers retract into protected recesses inside the pad, and are protected from damage. The ACS has both optical tracking and radio telemetry systems. Once in orbit, the Block DM-SL can relay telemetry data to tracking facilities in Russia, and a TDRSS antenna on the Boeing-buildt payload structure can transmit Block DM-SL telemetry to the U.S. satellite tracking system.

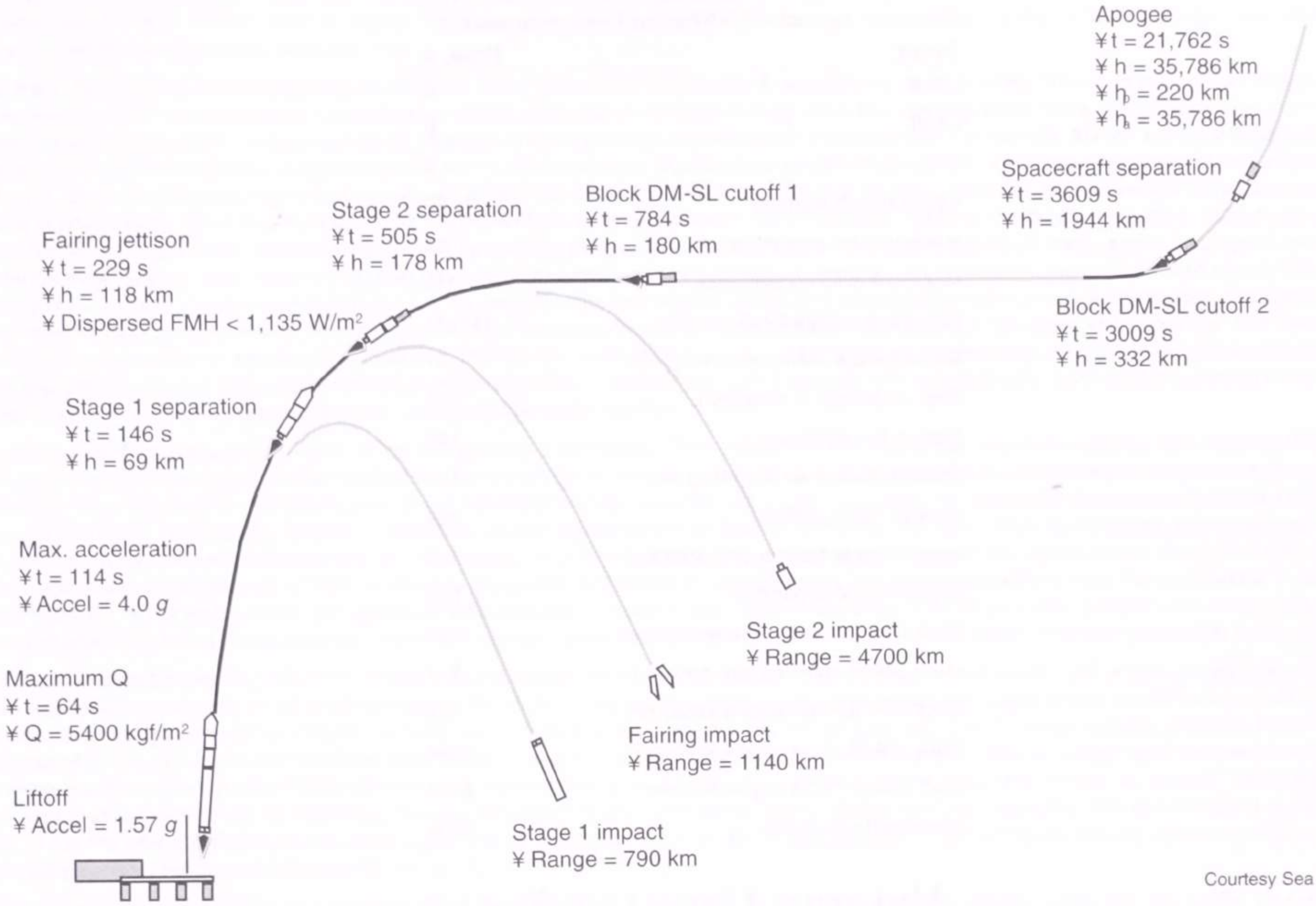


Courtesy Sea Launch.

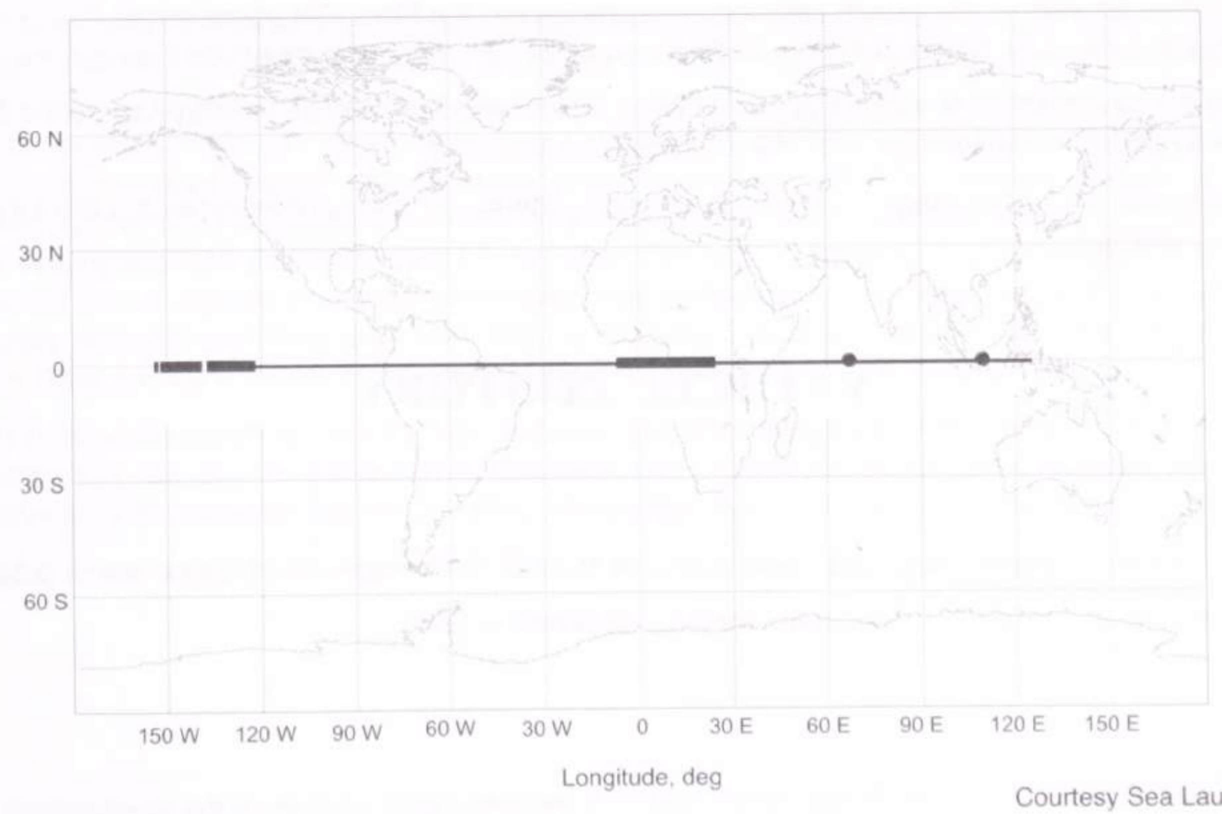
Zenit 3SL on Sea Launch's floating launch platform in the Pacific Ocean.

Production and Launch Operations

Flight Sequence



Courtesy Sea Launch.



Courtesy Sea Launch.

Production and Launch Operations

Flight Sequence

Zenit 3SL Typical GTO Mission Event Sequence

Event	Time, s
Liftoff	0
Liftoff	0
Begin pitchover	8
Roll to launch azimuth	10–20
Maximum dynamic pressure	64
Maximum axial acceleration	114
First-stage engine throttle to 50%	114–31
Second-stage vernier engine ignition	141
First-stage engine shutdown	143
First-stage separation	146
Second-stage main engine ignition	151
Payload fairing jettison	229
Second-stage main engine shutdown	429
Second-stage separation	505
Block DM-SL middle adapter jettison	506
Block DM-SL main engine ignition 1	514
Block DM-SL main engine shutdown 1	784
Block DM-SL main engine ignition 2	2584
Block DM-SL main engine shutdown 2	3009
Spacecraft separation	3609

Vehicle Upgrade Plans

In a project known logically as "Land Launch" efforts are underway to operate the Zenit 3SL from Baikonur. This would provide only around 3600 kg (8000 lbm) to a GTO orbit comparable to the standard Sea Launch delivery orbit. However, Baikonur launches would also be less expensive, making the option attractive for smaller satellites. Required tasks include modifications to the Zenit 2 launch pad (repeating the modifications that were made to the launch hardware used by Sea Launch on the launch platform), and changes to the Block DM upper stage. RSC Energia has begun work on the improved Block DM-SL B for this purpose. The first launch from Baikonur could occur as early as mid-2006 if a customer can be found.

Sea Launch is also considering the development of a larger payload fairing, which would provide an industry-standard 4.57-m (15-ft) diameter payload volume. The new fairing will be available no earlier than 2007 if a customer can be found.

Sea Launch is also considering launches of a two stage Zenit from the launch platform, for heavy-lift launches to LEO for government customers pending serious interest from those customers.

Vehicle History

Vehicle Description

•Zenit 2 Two-stage launch vehicle for intermediate and heavy payloads to LEO. Both stages use LOX/kerosene propellants.

•Zenit 3SL Three-stage version with Block DM-SL upper stage for large payloads to GTO.

Historical Summary

When the Zenit first appeared in 1985 it was the first new Soviet launch system to become operational since the development of the Proton almost 20 years earlier. Zenit was developed in conjunction with the Energia heavy-lift launch vehicle. The Zenit first stage shares its propulsion system and basic structural design with the large liquid strap-on boosters of Energia. Using a new second stage and payload fairing of its own, Zenit serves as a medium- to heavy-lift space launch system in its own right, with a payload capability halfway between Soyuz and Proton. In this role it was intended to serve as a modern replacement launch system for many Soyuz payloads that were beginning to outgrow the older launch vehicle and also for some Proton payloads. These planned missions apparently included a number of military satellites as well as some manned and space-station-related spacecraft. As a result of these requirements, Zenit is the only successful Russian or Ukrainian system that was developed solely as a space launch vehicle and was not based on the design of a missile. The failed N-1 and the quickly abandoned Energia are the only other examples.

Vehicle History

Because Zenit is one of the most modern of the Soviet launch systems, its design shows many improvements compared with older Soviet vehicles. The Zenit uses only two stages, each fueled with kerosene and oxygen propellants that are safer and more environmentally friendly than the storable propellants used on launch systems such as Proton and Cyclone. The propulsion system is much more advanced than the previous LOX/kerosene engines, such as those used on the Soyuz, for example. The RD-171 engine uses a high-pressure staged combustion cycle that results in 25% higher sea level specific impulse than the older RD-107 used on Soyuz. Zenit also demonstrates improved technologies in its highly automated launch infrastructure. Zenit can launch within 90 min of reaching the launch pad.

The first 11 flights of Zenit were designated as test flights. Many of these flights tested Zenit capabilities for unusual orbits. In some cases, very low perigee orbits were used to ensure that test hardware would quickly reenter as a result of atmospheric drag rather than creating orbital debris. The unusual mission profiles of many of these flights caused Western analysts to misinterpret them as undocumented launch vehicle failures. For example, the first two Zenit flights were suborbital tests, in which the trajectory perigee was within the Earth's atmosphere. Western observers interpreted one or both of these launches as failed orbital launch attempts, particularly because debris from the second launch was tracked in a very low orbit for a few days. However, later successful flights also produced debris in orbits slightly higher than the target orbit. The fourth flight, carrying Kosmos 1714 in December 1985, tested Zenit's capability to lift payloads to elliptical orbits with apogee in the northern hemisphere. The resulting perigee was very low, and the inclination, spacecraft mass, and orbit apogee were similar to the previous Kosmos 1697 mission and other subsequent missions at 71-deg inclination; therefore many Western analysts have assumed that this was another failed launch. The standard theory was that the second stage had failed to restart for a circularization burn, because the resulting orbit appeared to be a transfer orbit to reach the orbit used by Kosmos 1697. However, information released after the collapse of the Soviet Union indicates that these interpretations were incorrect. In addition to the disclosure that this was a test flight, it turned out that the Zenit second stage does not, in fact, have a restart capability and therefore uses a direct injection mission profile rather than a transfer orbit with a separate circularization burn. As a result, the inferred failure mode does not exist. Yuzhnoye reports that the second stage functioned as planned.

Ironically, after completing the 11 flights of the test program successfully, Zenit suffered three consecutive propulsion failures from 1990 to 1992. The first was caused by lubricating oil that dripped into the first-stage engine and entered the oxidizer manifold sometime after it completed its acceptance test firing. When the engine ignited for launch the oil came into contact with LOX, causing an explosion that destroyed the vehicle and the pad at Launch Complex 45 Right. The pad has not been repaired, so all subsequent launches have used Pad 45 Left. As a result of this failure, additional engine covers are now used. The remaining two failures were caused by a material change in a rotating component of the second-stage oxygen turbopump. The new material was stronger but had different thermal properties. The material change passed horizontal engine firing tests, but in a vertical orientation, as in flight, the new material would rub against an adjacent part, creating sparks that caused a fire in the engine. The problem was not properly identified until after the second flight, when the failure was reproduced by testing in the vertical orientation. The design was then modified to fix the problem.

Zenit was intended to become one of the primary Soviet launch systems once it became operational, with a launch rate of 20–40 flights per year (this total may include production of the Energia strap-on boosters). However, this did not occur. Some planned programs that would have used Zenit never materialized, and government funding for the three-stage Zenit 3 version, which could deliver payloads to GTO, never became available. Following the collapse of the Soviet Union, Zenit is in the awkward position of being a primarily Ukrainian launch vehicle, while its primary user has been the Russian government. Russia now prefers to launch most of its spacecraft on domestically built vehicles such as Soyuz and Proton. As a result, Zenit's flight rate has remained low, and its use for launching Russian spacecraft to the International Space Station has been canceled. The planned Zenit launch complex at the Plesetsk cosmodrome was never completed and was later adopted for Angara program. Zenit continues to carry occasional Russian government payloads, but its survival depends now on its commercial viability.

Given its high payload capacity and modern design, Zenit has been a promising candidate for commercialization since the late 1980s, although success has come slowly. In 1989, the privately owned Cape York Space Agency (CYSA) selected the planned three-stage Zenit 3 to provide commercial launch services from a proposed new launch site located on the eastern coast of Australia's Cape York. Under this arrangement, the former Soviet space marketing agency, Glavkosmos, was to have supplied the Zenit vehicles and provided training for the launch site crew, with marketing, launch site management, and operations to have been handled by CYSA and United Technologies of the United States. However, because of a lack of investor support, this plan eventually faded away. As a result, introduction of the Zenit 3 was delayed until the late 1990s.

In 1995 the new Sea Launch joint venture announced plans to provide commercial Zenit 3 launch services to GTO. Sea Launch proposed a unique system of launching the Zenit 3 (now the Zenit-3SL) from a floating launch platform in the South Pacific. Various aspects of the Zenit made it the most suitable large vehicle for launches from the floating launch complex: the automated launch system, the lack of strap-on boosters, the use of environmentally friendly LOX and kerosene propellant, and the minimal launch pad infrastructure. Sea Launch is led by Boeing of the United States, which has a 40% share of the venture. Boeing provides marketing and overall integration services and produces the payload unit. RSC Energia of Russia has a 25% stake and provides the upper stage. SDO Yuzhnoye and PO Yuzhmash collectively have a 15% share and provide the two lower stages from the Zenit 2. Kvaerner of Norway has a 20% share and provided the oceangoing ACS and the floating LP that gives Sea Launch its name. The participation of a major western aerospace company like Boeing has made the Sea Launch venture more successful than previous attempts to commercialize Zenit. Eighteen launches were sold to Hughes and Loral at the start of the project, and additional sales were later made to PanAmSat, Intelsat, Echostar, and Inmarsat. The first demonstration launch was successfully carried out in March 1999, and the first operational launch was successfully conducted in October 1999.

The Zenit 2 is also marketed for commercial launch services for LEO payloads. The first contract was signed between SDO Yuzhnoye and Globalstar, for three launches each carrying 12 satellites. Unfortunately, the first launch for Globalstar failed in 1998 and the remaining launches were instead performed by Delta and Soyuz vehicles. After the initial contract with Globalstar, the Sea Launch Company was given exclusive worldwide marking rights for all Zenit 2 flights (other than domestic government missions).

SPACEPORTS

Courtesy NASA

Cape Canaveral Air Force Station and NASA Kennedy Space Center as seen from Landsat 7.

Launch sites that are used for several different vehicles are described in detail in this chapter. For detailed information on sites that only support one launch system, please refer to the chapter on that launch system.

Launch Sites of the World

Launch Site	Location	Vehicles Supported	Available Inclinations	Annual Launches (1999–2002)	Description
Alcântara, Brazil	2.3° S, 44.4° W	VLS and VLM	2.2–100 deg	0–1	This site was developed to support space sounding rockets and Brazil's VLS launch vehicle. Alcantara has been considered for several launch vehicles from other countries because of its proximity to the equator.
APSC Cosmodrome, Christmas Island, Australia	10.4° S, 105.7° E	Aurora	11, 45–65, 90 deg–SSO	Not yet active	A new launch site in the Indian Ocean is proposed for the Russo–Australian Soyuz-derived Aurora launch vehicle.
Baikonur Cosmodrome, Kazakhstan	45.6° N, 63.4° E	Dnepr, Proton K, Proton M, Rockot, Soyuz, Molniya, Cyclone 2, and Zenit 2	50.5, 51.6, 53, 64.8–64.9, 66, 70–72.7, and 99 deg, others with dogleg or in-orbit plane changes	15–30	This was the launch site for Sputnik, Yuri Gagarin, and all subsequent Soviet/Russian crewed space flights. Located in Kazakhstan, but leased by Russia. Available launch azimuths are limited by land impact zones.
Barents Sea	Mobile; typical location is 69.3° N, 35.3° E	Shtil and Volna	79 deg demonstrated and others likely	0	A Russian Navy SLBM test area in the Barents sea near Murmansk is the standard location for Shtil and Volna launches. The launch submarine could launch from other areas in principle.
Cape Canaveral AFS, United States	28.5° N, 81.5° W	Athena I and II, Atlas II, III, and V, Delta II, III, and IV, Falcon, Pegasus XL, Taurus, and Titan IV	28.5–57 deg	8–13	CCAFS is the most active U.S. launch site. It is used primarily for GTO launches. It is operated by the U.S. Air Force 45th Space Wing.
China Lake Naval Ordnance Test Station, United States	35.4° N, 117.4° W	Pilot	?	Inactive	In 1958 NOTS developed a tiny space launch vehicle called Pilot and air-launched it from a fighter aircraft.
Edwards AFB, United States	34.5° N, 117.5° W	Pegasus (retired)	Polar	Inactive	Early B-52-launched flights of Pegasus originated here. Launches conducted over Pacific in Western Range.
Gando AFB, Canary Islands	27.6°N, 15.2°W	Pegasus XL	151 deg, polar and SSO	Inactive	This Spanish airfield in the Canary Islands supported the Pegasus XL launch of Spain's Minisat in 1998. No further launches are planned.
Guiana Space Center (Kourou), French Guiana	5.2° N, 52.8° W	Ariane 4 and 5	5.2–100 deg	8–12	Developed by France in French Guiana for Ariane. This site may also be used for Soyuz or other noncompeting launch vehicles in the future.
Hammaguir, Algeria	30.9°N, 3.1° W	Diamant (retired)	34–40 deg	Closed	A French missile range in the Algerian Sahara used for four orbital Diamant launches in the 1960s. This site was abandoned in 1967 after Algerian independence.
Jiuquan Satellite Launch Center, China	41.3° N, 100.3° E	LM-2C, LM-2E, and LM-2F	57–70 deg	0–2	China's oldest launch site went through a period of inactivity before being reactivated at the turn of the century to support China's human spaceflight program.
Kagoshima Space Center, Japan	31.2° N, 131.1° E	M-V	31–100 deg, dogleg required for polar orbits	0–1	This launch site is operated by ISAS for scientific missions.
Kapustin Yar, Russia	48.6° N, 46.3° E	Kosmos 3M	51 deg	0–1	The Soviet Union's least-used space launch site was mothballed in the late 1980s. It was reactivated in 1999 for one Kosmos launch.
Kodiak Launch Complex, United States	57.6° N, 152.2° W	Athena I and II	63–116 deg	0–1	This new site is operated by the state-sponsored Alaska Aerospace Development Corp. Its commercial pad can support small solid vehicles such as Athena, Taurus, or ICBM-derived vehicles.
Kwajalein Missile Range, Marshall Islands	18.5° N, 167.7° W	Pegasus XL	11–18 deg	0–1	This missile test range supports near-equatorial Pegasus launches. It is operated by the U.S. Army.

Launch Sites of the World

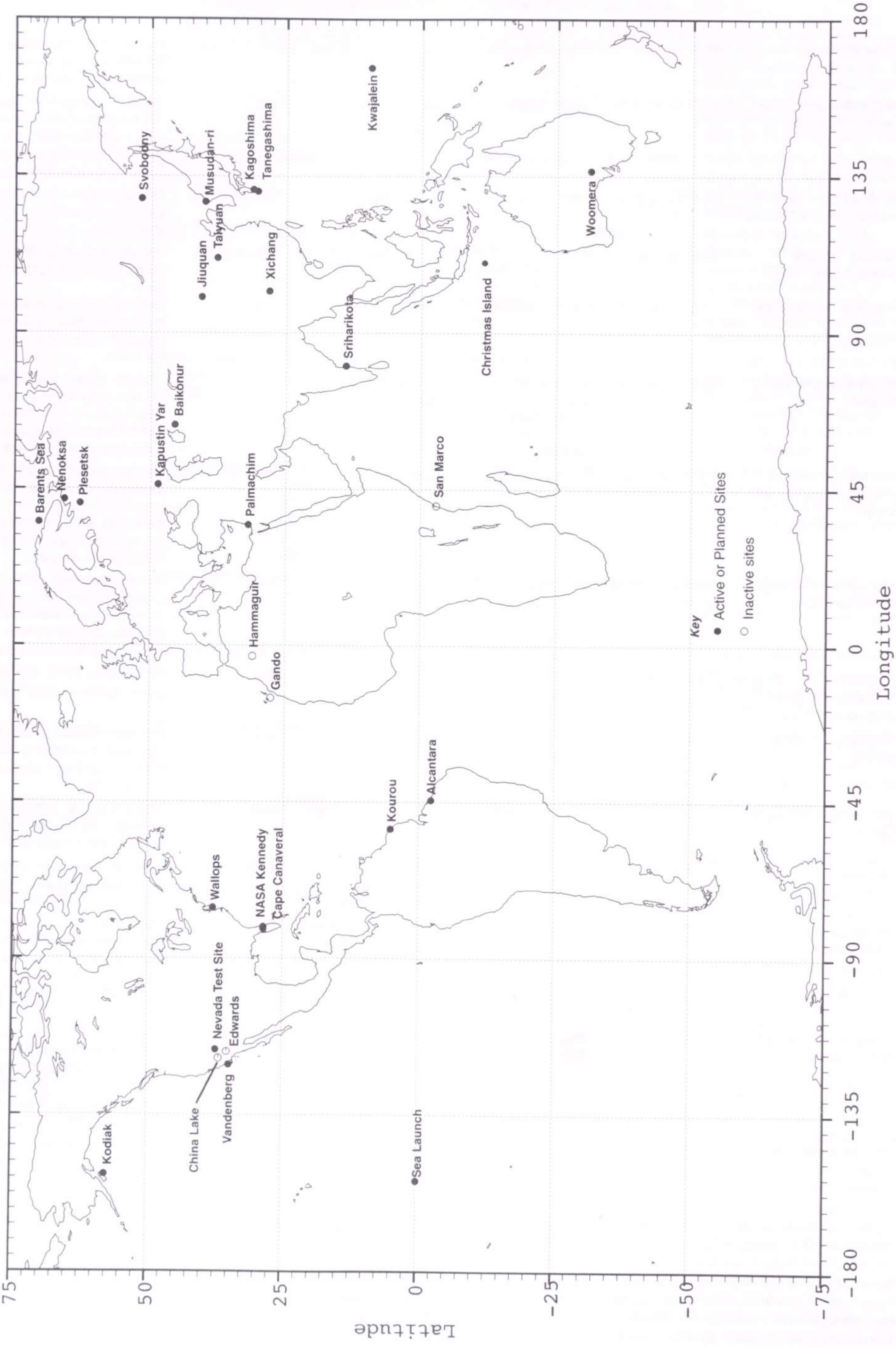
Launch Site	Location	Vehicles Supported	Available Inclinations	Annual Launches (1999–2002)	Description
Musudan-ri	40.9° N, 129.7° E	Taepodong	?	0	This facility on the eastern coast of North Korea was used for one failed space launch attempt in 1998.
NASA Kennedy Space Center, United States	28.5° N, 81.0° W	Space Shuttle	28.5–57 deg	3–6	This is the home of the Space Shuttle and the starting point for Apollo missions to the moon. It is operated by NASA and located north of Cape Canaveral AFS.
Nenoksa State Central Marine Test Site, Russia	64.6° N, 39.2° E	Shtil	77–88 deg	Not yet active	This land-based SLBM test facility is available to launch larger derivatives of Shtil.
Nevada Test Site, United States	37.2° N, 116.3° W	K-1	45–60 deg, 84–99 deg	Not yet active	This site is under development by Kistler to support its U.S. operations.
Palmachim AFB, Israel	31.9° N, 34.7° E	Shavit 1 and LK-A	143 deg	0–1	Shavit is launched to the west over the Mediterranean to avoid overflying neighboring Arab countries to the east.
Plesetsk Cosmodrome, Russia	62.9° N, 40.8° E	Kosmos 3M, Rockot, Soyuz, Molniya, Start-1, and Cyclone	63, 73, 82–86 deg and SSO	5–10	A Soviet military launch site, this site has accumulated the largest total number of launches in the world. Its available launch azimuths are limited by land impact zones.
San Marco Platform, (Broglio Space Center), Kenya	2.9° S, 40.3° E	Scout (retired)	2.9–38 deg	Inactive	Italy launched U.S.-built Scout rockets from this platform off the coast of Kenya until 1988. In 2002 the facility was renamed after Professor Luigi Broglio, the father of the Italian space program.
Sea Launch Odyssey, Pacific Ocean	Mobile; standard location is 0° N, 154° W	Zenit 3SL	0–100 deg	1–3	A Norwegian-built mobile oil drilling platform that was converted to support Zenit launches as part of Sea Launch joint venture. Its homeport is Long Beach, California.
Profressor Satish Dhawan Space Center SHAR (Sriharikota), India	13.9° N, 80.4° E	PSLV and GSLV	18–50 deg, SSO with dogleg	0–2	Operated by ISRO, this site supports all Indian space launches.
Svobodny Cosmodrome, Russia	51.8° N, 128.4° E	Start-1 and Strela	SSO	0–1	A former Russian missile base that has been converted to launch small ICBM-derived space launch vehicles in 1997.
Taiyuan Satellite Launch Center, China	37.8° N, 111.5° E	LM-4 and LM-2C/SD .	87 and 96–98 deg	0–3	This Chinese launch site supports smaller Long March versions for launches to polar and SSO
Tanegashima Space Center, Japan	30.4° N, 130.6° E	H-IIA	30–100 deg, dogleg required for polar orbits	0–3	This site is operated by NASDA and supports Japan's heavy launch vehicles and GTO launches.
Vandenberg AFB, United States	34.7° N, 120.6° W	Athena I and II, Atlas II and V, Delta II and IV, Falcon, Pegasus XL, Taurus, and Titan II and IV	63.4–110 deg directly from South Base	3–11	Operated by U.S. Air Force 30th Space Wing to support high inclination launches, this site was used primarily for military missions until the mid 1990s when commercial operations began.
Wallops Flight Facility, United States	37.9° N, 75.4° W	Pegasus XL Conestoga (retired)	38–55 deg	0–1	This launch site is operated by NASA for small launch vehicles and sounding rockets. The public–private Virginia Space Flight Center is developing a pad to support Athena and Taurus class launch vehicles.
Woomera, Australia	31.1° S, 136.6° E	K-1	45–60 deg, 84–99 deg	Not yet reactivated	Originally developed in the 1940s to support British and Australian missile and space tests, this site will be used by Kistler for K-1 operations.
Xichang Satellite Launch Center, China	28.2° N, 102.0° E	LM-2E, LM-3, LM-3A, LM-3B, and LM-3C	27.5–31.1 deg	0–4	This is the primary Chinese launch site for GTO payloads.

•Some launch sites span more than 0.1° in latitide or longitude, and may therefore have launch complexes at slightly different coordinates than those listed.

•Range safety approval is unique to each vehicle and mission orbit. Therefore, some inclinations shown above may not be available for specific orbits or launch vehicles. Some additional inclinations may also be available. Contact the launch service provider or range authority for more information.

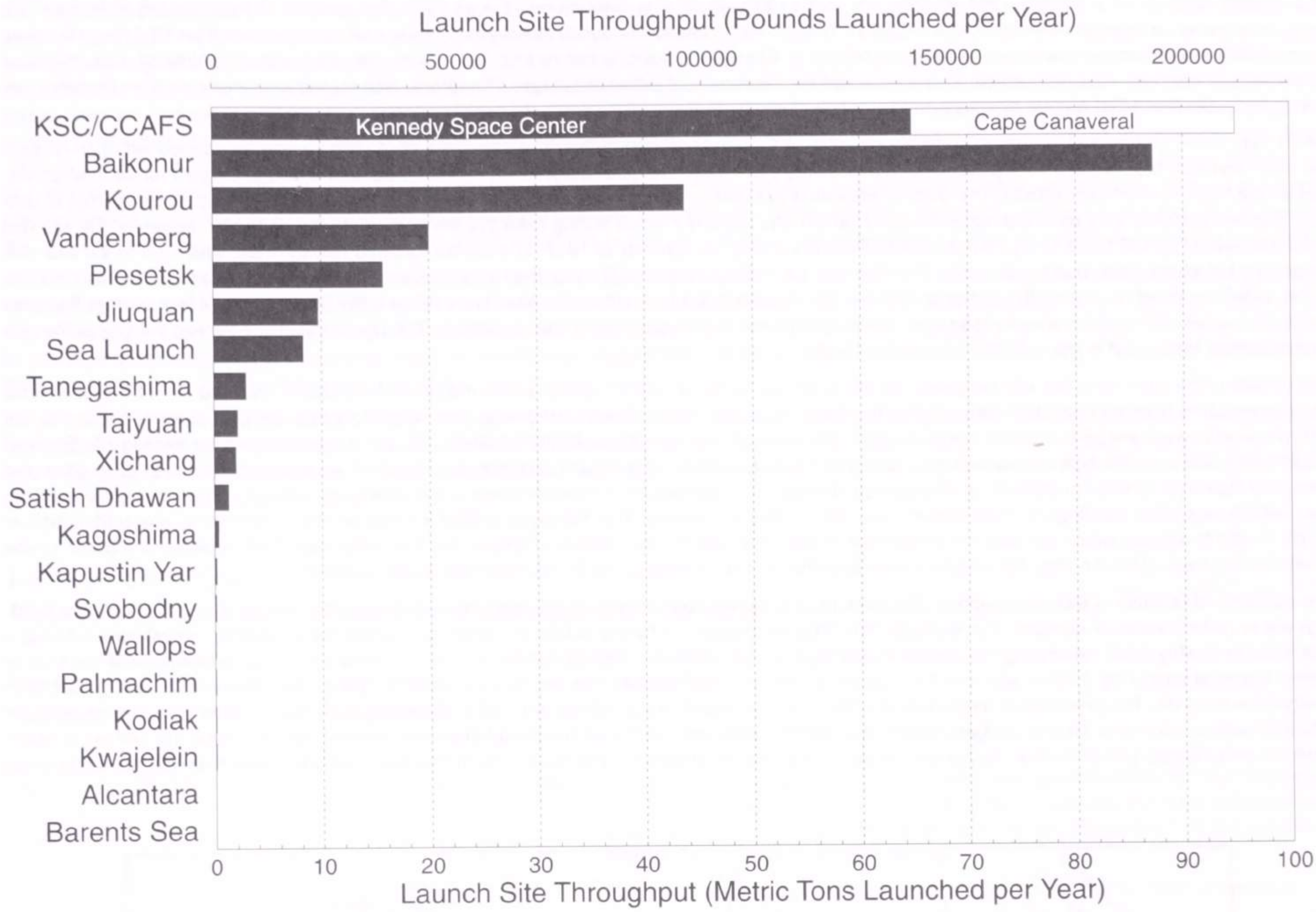
•The number of launches and operational status refer only to orbital launches from 1999 through 2002. Sites that are not active for orbital launches may actively support suborbital launches or aircraft.

LAUNCH SITES OF THE WORLD



Launch Sites of the World

Average Annual Payload Mass Launched from Active Spaceports, 1999–2002



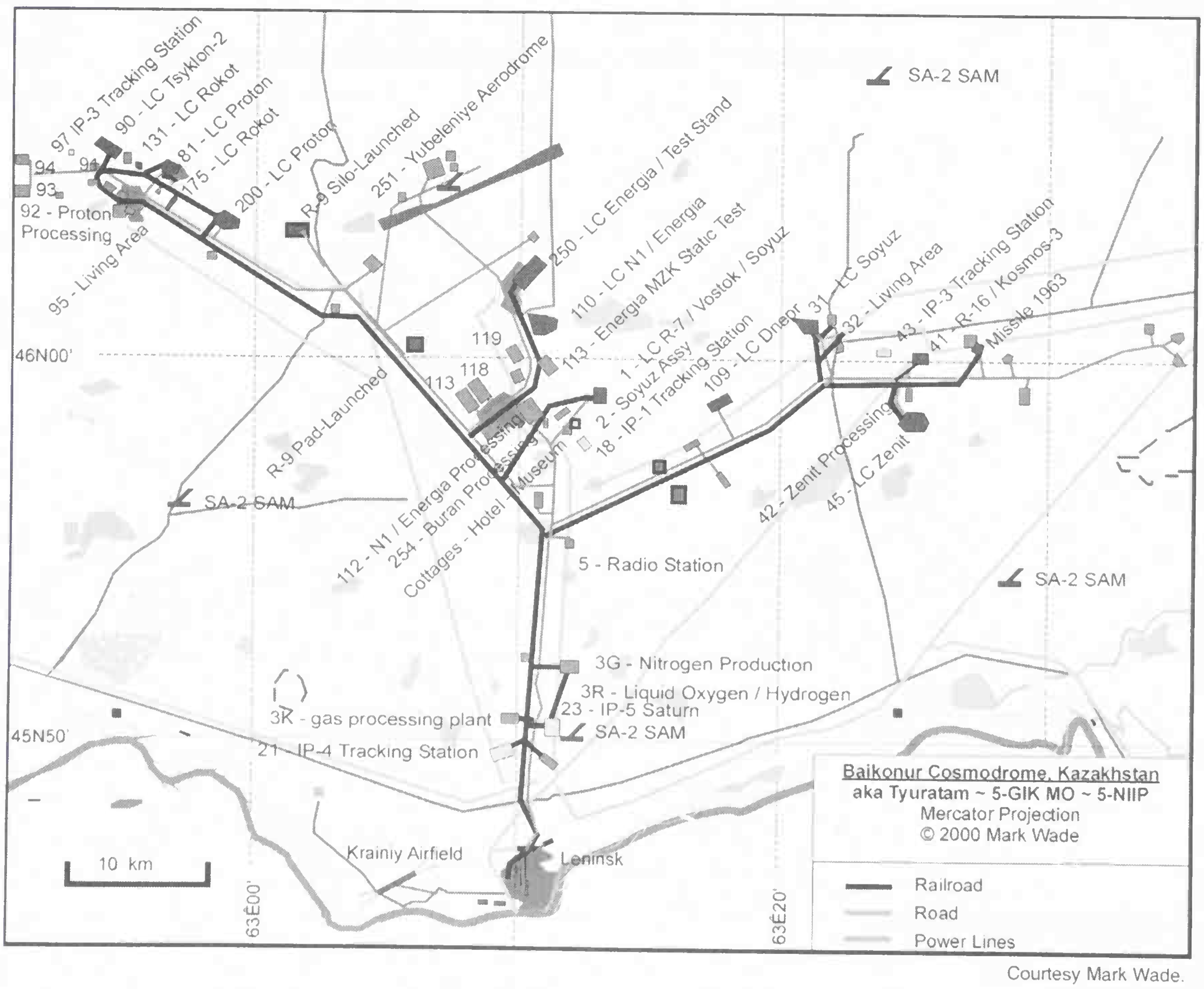
Baikonur Cosmodrome

Baikonur Cosmodrome Space Facility is one of Russia's two major space launch complexes. The site has also been known as Baykonur, Tyuratam, NIIP-5 (Fifth Scientific Research Firing Range) and GIK-5 (Fifth State Test Cosmodrome). Baikonur is located just east of the Aral Sea, in the Republic of Kazakhstan, approximately 2100 km (1300 mi) southeast of Moscow. Baikonur has been the launch site for all Soviet, and later Russian, human spaceflight programs, geostationary satellites, and missions to the moon and planets. Among its many space firsts, Baikonur is the site of the first launch of a satellite, Sputnik on 4 October 1957, and the first human into space, Yuri Gagarin on 12 April 1961. The location of Baikonur (45.6° N, 63.4° E), about as southerly a location as was possible within the Soviet Union, precludes launch directly into orbital inclinations of less than 45.6 deg. However, range safety restrictions (in particular, avoiding overflight of China), preclude launches into inclinations less than approximately 51 deg. Because Baikonur is landlocked, specific drop zones are reserved for the impact of jettisoned stages. Therefore, orbit inclinations available from Baikonur are strictly limited to those that can be acheived on the launch azimuths that are aligned with these drop zones.

On 31 May 1955, ground was broken to begin construction of the space launch facility. The site of the cosmodrome was an abandoned open-pit copper mine that may have served as a gulag labor camp during the 1930s. The cosmodrome was established near a village called Tyuratam, a rail station on the Moscow–Tashkent railway. The Soviets were slow to admit to the presence of the launch complex, and when they did, in an attempt to confuse Western analysts, they used the name Baikonur, which was actually a small mining town 370 km (230 mi) to the northeast. However, the site had been discovered by U-2 pilots flying from a base in Pakistan during the summer of 1957. The planes followed railway lines, which led to the site, discovering it before the base was made public. The first pad, Launch Complex 1 (LC-1) for the R-7 ICBM, was completed on 4 March 1957, and the first R-7 missile was launched shortly thereafter on 15 May. The missile failed 50 s into flight. After more failures, the first successful launch from Baikonur was on 3 August 1957. LC-1 is also the pad from which Sputnik and Yuri Gagarin were later launched. The pad is still in use serving the Soyuz/Molniya launch vehicle family and is now officially named the Gagarin Launch Complex.

The climate in the summer is hot with temperatures rising as high as 50°C (120°F), and in winter it is prone to blizzards and −40°C (−40°F) lows. This has required that Russian and Ukrainian vehicles be able to launch in harsh climate conditions. The large Y-shaped complex extends about 160 km (100 mi) east to west and 88 km (55 mi) north to south. The cosmodrome area totals 7360 km^2 (2840 mi^2) with a downrange zone totaling 104,305 km^2 (40,270 mi^2). Each vehicle type has its own processing and launch facilities. The vehicle processing and launch areas are connected to each other and the city of Baikonur by 470 km (290 mi) of wide-gauge railroad lines; rail being the principal mode of transportation, not road. Rockets are carried from their vehicle assembly buildings to their launch pads horizontally on railcars. The Yubileyny airfield located on the cosmodrome has a 52×3500 m (170×11,480 ft) runway, which can accommodate large transport aircraft for the delivery of spacecraft. The airfield was built to deliver the Buran space shuttle on the back of the An-225, the world's largest airplane, and as a landing site for the returning Buran space shuttle.

The condition of facilities at Baikonur is mixed. Facilities used to support commercial or ISS launches are reasonably well maintained, while other buildings are in various states of disrepair. For example, the massive preparation facility at Site 112 which processed the Energia/Buran vehicles suffered a roof collapse in May 2002, destroying the original Buran flight article and spare Energia hardware. One recurring maintenance headache is the theft of buried electrical cables by thieves who sell them as scrap copper. Nevertheless, the key facilities remain operational despite most being past their designed service life. The presence of multiple launch pads for Soyuz and Proton allows one pad to occasionally be taken offline for a year or more for refurbishment or upgrades. Limited budgets require that maintenance and repairs be performed in smaller batches without taking the pad out of operation for an extended period of time. Soyuz launch pad 1 was last refurbished in the late 1990s. It has been used for more than 400 launches since 1957.



Courtesy Mark Wade.

Baikonur Cosmodrome

Formerly Baikonur was operated by the Russian Space Forces (VKS). That organization merged into the Strategic Missile Forces (RVSN). Beginning in 1998, the RVSN began a process of handing over control of Baikonur to Rosaviakosmos (RAKA), the Russian civilian space agency. About 750 military specialists were transferred to Rosaviakosmos as part of the reorganization. Whereas day-to-day operations and maintenance of the launch complexes were previously performed by officers and enlisted men, today they are performed by the design bureaus that originally built the facilities.

The city of Baikonur, where cosmodrome employees and their families live, is located approximately 40 km (25 mi) south of the cosmodrome's main east-west service road. The city of Baikonur was originally known as Zarya with the original village of Tyuratam nearby. In 1958 Zarya's name was changed to Leninsk, which grew to be a Soviet "science city" of over 100,000 people at its peak—growing up alongside and absorbing the original Tyuratam. In December 1995 the name was changed to Baikonur. The city includes cosmonaut training facilities, laboratories, offices, movie theaters, a sports center and residences. Lodging conditions, food services, and other services sought by Western launch teams are spartan, though in recent years the growing number of launches of commercial satellites has resulted gradually in improved facilities and amenities. Communication services are available in the hotels regularly used by official launch site visitors. A direct satellite telephone and data link are available to Western spacecraft processing teams. Baikonur has its own post office, telegraph, telephone network, fax facilities, and domestic and international communications. Power outages are a frequent occurrence. Medical service is provided by a hospital complex, clinic, and a network of medical aid centers. Western emergency medical service, including air evacuation to European cities, is available. A museum includes the cottages of Korolev and the house where the original cosmonauts stayed before each flight.

The city of Baikonur's Krayner airport provides regularly scheduled passenger connections, though not daily, to and from Moscow; it is a 3-1/2 h flight. Western visitors must have a multiple entry or two-entry Russian visa. The most convenient way of getting from Baikonur to the cosmodrome is by road. Van transportation can be arranged, though road conditions are considerably below Western expectations and standards. The driving time from Baikonur to the cosmodrome spacecraft processing areas or launch sites ranges from 45 min to a couple of hours. Alternatively, a commuter train runs from Baikonur to the cosmodrome, once each way daily.

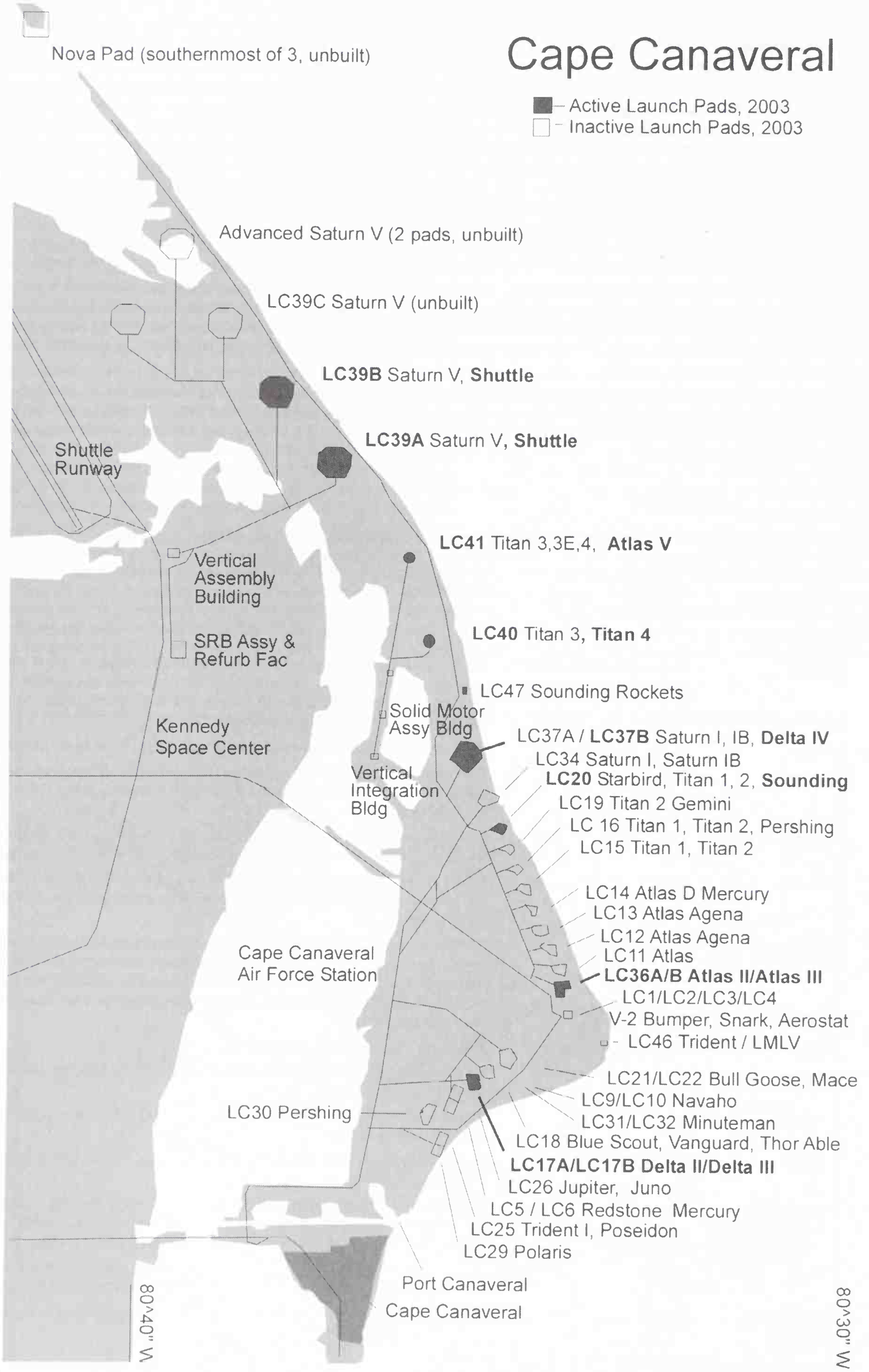
On 28 March 1994, following the collapse of the Soviet Union, Russia and Kazakhstan signed a long-term agreement whereby Russia would lease Baikonur from Kazakhstan for 20 years with day-to-day operations of the cosmodrome remaining under the control of the VKS. The lease price of $115 million per year was agreed to in October 1994 paid through a combination of barter of services, cash payments, and forgiveness of debt owed by Kazakhstan to Russia for infrastructure built during the Soviet era.

Launch Complexes

A large number of launch pads and missile silos are present at Baikonur. The following have been used for space launches.

Launch Complex	Pad	Vehicles Supported	Comments
LC 1	Pad 5	Sputnik, Vostok, Voskhod, Soyuz, Molniya	The first launch complex at Baikonur was developed initially for the R-7 missile and Sputnik launch vehicle. It is still one of two complexes that support Soyuz-class launches. LC 1 was used to launch Sputnik and Yuri Gagarin and is still used for most Russian manned space missions.
LC 31	Pad 6	Soyuz, Molniya	This is the second launch complex developed for the Soyuz family of vehicles.
LC 41	3 pads	Kosmos 2 and 3	Three pads were built for tests of the R-16 missile. One of these was converted to support test launches of the Kosmos 2 and later Kosmos 3 in the 1960s; this pad is no longer active.
LC 45	2 pads (Left and Right)	Zenit 2	Two pads were built at LC 45 for launches of the Zenit 2. In 1990, a launch failure shortly after liftoff heavily damaged pad 45 Right, and it has not been repaired.
LC 69	2 silos	R-36-O	Two or more silos at LC 69 supported launches of the R-36 missile, the precursor to Cyclone. At least one of these sites was used for orbital launches. This site is probably inactive.
LC 81	Pad 24 (81 Right) Pad 23 (81 Left)	Proton	Pad 24 was the first launch pad for Proton. It was refurbished by Krunichev in 1997 and will be the primary pad for Proton M and future commercial launches. Pad 23 was the second launch pad for Proton and was used for initial commercial launches. Pad 23 will be refurbished and serve as a secondary pad for future missions. (See also LC 200.)
LC 90	Pad 20, 21	Cyclone 2	Apparently, only Pad 20 is still active.
LC 110	2 pads	N-1, Energia	Initially developed to launch the N-1 lunar rocket, LC 110 was later converted to support Energia launches.
LC 131	Silo 29	Rockot	Probably previously used for RS-18 ICBM test flights, it has been converted to support launches of Rockot.
LC 175	Silo 59	Rockot	Probably previously used for RS-18 ICBM test flights, it was used for initial test launches of Rockot.
LC 200	Pad 39 (200 Left) Pad 40 (200 Right)	Proton	The second launch complex for Proton launch vehicles, it was developed to support a planned fluorine/ammonia upper stage. Pad 40 is apparently inactive. LC 200 is not used for commercial missions, which are conducted from LC 81.
LC 250	1 pad	Energia	The Energia test firing facility at LC 250 was also used as the launch site for the first Energia launch because the primary pads at LC 110 were not yet finished.
LC ?	Silo 108, 109, + others	Dnepr	Four missile silos north of Area 31 on the eastern side of Baikonur are available to support Dnepr launches.

CAPE CANAVERAL AIR FORCE STATION/NASA KENNEDY SPACE CENTER



Courtesy Mark Wade.

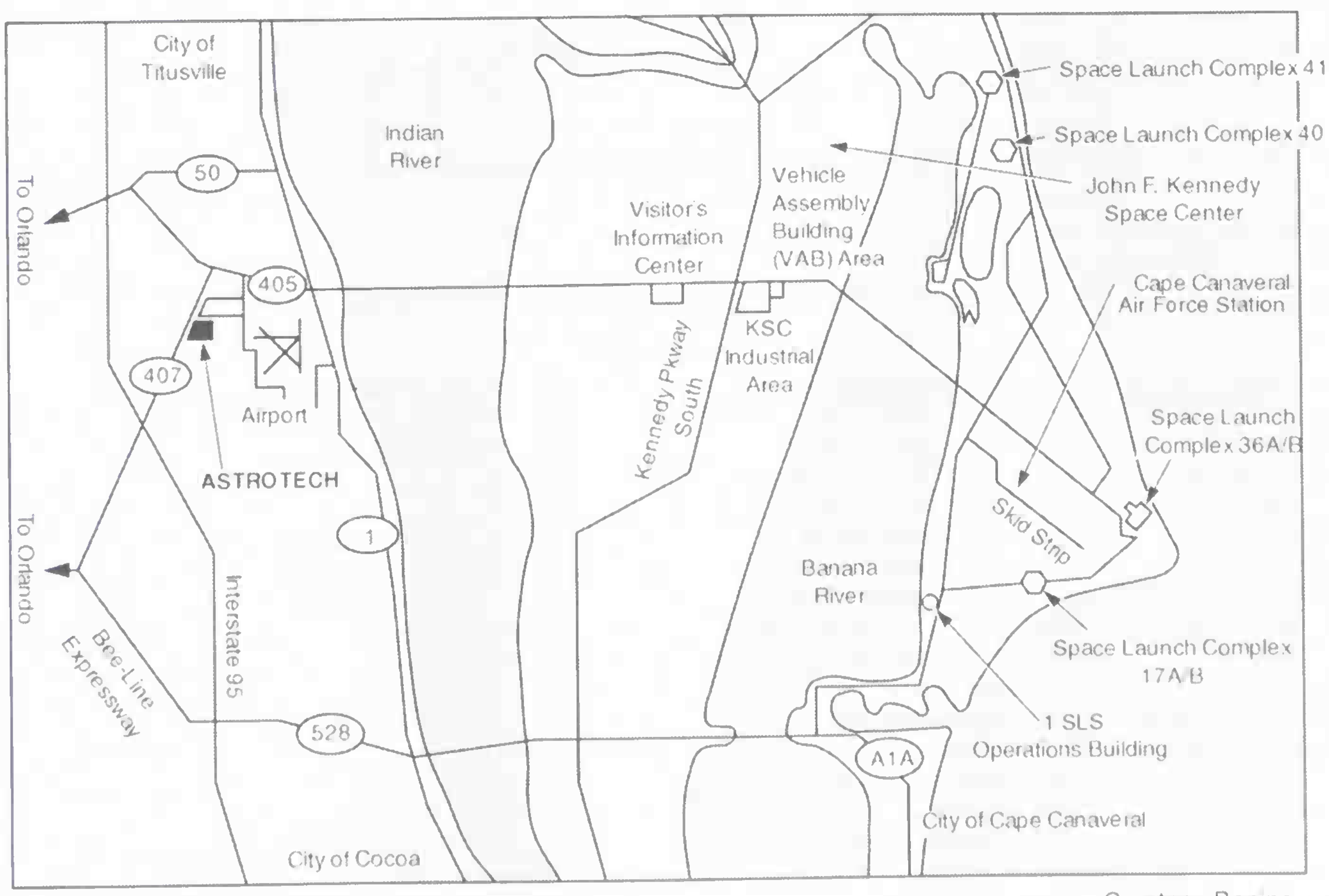
Cape Canaveral Air Force Station/NASA Kennedy Space Center

Cape Canaveral Air Force Station (CCAFS) and NASA Kennedy Space Center (KSC) are collocated on the central Atlantic coast of Florida in the United States. The sites have been used for all U.S. human spaceflights including the Apollo missions to the moon, all launches to GTO, and nearly all lunar and planetary science missions (the unique exception being the Clementine lunar mission launched from Vandenberg AFB). CCAFS is operated by the U.S. Air Force 45th Space Wing based at nearby Patrick AFB and supports a large number of expendable launch vehicles. KSC is operated by NASA. Only the Space Shuttle launches from KSC, using Launch Complex 39. KSC is actually located on Merritt Island, between the Florida mainland and the land that makes up Cape Canaveral. Therefore, the colloquial description of Space Shuttle or Apollo missions launching from "the Cape" is not accurate geographically. CCAFS and KSC share many facilities in common. KSC industrial and test facilities support many of the expendable launch vehicles launched from CCAFS, while CCAFS radars and telemetry stations support KSC launches. Collectively, the two facilities and the downrange tracking assets are known as the Eastern Range or the Cape Canaveral Spaceport.

CCAFS began as the Joint Long Range Proving Ground (JLRPG) for testing of missiles being developed by the joint military services of the U.S. Air Force, Navy, and Army. The JLRPG required a long firing range over the ocean with land or islands along much of the downrange flight path for command and tracking stations. A site based in Washington state with launches flying over the Aleutian Islands was rejected quickly because of the difficult logistics of operating in the cold and remote islands. The preferred site for the JLRPG was actually El Centro NAS, California, with launches flying south over Baja California, Mexico. However, the Mexican president refused to allow missiles to overfly his country, perhaps remembering the May 1947 launch of a V-2 from White Sands that crash-landed near a cemetery on the outskirts of Juarez. Fortunately, the British were willing to allow overflights of the Bahamas and to lease land on British-held islands in the Caribbean for tracking stations. As a result, the second choice of the site selection committee, a barren stretch of Florida coast called Cape Canaveral, became home to the proving ground. President Harry Truman signed Public Law 60 on 11 May 1949 creating the JLRPG at Cape Canaveral as a joint-service facility under the ultimate authority of the U.S. Air Force. The first experimental launches of two-stage Bumper rockets were conducted in July 1950 near what is now the Atlas SLC-36 complex. To test the winged Snark and Navaho missiles the total length of the range was extended to more than 8000 km (5000 mi), ending at Ascension Island in the mid Atlantic. In the late 1950s, winged missiles were surpassed by ballistic missiles such as the Jupiter, Thor, Atlas, Titan, and Polaris, all of which were tested at CCAFS. When the U.S. space program began in 1957, it was only natural that space launches use the facilities at CCAFS. Cape Canaveral has continued to be home to space launches and land- and sea-based missile tests since then.

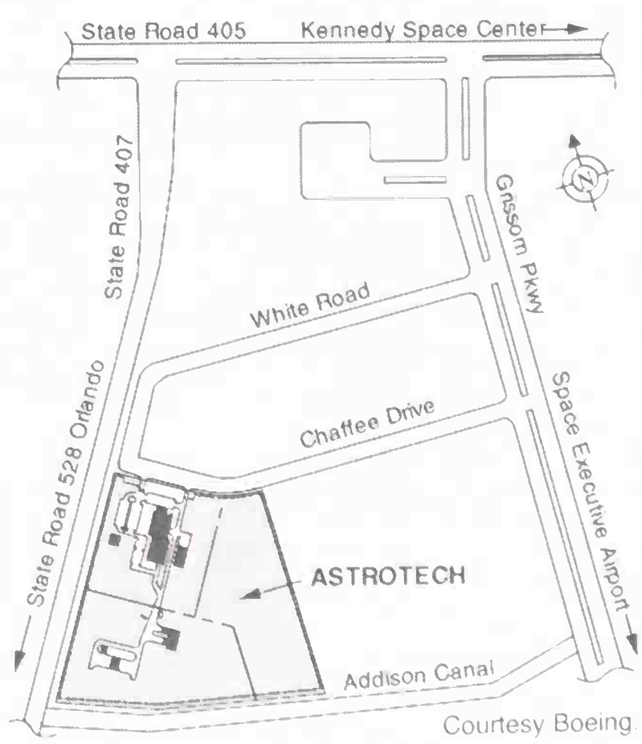
CCAFS is located on the Atlantic coast of Florida approximately midway between Jacksonville and Miami, 87 km (54 mi) east of Orlando, Florida, United States. KSC is 55 km (34 mi) long from north to south, and 16 km (10 mi) across at its widest point. The nearest communities are Cocoa Beach, located between Patrick AFB and Merritt Island, 40.5 km (25 mi) to the south; Titusville 14 km (9 mi) to the north; and Melbourne 66 km (41 mi) to the south. Measured distances are to KSC main entrance, and not CCAFS. Approximately 465,000 people live in the nearby communities of Brevard County. Communication, medical care, transportation services, lodging, food, outdoor recreation, museums, and other amenities are all readily available. During the week, overnight package delivery is available from points within the continental United States.

The climate is warm, but subject to thunderstorms and occasional hurricanes between the months of June and November. The closest commercial airport is in Melbourne, Florida. Commuter flights are available from Atlanta to the Melbourne International Airport (MLB). Connections worldwide are available through the Orlando International Airport (MCO). Rental cars are available at all of the airports. Cargo can be delivered by air into Orlando, or directly to the Cape Canaveral skid strip airfield. Ocean-going or intracoastal vessels can dock at Port Canaveral, and rail transportation to the site is also available. Highways in the area, and between Cape Canaveral and Orlando, are paved, usually multilaned, and well maintained. Many facilities are commercially operated, including the Astrotech payload processing facilities. The Florida Space Authority, a state-sponsored organization, operates Space Launch Complex 46 (SLC-46), which is available on a commercial basis to a variety of small launch vehicles. SLC-20 is also being refurbished for the same purpose.



Courtesy Boeing

Astrotech Site Location



Courtesy Boeing

Astrotech Complex Location

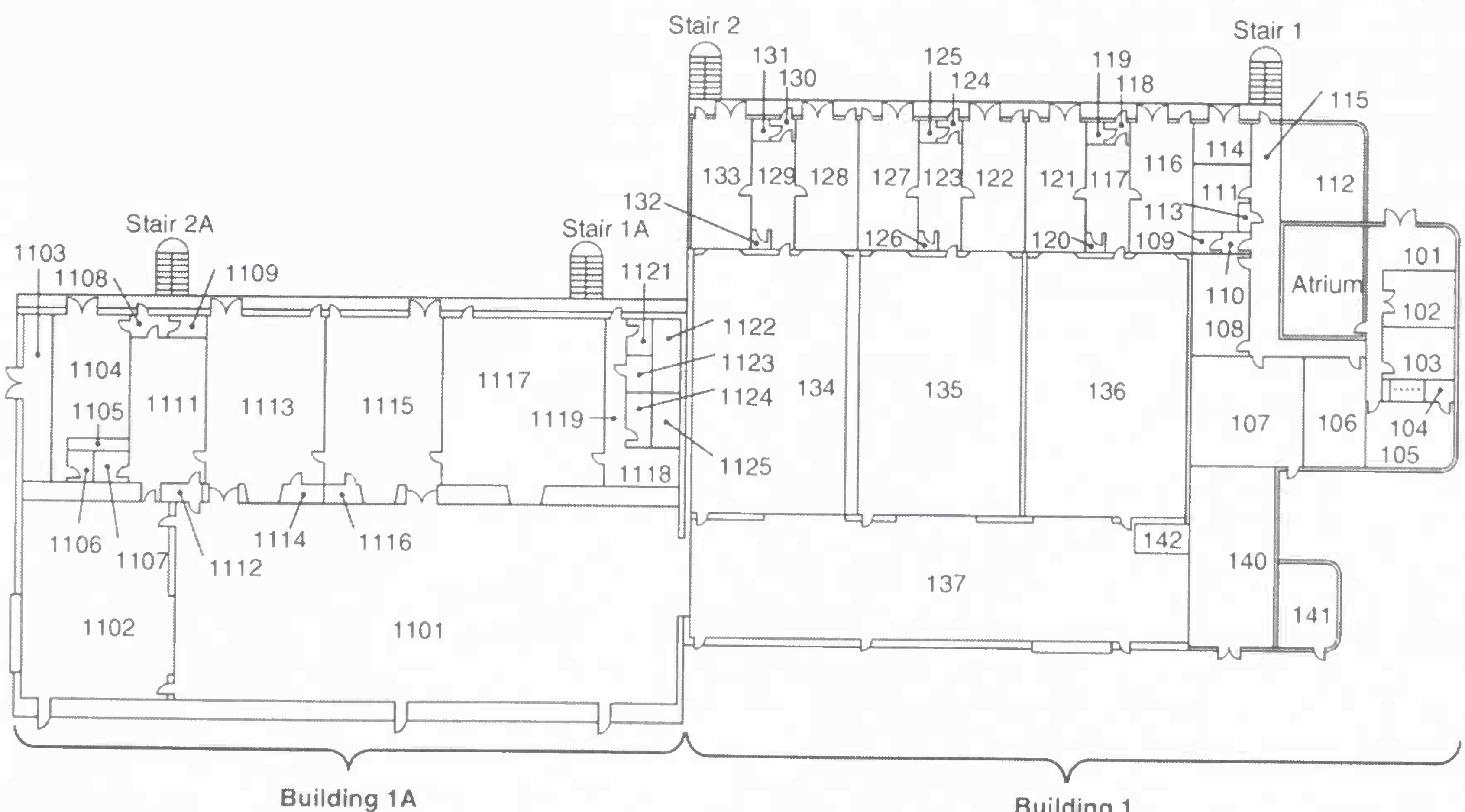
Cape Canaveral Air Force Station/NASA Kennedy Space Center

Launch Complexes

CCAFS and KSC contain approximately 40 launch complexes. (The pad numbers go up to 47, but some numbers were assigned to pads that were never built or were reassigned as duplicates to refurbished pads.) Only LC-39 is part of KSC, all other complexes are part of CCAFS. Most of the launch complexes were built in the 1950s and 1960s and have since been torn down, abandoned, or reassigned to other uses. For example, Pad 26 is now the home of the U.S. Air Force Space Museum. Only the pads currently used for space launches are listed. In the late 1990s, CCAFS began using the SLC prefix instead of the older LC designation. Both terms are still in use.

Launch Complex	Vehicles Supported	Comments
SLC-17 A and B	Delta II and III	This is the launch complex for Delta II (both pads) and Delta III (Pad B only). It consists of one shared block house and two pads, each with a fixed umbilical tower and mobile service tower.
SLC-20	Miscellaneous	The former Titan I and II pad is now operated by Spaceport Florida Authority for suborbital and small orbital launch vehicles.
SLC-36 A and B	Atlas II and III	This is the launch complex for Atlas II (both pads) and III (Pad B only). It consists of two pads, each with a fixed umbilical tower and mobile service tower. Operations are scheduled to cease in 2005.
SLC-37	Delta IV	The former Saturn IB complex was rebuilt as the pad for Delta IV.
LC-39 A and B	Space Shuttle	The launch complex for the Space Shuttle, previously used for Saturn V/Apollo launches. Vehicles are assembled in large VAB then transported to one of two launch pads.
SLC-40	Titan IVB	SLC-40 and 41 are part of a combined launch area, but only SLC-40 is still used for Titan launches. Vehicles are assembled vertically in the VIB and SMARF buildings, then transported by rail on a mobile launcher to the launch complex.
SLC-41	Atlas V	The former Titan pad was demolished and rebuilt with a “clean pad” design for the new Atlas V.
SLC-46	Athena I and II	A former U.S. Navy pad, it is now operated by Spaceport Florida Authority as commercial launch pad for small solid rockets.

Payload Processing Facilities



Building 1A

1101 Large High Bay D
1102 Large Airlock
1103 Mechanical Room
1104 Soundproof Conference Room D1
1105 Closet
1106 Restroom
1107 Restroom
1108 Vestibule
1109 Janitor Storage
1110 Not Used
1111 Change Room D
1112 Air Shower
1113 Control Room D2
1114 Equipment Room
1115 Control Room D1
1116 Equipment Room
1117 Office Area D1
1118 Break Room
1119 Corridor
1120 Not Used
1121 Men's Washroom
1122 Men's Restroom
1123 Janitor Closet
1124 Women's Washroom
1125 Women's Restroom

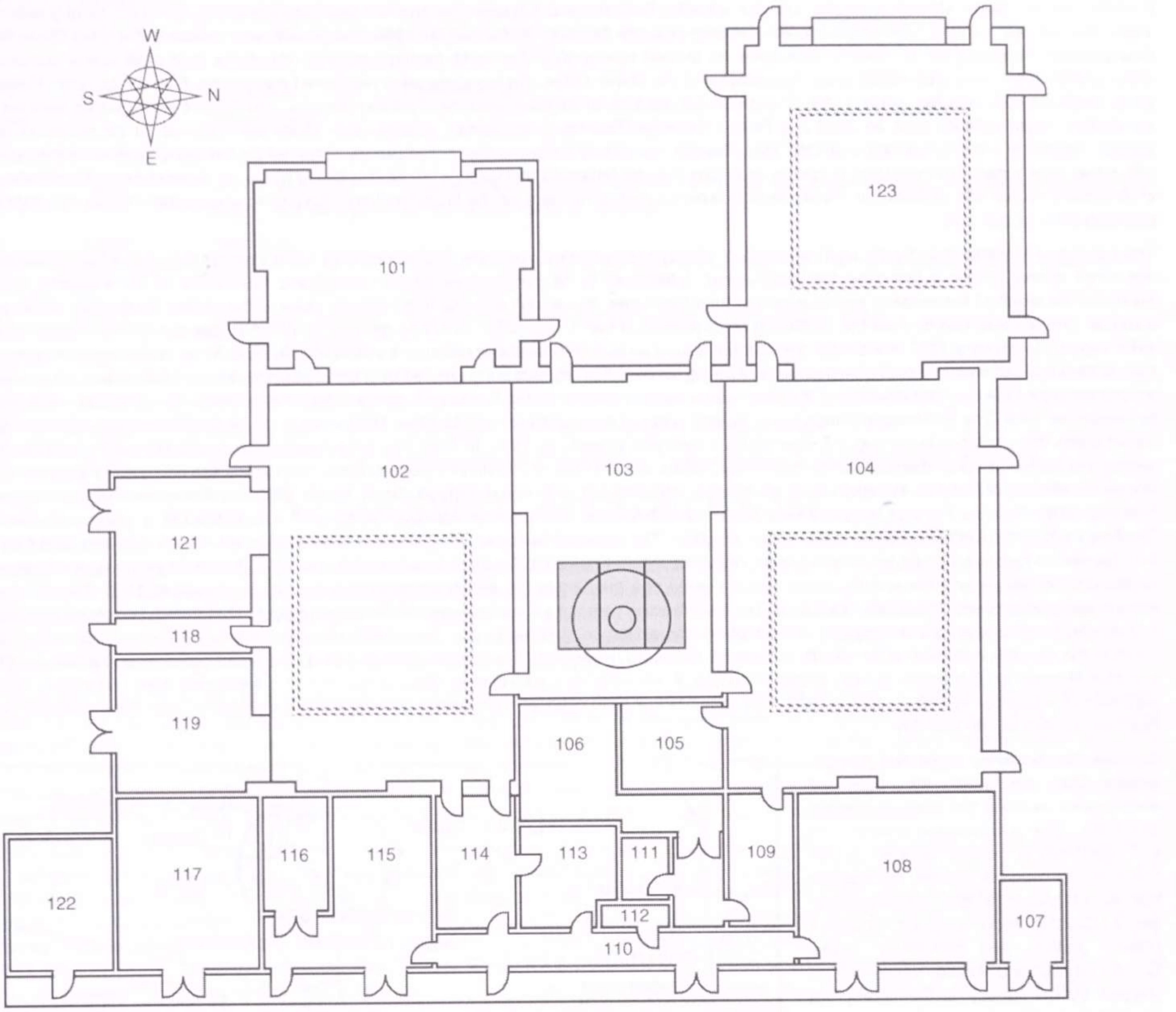
Building 1

101 ASO Reception Area
102 ASO Repro/Fax
103 ASO Staff Office
104 ASO Office Restroom
105 ASO Staff Office
106 ASO Staff Office
107 ASO Staff Office
108 Conference Room
109 Women's Restroom
110 Women's Lounge
111 Men's Restroom
112 Break/Lunch Room
113 Janitor Closet
114 ASO Machine Shop
115 Corridor
116 Control Room A1
117 Change Room A
118 Vestibule A
119 Storage A
120 Restroom A
121 Control Room A2
122 Control Room B1
123 Change Room B
124 Vestibule B
125 Storage B
126 Restroom B
127 Control Room B2
128 Control Room C1
129 Change Room C
130 Vestibule C
131 Storage C
132 Restroom C
133 Control Room C2
134 High Bay C
135 High Bay B
136 High Bay A
137 Common Airlock
138 Not Used
139 Not Used
140 Mechanical Room
141 Electrical Vault
142 Telephone Room

First Level Floor Plan, Building 1/1A, Astrotech

Courtesy Boeing

CAPE CANAVERAL AIR FORCE STATION/NASA KENNEDY SPACE CENTER



Courtesy Boeing.

101 South Airlock
102 South High Bay
103 Center High Bay
104 North High Bay
105 Office
106 Mechanical Room 2
107 Motor Generator Room
108 North Control Room
109 North Change Room
110 Corridor
111 Women's Restroom
112 Janitor
113 Men's Restroom
114 South Change Room
115 South Control Room
116 Balance Machine Control Room
117 Mechanical Room 1
118 Corridor
119 Oxidizer Cart Storage Room
120 Not Used
121 Fuel Cart Storage Room
122 Electrical Vault
123 Building 2A – North Airlock High Bay

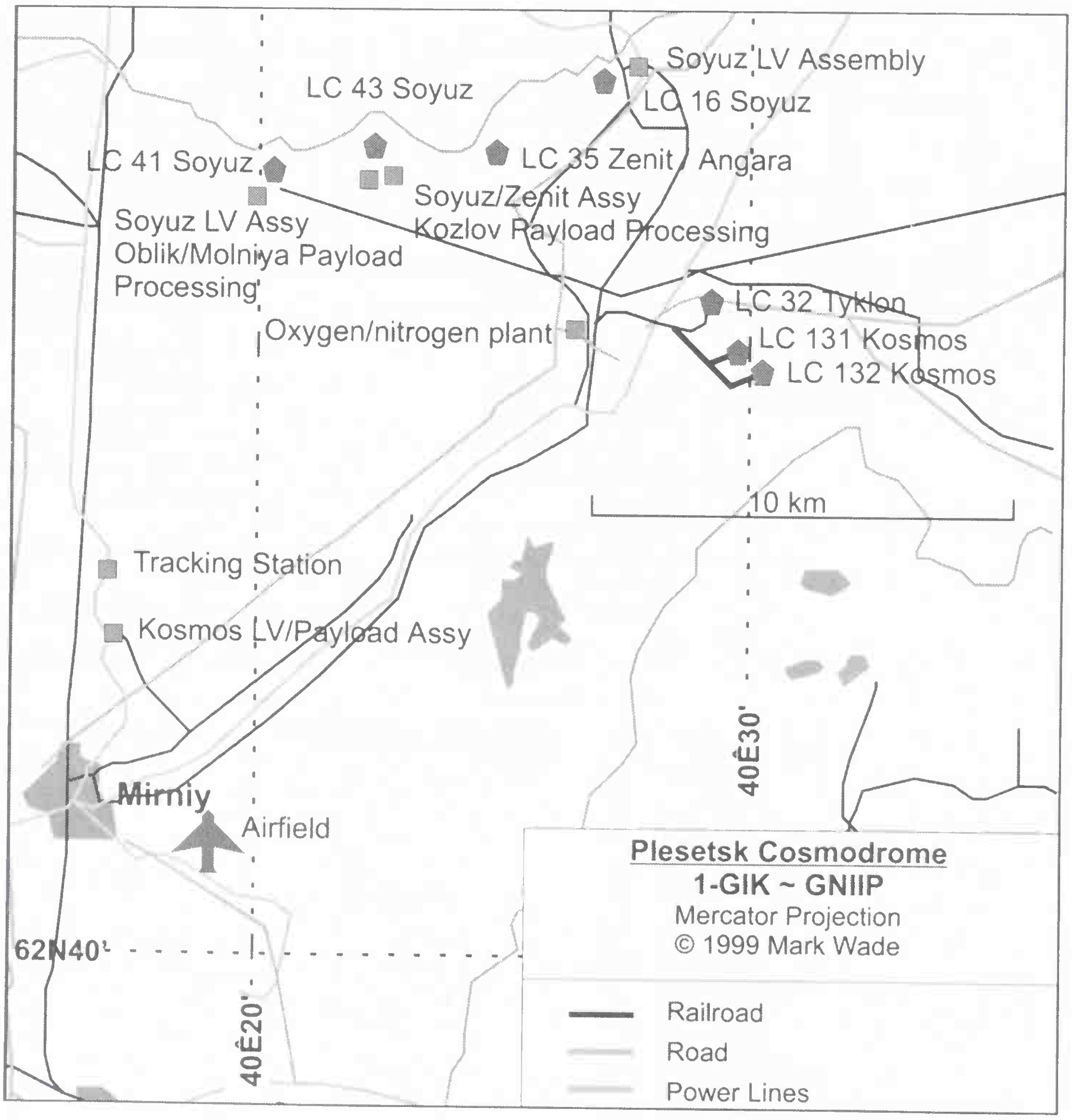
Building 2 Detailed Floor Plan, Astrotech

Plesetsk

Plesetsk was the Soviet Union's third cosmodrome following Baikonur and Kapustin Yar and has been used primarily to launch military spacecraft. In 1963, the site was named NIIP-53 (Scientific Research Facility No. 53). In November 1994, the facility was renamed the First State Research Cosmodrome Plesetsk (GIK-1). Plesetsk was once the busiest spaceport in the world, having supported about one third of all space launches worldwide. Launch rates have plummeted since the collapse of the Soviet Union. No launches were performed there in the first half of 1999. Plesetsk supports smaller launch vehicles, primarily the Soyuz/Molniya medium-lift vehicles, and the Cyclone, Kosmos, and Rockot small launch vehicles. It does not support larger vehicles such as Zenit and Proton. Because Plesetsk is landlocked, specific drop zones are reserved for the impact of jettisoned stages. Therefore, orbit inclinations available from Plesetsk are strictly limited to those that can be achieved on the launch azimuths that are aligned with these drop zones. The reduction in military launches may be balanced by Russia's increasing desire not to be dependent on Kazakhstan for use of its primary launch site at Baikonur. Future launch vehicles such as Angara will be launched from Plesetsk to ensure that Russia can deploy heavy payloads from its own soil.

The location of Plesetsk is by far the most northerly of all major launch sites—only the Shtil launch area in the Barents Sea is at a higher latitude. While this is not optimal for space launches, the location was determined by its early role as the first operational ICBM base for R-7A ballistic missiles. To maximize the reach of the missiles and to minimize their flight time, the launch site had to be built as close to the United States (i.e., as far north) as possible. The site also had to meet the conflicting requirements of being accessible by rail for delivery of rocket stages but remote enough to maintain strict secrecy. In January 1957, the Soviet government issued a resolution for the creation of a secret missile base to be code named Angara near the Plesetskaya railroad station, south of Arkhangelsk. Arriving construction workers were greeted by a fierce blizzard, and by spring many areas were flooded and swampy. However, construction of the base was a national priority so the first launch complex for R-7A missiles was completed and operational by December 1959. The first missile launch from Plesetsk was not conducted until 22 October 1963 when an R-16 ICBM was fired, although Plesetsk-based crews had practiced launching R-7 missiles from Baikonur as early as 1959. In 1963 The Soviet leadership decided to build a new site for ICBM testing and high-inclination space launches near Velsk, about 300 km (186 mi) south of Plesetsk. Excavation at the site had started when Korolev convinced the Ministry of Defense to make use of the existing infrastructure at Plesetsk instead. On 17 March 1966, the first spacecraft was launched from Plesetsk, when Kosmos 112 was launched by a Vostok launch vehicle. The unusual inclination of the orbit was noticed by a group of students led by Geoffrey Perry at the Kettering School in the United Kingdom. The students had been using amateur radio equipment to track satellites, and they noticed that the orbit of Kosmos 112 did not originate from Baikonur. When Kosmos 129 was launched in October 1966, the students were able to pinpoint the location of the new launch site and disclosed it to the world at a meeting of the British Interplanetary Society in November 1966. Perhaps the timing was only coincidental with this public disclosure, but the following year the primary function of Plesetsk began to shift from an operational missile base to a space launch and missile testing site. Nevertheless, the existence of Plesetsk was not publicly disclosed within the Soviet Union until 1983. As a result of this secrecy, a number of accidents at Plesetsk remained unknown to the outside world for years. The worst occurred on 18 March 1980 when a Vostok M exploded 2 h before launch, killing 51 people. A memorial now stands near Mirny, where the 51 were buried, next to 9 people killed in the explosion of a Kosmos launch vehicle in 1973. As a result of the 1980 disaster, 18 March is a memorial day at Plesetsk, and since that time no launch has been carried out on that day.

Plesetsk Cosmodrome is located approximately 200 km (125 mi) south of Arkhangelsk, north of the cities of Plesetsk and Mirny. The cosmodrome covers 1750 km^2 (660 mi^2) of heavily wooded terrain. The complex includes booster and payload test and integration facilities, tracking antennas, a LOX/LN_2 production plant, an airport, railway station, and residential areas. Launch vehicles and Russian spacecraft are shipped by rail directly to the appropriate integration facility. Commercial spacecraft can be shipped by rail or aircraft. As the 50×2000 m (160×6550 ft) runway at Plesetsk cannot accommodate large aircraft heavier than 50 t (110 klbm), payloads shipped by air must be transferred to smaller aircraft (such as the An-12) in Moscow or Arkhangelsk. The runway is being upgraded so that by late 2003 it will be able to handle larger aircraft. Temperatures range from a low of −38° C (−36° F) in the winter to a high of 33° C (91° F) in the summer. Plesetsk has supported only a small number of commercial launches for foreign customers, so most amenities are still spartan by Western standards. Upgrades to these facilities are likely for upcoming commercial launches on Rockot and Angara. Accommodations are provided at a hotel in Mirny with international television and telephone services. Medical care is provided at a well-equipped military-run hospital, and medical evacuation to hospitals in Western Europe can be performed if necessary.



Courtesy Mark Wade.

PLESETSK

Launch Complexes

Only pads used for space launches are shown below. A number of missile pads or silos are also present at Plesetsk

Launch Complex	Pad	Vehicles Supported	Comments
LC 16	Pad 2	Vostok, Voskhod, Soyuz, and Molniya	Originally built as an operational R-7 missile pad, it was switched to a space launch complex for the Soyuz family of vehicles.
LC 32	2 pads	Cyclone 3	
LC 35	Pad 1 and 2	Angara	This was built for Zenit, but never completed. Pad 1 will be refurbished and modified for Angara flights.
LC 41	Pad 1	Vostok and Voskhod, Soyuz and Molniya	The first operational pad at Plesetsk, it was retired in 1989.
LC 43	Pad 3 and 4	Vostok, Voskhod, Soyuz, Molniya	
LC 132	2 pads	Kosmos 3M	This launch site is still in use for Kosmos launches.
LC 133	1 pad	Rockot	This is a former Kosmos pad converted to launch commercial Rockot vehicles.

VANDENBERG AIR FORCE BASE

Vandenberg Air Force Base (VAFB) is one of the two major U.S. space launch facilities. VAFB is located at Point Arguello, California, 240 km (150 mi) northwest of Los Angeles. VAFB was established as a facility for missile test launches as well as launches of military satellites to polar orbits. Throughout its history it has served as the site for most U.S. launches into polar orbit for military, civil, and in recent years, commercial customers. VAFB was the launch site of the world's first satellite launched into polar orbit. The location of VAFB at 34.7°N, 120.6°W permits access to polar and other high-inclination orbits launching to the south and southwest without overflying land.

Before World War II the site consisted of several Mexican land grant ranches. In May 1941 the land was acquired by the U.S. government and converted into a U.S. Army base named Camp Cooke in honor of Maj. Gen. Philip St. George Cooke. On 4 October 1958, Cooke AFB was renamed Vandenberg AFB in honor of the late Gen. Hoyt S. Vandenberg, the Air Force's second chief of staff. On 8 May 1957, ground was broken to begin construction of the space launch and missile test facility. The first missile launched from VAFB was a Thor IRBM on 16 December 1958. Two months later, on 28 February 1959, the world's first polar orbiting satellite, Discoverer I, was lifted into space aboard a Thor/Agena. Under the cover story of scientific research, Discoverer was actually the cover name for Corona, the first U.S. photo reconnaissance satellite program.

The climate year round is temperate, the base receives moderate rainfall, little snow, but daily fog. Summer temperatures can rise as high as 38°C (100°F), but more typically only to 24°C (75°F). Winter lows average 3°C (38°F), but have gone as low as −7°C (20°F). With 56 km (35 miles) of coastline, VAFB's position at the end of one of the only major stretches of coastline running east to west in California allows launches to the south without overflying land. The spaceport area totals 400 km^2 (150 mi^2). Each vehicle type has its own processing and launch facilities. Space launch complexes are primarily located on South Base, where launch vehicles can fly due south, or even southeast to reach Molniya-type orbits at 63.4 deg inclination. From the Delta and Taurus complexes on North Base, launch vehicles must fly to the southwest to avoid overflight of the South Base facilities, then perform a dogleg turn to a more southerly flight azimuth resulting in a modest performance reduction. Missile test sites are located on the far northern side of the base. These launches typically head west toward the Kwajalein Missile Range in the Pacific.

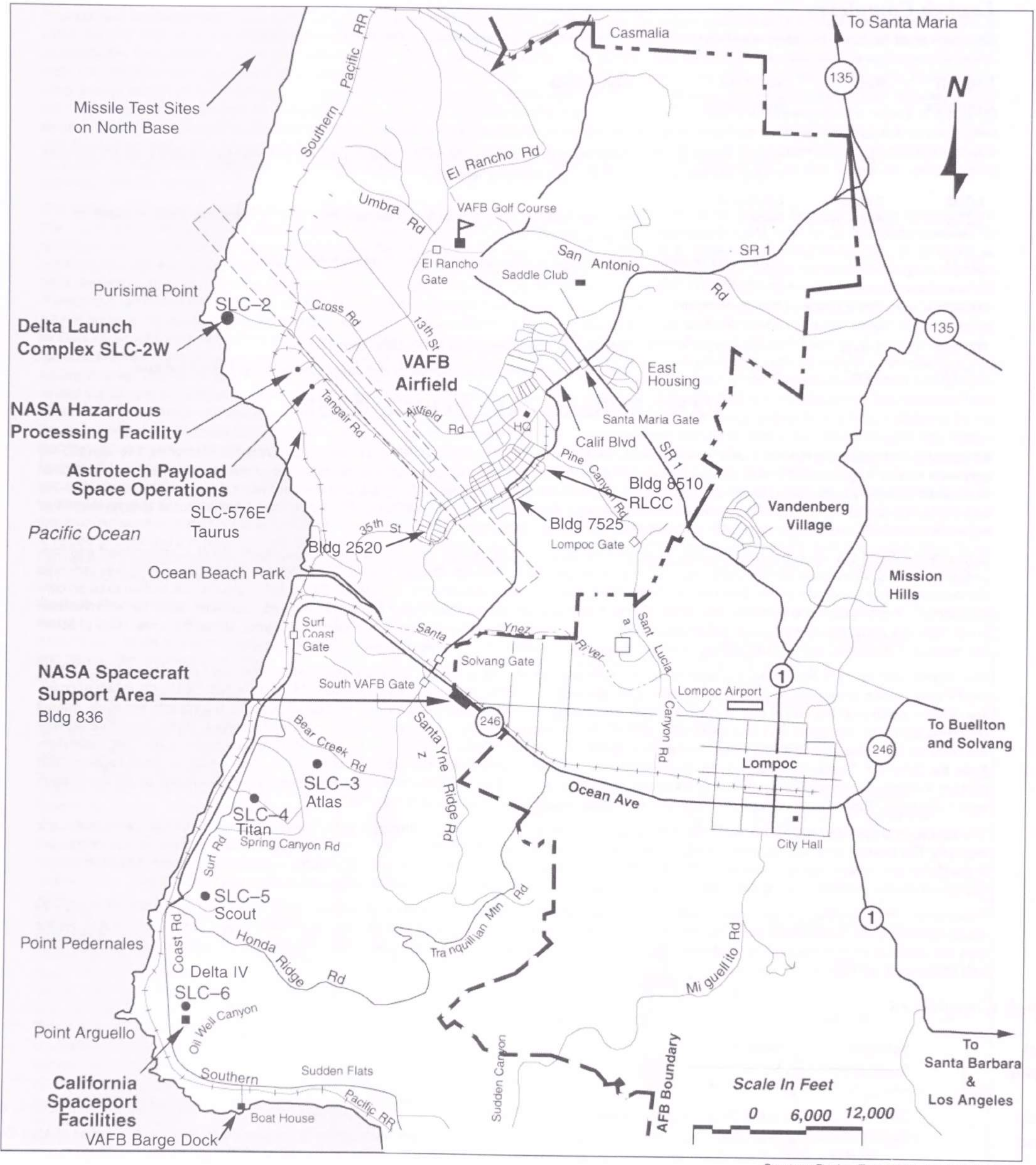
The spaceport is operated by the U.S. Air Force 30th Space Wing. Military, civilian, and commercial space launches as well as missile test flights, occur regularly. The nearest communities are Lompoc, 11 km (7 mi) southeast, and Santa Maria, 27 km (17 mi) northeast. Approximately 130,000 people live in the towns and valleys nearby. Communication, medical care, transportation services, lodging, food, outdoor recreation, museums, and other amenities are all readily available. During the week, overnight package delivery is available from points within the continental United States.

The closest commercial airports are in Santa Maria and Santa Barbara, California. Commuter flights are available from Los Angeles, San Francisco, or Santa Barbara to the Santa Maria Municipal Airport. Connections worldwide are available through the Los Angeles International Airport (LAX). Rental cars are available at all of the airports. Highways in the area and between VAFB and Los Angeles or San Francisco are paved, usually multilaned, and well maintained, although fog poses hazards.

Launch Complexes

Launch Complex	Vehicles Supported	Comments
SLC-2W	Delta II	This is the Delta II launch complex consisting of fixed umbilical tower and mobile service tower. SLC-2 East was torn down in 19
SLC-3E	Atlas	After many years launching converted Atlas ICBMs, this pad was redeveloped in the late 1990s to support launches of Atlas-Cen on the West Coast. Additional modifications will enable Atlas V launches by the end of 2005.
SLC-3W	Falcon	The former Atlas and Thor launch tower was demolished in 2000. In 2002 the site was selected for the new Falcon small launch vehicle.
SLC-4E and W	Titan	SLC-4 East supports Titan IV, while SLC-4 West is used for Titan II.
SLC-5	Scout	This closed pad was used for launches of the small Scout launch vehicle.
SLC-6	Delta IV	Originally built in the 1960s for manned Titans and later modified at great expense for military Space Shuttle launches, it has never used for either. Four launches of the small Athena were conducted here before Boeing adopted the complex for Delta IV.
SLC-7	Miscellaneous	This is a planned commercial pad for small solid launch vehicles.
SLC-576E	Taurus	This simple facility is used for Taurus launches.
California Spaceport	Minotaur	The commercially operated facility has a flat pad and flame duct and lacks a tower.

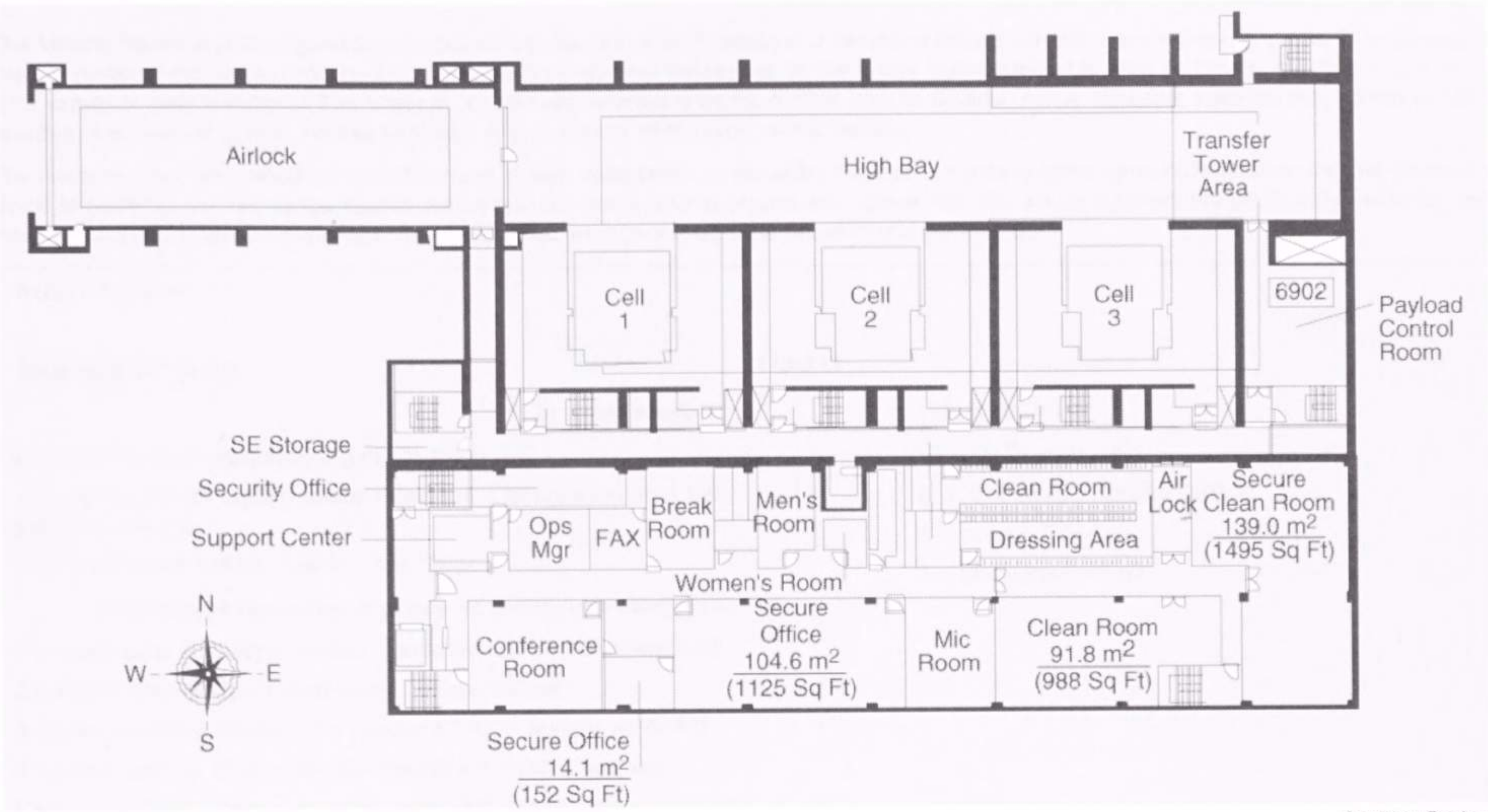
Vandenberg Air Force Base



Courtesy Boeing Expendable Launch Systems.

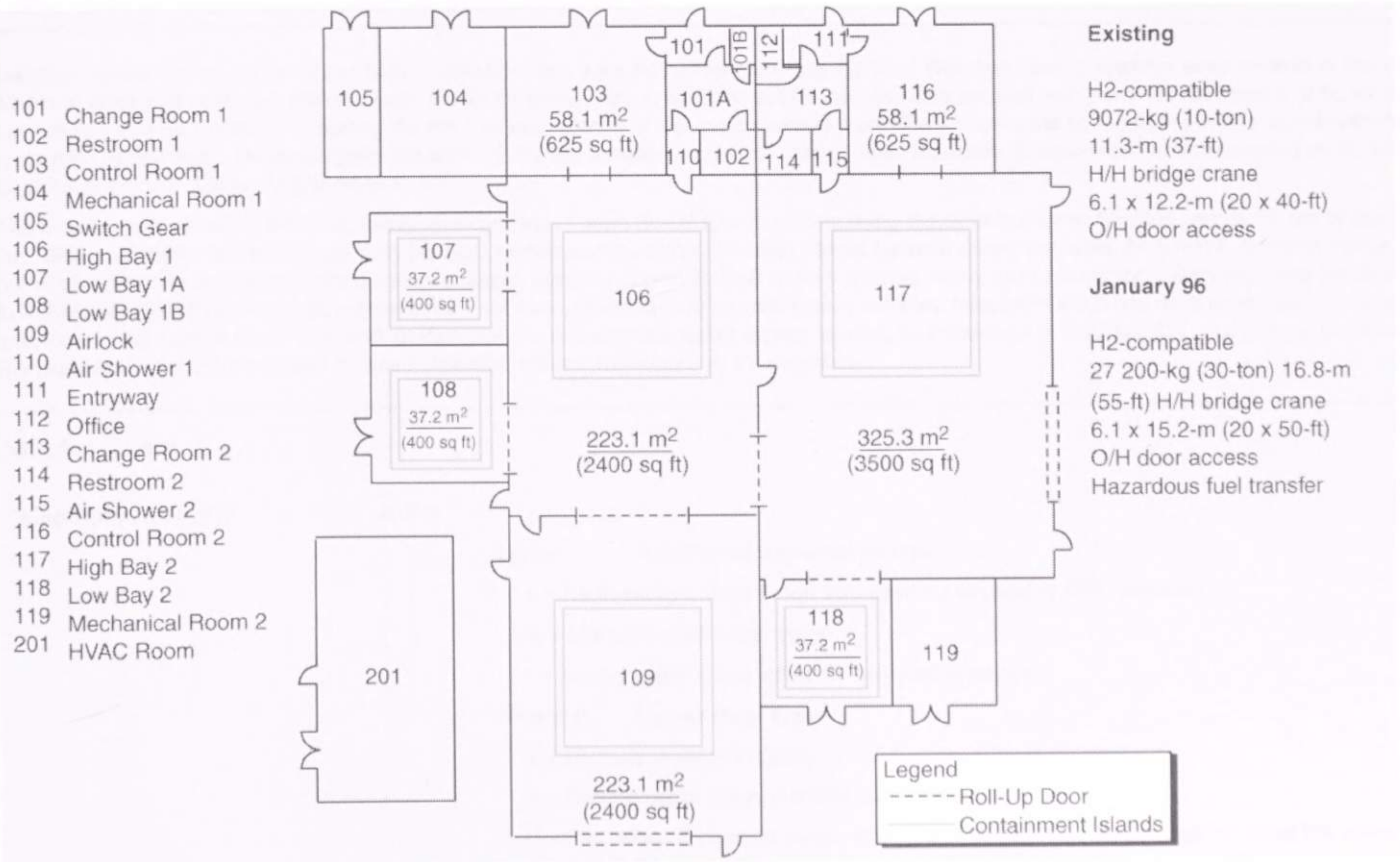
Vandenberg Air Force Base

Payload Processing Facilities



Courtesy Boeing.

California Spaceport–Plan View of the IPF



Astrotech Payload Processing Facility, Building 1032

Courtesy Boeing.

APPENDIX

NOMENCLATURE OF RUSSIAN AND UKRAINIAN LAUNCH VEHICLES

The Vehicle Family and Configuration columns reflect the names for Russian and Ukrainian launch vehicles used within this guide. They generally follow convention, with one exception—Soyuz. There is no general consensus on the family name referred to here as Soyuz, resulting in part from the wide variety of vehicle variants. The family is occasionally referred to as the A class from its Sheldon name. However, because that reflects a U.S. perspective, it is referred to here instead as Soyuz after the name of the most prolific variant.

The article number, also called the model number, design designation, or the index number, is a unique alphanumeric designator assigned by the Soviet Union to each type of long-range missile, space launch vehicle, rocket engine, and spacecraft. The article numbers are particularly useful for making more precise distinctions between launch vehicles than are possible with the Sheldon and DoD names.

Article Number

Example (Soyuz): **11A511**

System category

8 = Land based ballistic missile (prior to 1968)

11 = Space launch system (many retain the 8 designation from the precursor missile)

15 = Land based ballistic missile (after 1968)

System type (meaning depends on the system category)

A (missile, prior to 1968) = Ballistic missile with conventional warhead

A (missile, after 1968) = Liquid-fueled ballistic missile

A (space launch) = Designed by Korolev's OKB-1 bureau, or TsSKB

K (missile, prior to 1968) = Ballistic missile with nuclear warhead

K (space launch) = Designed by Yangel's OKB-586 bureau

Zh (missile, after 1968) = Solid-propellant ballistic missile

D = Rocket engine

S = Stage

F = Payload (warhead for missiles or spacecraft for launch vehicles)

Other codes for various types of spacecraft, ground equipment, etc.

Project number

Design Bureau code

5, 7, 9 = Usually Korolev OKB-1

6 = Yangel OKB-586

8 = Chelomei OKB-52

Because official Soviet-era names of launch vehicles often were not released during the Cold War, two naming systems were created in the United States to distinguish between vehicle types. The DoD names, also referred to as U.S. names, were created first. These consist simply of SL for Space Launch followed by a number indicating the order of appearance of the launch vehicle type. The numbers are not assigned until a launch vehicle has performed its first flight. These designations were not initially available to civilian researchers. Therefore, Sheldon names were created by Dr. Charles Sheldon of the U.S. Library of Congress in 1968.

The Sheldon names reflect the basic family and variation of each Soviet launch vehicle using the system shown. Sheldon names fell out of use in the late 1980s and early 1990s because both the DoD names and the official Russian names became widely available. As a result, Sheldon names have not been applied to several recently developed launch systems. Similarly, DoD names are now rarely used except for military reporting (for example, in NORAD documentation) and SL numbers have not been assigned to some new launch vehicles. Instead NORAD has used either NATO missile designations, in the case of Dnepr and Shtil, or designations of a previous space launch version, as in the case of the Zenit 3SL. For general purposes the Russian names listed here should be used. Western names are listed only for reference.

Sheldon Names

Example (Soyuz): **A-2-e**

Letter **Additional upper-stage type**

e = Earth-escape stage (often also used for elliptical or GEO missions)

m = Maneuverable upper stage

r = reentry upper stage (often actually part of payload)

Numeral **Upper-stage type**

1 = First upper stage in family

2 = Second upper stage in family (etc.)

Number refers to different stages in different families, i.e., upper stage "1" is not the same for A-1 and D-1.

First letter **Family (E, H, and I not used)**

Nomenclature of Russian and Ukrainian Launch Vehicles

Vehicle Family	Configuration	Article Number	Sheldon Name	DoD Name	Predecessor Missile (NATO Designation)
Dnepr	Dnepr-1	15A18M	None	SL-24	R-36M2/RS-20 15A18M (SS-18 Mod 4 Satan)
Energia	Energia	11K25	K-1	SL-17	New design
Kosmos	Kosmos	63S1	B-1	SL-7	R-12 8K63 (SS-4 Sandal)
	Kosmos	11K63	B-1	SL-7	
Kosmos	Kosmos 2	65S3	C-1	SL-8	R-14 8K65 (SS-5 Skean)
	Kosmos 3	11K65	C-1	SL-8	
	Kosmos 3M	11K65M	C-1	SL-8	
	?	K65M-RB	C-1	SL-8	
N-1	N-1	11A52	G-1-e	SL-15	New design
Proton	Proton	8K82	D	SL-9	UR-500 (not deployed as ICBM)
	Proton K	8K82K	D-1	SL-12	
	Proton K/Block D	8K82K/Block D	D-1-e	SL-13	
	Proton K/Block DM	8K82K/Block DM	D-1-e	SL-13	
	Proton M/Breeze M	8K82KM	None	None to date	
Rockot	Rockot/Breeze K	?	None	SL-19	RS-18 15A30, 15A35 (SS-19 Stilleto)
	Rockot/Breeze KM	?	None	None to date	
Shtil	Shtil-1	?	None	SS-N-23	R-29RM RSM-54 (SS-N-23)
	Shtil-1N	?	None	None to date	
Soyuz	Sputnik	8K71O or 8K71PS	A	SL-1	R-7 8K71, R-7A 8K74 Semyorka (SS-6 Sapwood)
	Sputnik	8A91	A	SL-2	
	Vostok, Luna	8K72	A-1	SL-3	
	Vostok K	8K72K	A-1	SL-3	
	Vostok-2	8A92	A-1	SL-3	
	Vostok-2M	8A92M	A-1	SL-3	
	Voskhod	11A59	A-1-m	SL-5	
	Molniya	8K78	A-2-e	SL-6	
	Molniya-M	8K78M	A-2-e	SL-6	
	Voskhod	11A57	A-2	SL-4	
	Soyuz	11A511	A-2	SL-4	
	Soyuz	11A510	A-2?	SL-10	
	Soyuz L	11A511L	A-2	SL-4	
	Soyuz M	11A511M	A-2	SL-4	
	Soyuz U	11A511U	A-2	SL-4	
	Soyuz U2	11A511U2	A-2	SL-4	
	Soyuz U/Ikar	11A511U?	None	SL-4	
	Soyuz U/Fregat	11A511U?	None	None to date	
	Soyuz FG	11A511U-FG	None	None to date	
	Soyuz ST/Fregat	None to date	None	None to date	
Start	Start-1	?	L-1	SL-18	RT-2PM Topol 15Zh58 (SS-25 Sickle), RSD-10M Pioneer 11Zh45, 11Zh53 (SS-20 Saber)
	Start	?	None	None to date	
Strela	Strela	15A30 ?	None	None to date	RS-18 15A30, 15A35 (SS-19 Stilleto)
Tsiklon	R-36-O	8K69orb	F-1-r	SL-11	R-36 8K69 (SS-9 Scarp)
	Tsiklon 2A	11K69	F-1	SL-11	
	Tsiklon 2	11K69	F-1-m	SL-11	
	Tsiklon 3	11K68	F-2	SL-14	
Zenit	Zenit-2	11K77	J-1	SL-16	New design
	Zenit-3SL (Sea Launch)	?	None	SL-16	

Abbreviations and Acronyms

-A-

AADC	=	Alaska Aerospace Development Corporation
AAM	=	AVUM avionics module
ABRES	=	advanced ballistic reentry system
ACLV	=	Ariane complimentary launch vehicle
ACS	=	attitude control system; assembly and command ship
ADPM	=	active dispenser payload module
AEB	=	Agencia Espacial Brasileira (Brazilian Space Agency)
AFRL	=	U.S. Air Force Research Laboratory
AFS	=	Air Force Station
AFSLV	=	U.S. Air Force Small Launch Vehicle
AGE	=	aerospace ground equipment
AIT	=	assembly and integration trailer
AKO	=	Aviatsionno-Kosmicheskoye Ob'edinenie (Aviation-Space Organization)
ALB	=	Air Launched Vehicle
ALD	=	Ariane light derivatives
AP	=	ammonium perchlorate
APC	=	Aft Payload Capsule
APM	=	apogee propulsion module; AVUM propulsion module
APSC	=	Asia Pacific Space Centre
ARC	=	Atlantic Research Corporation
ARCM	=	Atlas roll control module
AS	=	autostabilizer
ASAP	=	Ariane structure for auxiliary payloads
ASE	=	airborne support equipment
ASLV	=	augmented satellite launch vehicle
ASOC	=	Atlas V spaceflight operations center
ASRM	=	advanced SRM
ATV	=	automated transfer vehicle
ATB	=	Assembly and Test Building, Plesetsk, Russia
AVUM	=	attitude invernier upper module

-B-

BAF	=	batiment d'assemblage final (Ariane final assembly building)
BIL	=	batiment d'integration lanceur (Ariane booster integration building)
BIP	=	boosting integration building
BIV	=	Batiment dí intégration Vega
BOR	=	Bespilotnye Orbitalnye Raketoplany (unmanned orbital rocketplane)

-C-

CALT	=	China Academy of Launch Vehicle Technology
CASA	=	Construcciones Aeronáuticas S.A.
CASIC		China Aerospace Sciences & Industry Corporation
CBC	=	common booster core
CBM	=	common birthing mechanism
CCAFS	=	Cape Canaveral Air Force Station
CCAM	=	collision/contamination avoidance maneuver
CCB	=	common core booster
CCAS	=	Cape Canaveral Air Station
CDL-3	=	centre de lancement (Ariane launch control center)
CDR	=	command destruct receiver
CELV	=	complementary expendable launch vehicle
CFRP	=	carbon–fiber reinforced plastic
CGWIC	=	China Great Wall Industrial Corporation
CIV	=	civilian
CLA	=	Centro de Lançamento de Alcântara (Alcantara Launch Center, Brazil)
CLF	=	Commercial Launch Facility
CLTC	=	China Satellite Launch and Tracking Control
CML	=	commercial
CNES	=	Centre National d'Etudes Spatiale (National Center for Space Studies, France)
COBAE	=	Comissão Brasileira de Atividades Espaciais (Brazilian Space Activities Commission)
COMET	=	Commercial Experiment Transporter
COMSTAC	=	Commercial Space Transportation Advisory Committee (FAA)
COSPAR	=	Committee on Space Research
CPKM	=	Long March 2C perigee kick motor
CRT	=	Checkout Terminal Room
CSD	=	Chemical Systems Division of Pratt & Whitney
CSG	=	Centre Spatial Guyanais (Guiana Space Center, French Guiana)
CST	=	combined system test
CTA	=	Centro Técnico Aeroespacial (Aerospace Technical Center, Brazil)
CTS	=	LM-2C Top Stage
CUS	=	Cryogenic Upper Stage
CY	=	calendar year
CYSA	=	Cape York Space Agency

-D-

DARPA	=	U.S. Defense Advanced Research Projects Agency
DAS	=	data acquisition system
DASA	=	Daimler-Benz Aerospace
DEC	=	dual-engine Centaur
DIGS	=	Delta inertial guidance system
DIV-H	=	Delta IV Heavy
DIV-M	=	Delta IV Medium
DLA	=	dual launch adapter
DLV	=	demonstration launch vehicle
DMCO	=	Delta mission checkout
DMS	=	data management subsystem
DoD	=	Department of Defense (USA)
DPAF	=	dual payload attachment fitting
DRSN	=	downrange station
DSCS	=	Defense Satellite Communications System
DSTF	=	Delta spin test facility
DTG	=	dry-tuned gyros

-E-

EADS	=	European Aeronautic Defence & Space Company
EAFB	=	Edwards AFB
EAP	=	etage d'appoint pourdre (Ariane 5 solid boosters)
ECB	=	energy center building
ECLSS	=	environmental control and life support system
ECS	=	environmental control system
EDO	=	extended duration orbiter
EEO	=	elliptical earth orbit
E/EEC	=	extendable and expandable exit cone
EEC	=	extendable exit cone
EELV	=	evolved expendable launch vehicle
ELA	=	ensemble de lancement Ariane (Ariane launch complex)
ELDO	=	European Launcher Development Organization
ELV	=	expendable launch vehicle
EPC	=	etage à propergol cryogenique (Ariane 5 core stage)
EPCU	=	payload preparation complex
EPDS	=	electrical power and distribution subsystem
EPF	=	extended payload fairing
EPKM	=	Long March 2E perigee kick motor
EPM	=	encapsulated payload module
EPS	=	etage propergol stockable (Ariane 5 upper stage)
ESA	=	European Space Agency
ESBM	=	equipment section boost motor
ESC	=	etage supérieur cryogenique (Ariane 5 cryogenic upper stages)
ESL	=	European small launcher
ESMDC	=	expandable shielded mild detonating cord
ESRO	=	European Space Research Organization
ESS	=	equipment support section
ET	=	external tank
ETS	=	E Top Stage
EVE	=	external vernier engine

-F-

FAA	=	U.S. Federal Aviation Administration
FAS	=	fin actuator system

Abbreviations and Acronyms

FCC = U.S. Federal Communications Commission
FCP = flight control processor
FCS = flight control subsystem
FDI = failure detection and isolation
FLSC = flexible linear shaped charges
FOBS = Fractional Orbital Bombardment System
FOG = fiber-optical gyro
FPR = flight performance reserve
FSA = Federal Space Agency
FSS = fixed service structure
FTS = flight termination system
FUT = fixed umbilical tower
FY = fiscal year

-G-

GAM = group d'activation moteur (Ariane 5 main engine gimbal system)
GAS = Get Away Special
GAT = group d'activation tuyères (Ariane 5 booster nozzle gimbal system)
GCC = guidance control computer
GCS = guidance command shutdown
GDS = gas dynamic shield
GEM = graphite–epoxy motors
GEO = geosynchronous earth orbit
GGC = ground control computer
GH_2 = gaseous hydrogen
GHe = gaseous helium
GIK = State Test Cosmodrome
GLONASS = Global navigation satellite system
GN_2 = gaseous nitrogen
GNS = guidance and navigation system
GPB = GPS position beacon
GPS = global positioning system
GOX = gaseous oxygen
GSE = ground support equipment
GSLV = Geosynchronous Satellite Launch Vehicle
GSO = geostationary earth orbit
GTO = geosynchronous transfer orbit

-H-

HAPS = hydrazine auxiliary propulsion system
HDF = hydrogen disposal facility
HGS = high-pressure gas storage
HIF = horizontal integration facility
HLV = heavy-lift vehicle
HPF = horizontal processing facility
HPFT = high-pressure fuel turbopump
HTPB = hydroxyl-terminated polybutadiene
HTV = H-IIA transfer vehicle

-I-

IAE = Instituto de Aeronáutica e Espaço (Institute of Aeronautics and Space, Brazil)
IAI = Israel Aircraft Industries
ICBM = intercontinental ballistic missile
IDIQ = indefinite delivery/indefinite quantity
IFR = in-flight retargeting
ILC = initial launch capability
ILS = International Launch Services
ILRV = integrated launch and reentry vehicles
IMI = Israel Military Industries
IMU = inertial measurement unit
INF = intermediate nuclear forces
INPE = Instituto Nacional de Pesquisas Espaciais National (Institute for Space Research, Brazil)
INSAT = Indian National Satellite
INU = inertial navigation unit
IP = interlocks package
IPF = integration and processing facility
IRBM = intermediate range ballistic missile
IRIS = integrated RCS (J-1 vehicle)
IRS = inertial reference system; Indian Remote Sensing satellites
ISAS = Institute of Space and Astronautical Science
ISC = International Space Company
ISDS = inadvertent separation destruct system
ISS = International Space Station
ISRO = Indian Space Research Organization
ISTRAC = ISRO telemetry, tracking, and telecommand network
ITIP = improved transtage injector program
ITL = integrate-transfer-launch
IUS = inertial upper stage

-J-

JAXA = Japan Aerospace Exploration Agency
JLRPG = Joint Long Range Proving Ground
JSC = NASA Johnson Space Center
JSLC = Jiuquan Satellite Launch Center

-K-

KB = Konstruktorskoye Byuro (Design Bureau)
KLC = Kodiak Launch Complex
KSC = NASA Kennedy Space Center; Kagoshima Space Center

-L-

LAP = Launch Assist Platform (K-1 first stage)
LAX = Los Angeles International Airport, Los Angeles, California, USA
LC = launch complex
LCC = launch control center
LDXL = large diameter extended length
LEO = low Earth orbit
LH_2 = liquid hydrogen
LITVC = liquid-injection thrust-vector control
LKE = Lockheed–Khrunichev–Energia
LLV = Lockheed Launch Vehicle; precursor to LMLV and Athena
LMLV = Lockheed Martin Launch Vehicle; precursor to Athena
LOX = liquid oxygen
LN2 = liquid nitrogen
LP = launch platform; liquid propellant
LPF = large payload fairing
LPV = launch preparation van
LRB = liquid rocket booster
LSB = launch and service building

-M-

MAB = missile assembly building
Marianne = Medium Ariane
MAS = mobile access structure
MAU = Million Accounting Units
MCC = mission control complex
MCO = Orlando International Airport, Orlando, Florida, USA
MDL = mission data load
MDU = master data unit
MECB = Missão Espacial Completa Brasileira (Complete Space Mission, Brazil)
MECO = main engine cutoff
MEDS = multifunction electronic display subsystem
MEO = medium Earth orbit
MIHT = Moscow Institute of Heat Technology
MIK = Montazhno-ispytatelnyj korpus (assembly-test complex)
MIS = motor inert storage
ML = mobile launcher
MLB = Melbourne Airport, Melbourne, Florida, USA
MLP = mobile launch platform
MMH = monomethyl hydrazine
MNTVC = movable nozzle thrust-vector control
MOD = Missions Operations Directorate

ABBREVIATIONS AND ACRONYMS

MOL = manned orbiting lab
MON-3 = 3% nitric acid/97% nitrogen textroxide
MPA = multiple payload adapter
MPF = medium payload fairing
MPS = moteur à propergol solide (Ariane 5 solid motor)
MRS = minimum residual shutdown
MSFC = NASA Marshall Space Flight Center
MST = mobile service tower
MTCR = Missile Technology Control Regime
MZ = Mashinostroitelny Zavod (machine building plant)

-N-

NAL = National Aerospace Laboratory
NAS = naval air station
NASA = U.S. National Aeronautics and Space Administration
NASDA = National Space Development Agency (Japan)
NATO = North Atlantic Treaty Organization
NCS = nutation control system
NCSR = National Committee on Space Research (India)
NGO = nongovernmental organization
NGP = navigation processor
NII = Nauchno-Issledovatelsky Institut (Scientifc Research Institute)
NISE = naval in-service engineering
NOAA = U.S. National Oceanographic and Atmospheric Administration
NORAD = North American Aerospace Defense Command
NPF = NAVSTAR processing facility
NPO = Nauchno-Proizvodstvennoe Ob'edinenie (Scientific Production Association)
NRO = U.S. National Reconnaissance Office
NUS = no upper stage

-O-

OAM = orbit adjust module
OBC = onboard computer
OCA = orbital carrier aircraft
O/F = oxidizer-to-fuel ratio
OKB = Osoboye Konstruktorskoye Buro (Special Design Bureau)
OMB = U.S. Office of Management and Budget
OMS = orbital maneuvering system
OPF = orbiter processing facility
OREX = orbital reentry vehicle
OSC = Orbital Sciences Corporation
OSP = Orbital/Suborbital Program
OV = orbital vehicle

-P-

PAA = payload attach assembly
PAF = payload attach fitting
PAL = propulseur d'appoint à liquid (Ariane 4 liquid rocket booster)
PALB = Palmachin Air Force Base (Israel)
PAM-D = payload assist module
PAP = propulseur d'appoint à poudre (Ariane 4 solid rocket booster)
PAS = propellant acquisition system
PCM = pulse code modulated
PCS = probability of command shutdown
PIF = polyisocyanurate foam
PLF = payload fairing
PLFPF = payload fairing processing facility
PM = payload module
PO = Proizvodstvennoe Ob'edinenie (Production Association)
POD = preliminary orbit determination
PPF = payload processing facility
PPG = programmable pulse generator
PS = preliminary satellite
PSACEI = propulsion subsystem for attitude control and engine ignition
PSLV = Polar Satellite Launch Vehicle
PST = pad service tower
PSWC = payload systems weight capability

-R-

RAKA = Rosaviakosmos (Russia)
RCS = roll control system; reaction control system
RCU = remote control unit
RD = Raketnay Dvigatel (rocket motor)
RDU = remote data unit
RESINS = redundant strapdown inertial navigation system
RF = radio frequency
RGU = rate gyro unit
RIFCA = redundant inertial flight control assembly
RIMU = redundant inertial mesurement unit
RIS = ready inspection building
RKA = Rossiiskoye Kosmicheskoye Agentstvo (Russian Space Agency)
RLCC = remote launch control center
RLV = reusable launch vehicle
R-PCM = remote pulse code moduclators
RP, RP-1 = Rocket Propellant 1 (high-grade kerosene)
RSC = Rocket System Corporation
RSLP = Rocket Systems Launch Program
RSRM = redesigned solid rocket motor
RSS = rotating service structure
RVSN = Strategic Missile Forces (Russia)

-S-

SAB = shuttle assembly building
SAC = Space Activities Council
SAS = segment arrival storage
SAEF = spacecraft assembly and encapsulation facility
SAS = add safeguard system
SAST = Shanghai Academy of Spaceflight Technology
SBIR = small business innovative research
SCA = système de contrôle d'attitude (attitude control system)
SCAT = spacecraft assemblies transfer
SCD = Satelite de Coleta de Dados (Brazil)
SD = smart dispenser
SEC = single-engine Centaur
SECO = second-stage engine cutoff
SELVS = small expendable launch vehicle services
SETA = Systems Engineering/Technical Assistance (TRW)
SFA = Spaceport Florida Authority
SHS = space head section
SIL = site d'integration des lanceurs
SITVC = secondary injection thrust vector control
SKB = Spetsializirovannoye Konstruktorckoye Byuro (Specialized Design Bureau)
SLBM = submarine-launched ballistic missile
SLC = space launch complex
SLV = space launch vehicle; standardized launch vehicle (Atlas); satellite launch vehicle
SLWT = super lightweight tank
SMAB = solid motor assembly building
SMARF = solid motor assembly and readiness facility
SMRC = solid motor for roll control
SMSJ = solid motor side jet
SMU = subsystem management unit
SpaceX = space exploration technologies
Spelda = strucutre porteuse externe pour lancements double Ariane
Speltra = structure porteuse externe pour lancement triple Ariane (Ariane 5 dual launch system)
SPL = Setor de Preparação e Lancamanto (Preparation and Launching Sector, Brazil)
SPROB = solid propellant space booster plant
SPM = standard payload module
SRB = solid rocket booster
SRM = solid rocket motor

Abbreviations and Acronyms

SRMU	=	solid rocket motor upgrade
SROSS	=	stretched Rohini-class satellites (India)
SRS	=	segment ready storage
SSB	=	solid strap-on boosters
SSC	=	Samara Space Center
SSI	=	Spaceport Systems International
SSLC	=	Scorpius Space Launch Company
SSLV	=	standard small launch vehicle
SSME	=	Space Shuttle Main Engine
SSPS	=	second-stage propulsion system
SSO	=	sun-synchronous orbit
SSRM	=	solid strap-on rocket motor
START	=	Strategic Arms Reductions Treaty
STS	=	Space Transportation System (U.S. Space Shuttle)
SVPF	=	space vehicle processing facility
SYLDA	=	systeme de lancement double Ariane

-T-

TAOS	=	thrust-assisted-orbiter system
TARVAN	=	transportation van
TDRSS	=	Tracking and Data Relay Satellite System
T/E	=	transport/erector
TEL	=	transportation/errector/launcher
TMS	=	telemetry subsystem
TPS	=	thermal protection system
TSLC	=	Taiyuan Satellite Launch Center
TsSKB	=	Tsentralnoe Spetsializirovannoye KB (Central Specialized Design Bureau)
TTC	=	telemetry, tracking, and telecommand; telemetry, tracking, and control
TVC	=	thrust-vector control

-U-

UDMH	=	unsymmetrical dimethyl hydrazine
UOHM	=	oxygen hydrogen module
UR	=	Universalnaya Raketa (universal rocket)
USA	=	United Space Alliance
USAF	=	U.S. Air Force
USN	=	U.S. Navy
UT	=	umbilical tower
UTC	=	universal coordinated time

-V-

VAB	=	vehicle assembly building
Vac	=	volts alternating current
VAFB	=	Vandenberg Air Force Base
Vdc	=	volts direct current
VEB	=	vehicle equipment bay
Vega	=	Vetlore Europeo di Generazioine
VEP	=	vehicle evaluation payload
VIB	=	vertical integration building
VKS	=	Russian Military Space Forces
VLM	=	Veiçulo Lançador de Microsatellies (Brazil)
VLS	=	Veiçulo Lançador de Satelites (Brazil)
VMC	=	vehicle mission computer
VPF	=	vehicle processing facility

-W-

WDF	=	water disposal facility

-X-

XSLC	=	Xichang Satellite Launch Center

-Y-

YF	=	yei-ti fa-dong-ji (Long March liquid engine)

REFERENCES

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